Modular Growth NTR Space Transportation System for Future NASA Human Lunar, NEA and Mars Exploration Missions

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The nuclear thermal rocket (NTR) is a proven, high thrust propulsion technology that has twice the specific impulse (I_{sp} ~900 s) of today's best chemical rockets. During the Rover and NERVA (Nuclear Engine for Rocket Vehicle Applications) programs, twenty rocket reactors were designed, built and ground tested. These tests demonstrated: (1) a wide range of thrust; (2) high temperature carbide-based nuclear fuel; (3) sustained engine operation; (4) accumulated lifetime; and (5) restart capability – everything required for affordable human missions beyond LEO. In NASA's recent Mars Design Reference Architecture (DRA) 5.0 study, the NTR was selected as the preferred propulsion option because of its proven technology, higher performance, lower IMLEO, versatile vehicle design, and growth potential. Furthermore, the NTR requires no large technology scale-ups since the smallest engine tested during the Rover program – the 25 klb_f "Pewee" engine is sufficient for human Mars missions when used in a clustered engine configuration. The "Copernicus" crewed Mars transfer vehicle developed for DRA 5.0 was an expendable design sized for fast-conjunction, long surface stay Mars missions. It therefore has significant propellant capacity allowing a reusable "1-year" round trip human mission to a large, high energy near Earth asteroid (NEA) like Apophis in 2028. Using a "split mission" approach, Copernicus and its two key elements – a common propulsion stage and integrated "saddle truss" and LH_2 drop tank assembly – configured as an Earth Return Vehicle / propellant tanker, can also support a short round trip (~18 month) / short orbital stay (60 days) Mars reconnaissance mission in the early 2030's before a landing is attempted. The same short stay orbital mission can be performed with an "all-up" vehicle by adding an "in-line" LH_2 tank to Copernicus to supply the extra propellant needed for this higher energy, opposition-class mission. To transition to a reusable Mars architecture, Copernicus' saddle truss / drop tank assembly is replaced by an in-line tank and "star truss" assembly with paired modular drop tanks to further increase the vehicle's propellant capacity. Shorter "1-way" transit time fast-conjunction Mars missions are another possibility using this vehicle configuration but, as with reusability, increased launch mass is required. "Scaled down" versions of Copernicus (sized to a SLS lift capability of ~70 t – 100 t) can be developed initially allowing reusable lunar cargo delivery and crewed landing missions, easy NEA missions (e.g., 2000 SG344 also in 2028) or an expendable mission to Apophis. Mission scenario descriptions, key vehicle features and operational characteristics are provided along with a brief discussion of NASA's current activities and its "pre-decisional" plans for future NTR development.
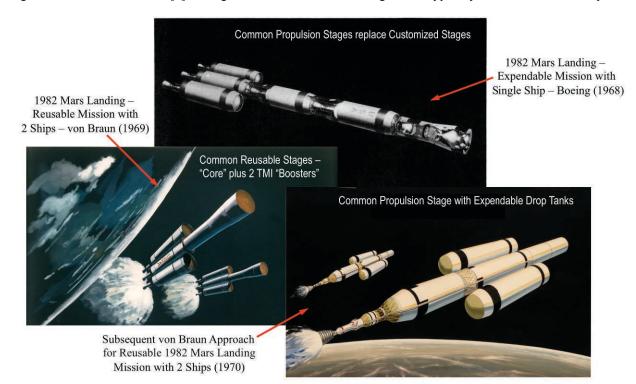
Nomenclature

SLS / HLV	=	Space Launch System / Heavy Lift Vehicle
IMLEO	=	Initial Mass in Low Earth Orbit
K	=	temperature (degrees Kelvin)
klb_f	=	thrust (1000's of pounds force)
LEO	=	Low Earth Orbit (= 407 km circular)
LOX / LH_2	=	Liquid Oxygen / Liquid Hydrogen propellant
t	=	metric ton (1 t = 1000 kg)
ΔV	=	velocity change increment (km/s)

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I. Introduction and Background

The use of common or "modular" elements in NTR Mars transfer vehicle (MTV) designs is not a new idea. Initial vehicle concept designs [1,2,3] developed for human Mars missions in the early 1960's started off with stages "customized" for a particular mission maneuver. Engine restart capability was not considered and engine burn times were limited to ~30 minutes. For short round trip/short stay time "opposition-class" missions, three expendable tandem stages, each with a 100 – 150 klb$_f$-class NERVA-derived engine, were used for the trans-Mars injection (TMI) maneuver and were then jettisoned. Separate expendable stages with lower thrust engines were used for the Mars orbit capture (~75 klb$_f$ for MOC) and trans-Earth injection (~50 klb$_f$ for TEI) maneuvers as well, with crew recovery utilizing direct ballistic capsule re-entry at the end of the mission.

Customized stages were later replaced by common stages [3] and then by a common propulsion module [4] with supplemental "in-line" and expendable LH$_2$ propellant "drop tanks" (Fig. 1) once engine burn times exceeding one hour, and multiple engine restarts were successfully demonstrated by the NERVA program's NRX-A6 and NRX-XE engine tests in 1967 and 1969 [5]. A single 100 klb$_f$-class NERVA engine was typically used with this MTV option.
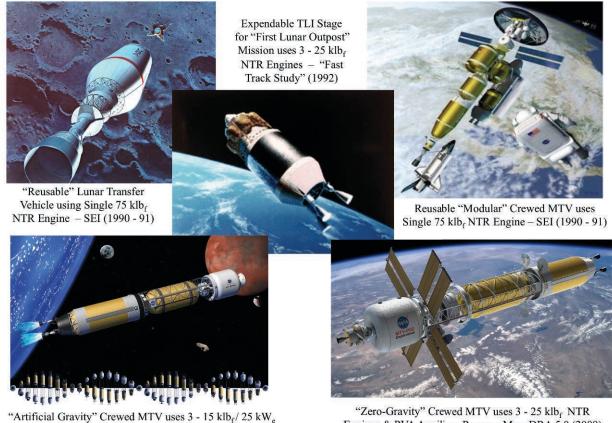

Figure 1.   Early NTR Mars Vehicle Design Transition from Expendable Stages to Expendable Drop Tanks

Less than a month after the successful landing of Apollo 11 on the Moon, Wernher von Braun, then director of NASA's Marshall Space Flight Center, outlined a plan for a human Mars landing in 1982 to the Senate Committee on Aeronautics and Space Science [6]. It involved two ships each using 100 klb$_f$-class NERVA engines and carrying 6 crewmembers. The initial ship design featured three common NTR propulsion modules – a central "core" module plus two reusable, tandem "booster" stages providing TMI assist. This design was later replaced by a single core propulsion module option with in-line and expendable drop tanks that supplied the additional LH$_2$ propellant required for the mission (Fig. 1). In NASA's proposed "post-Apollo" Integrated Space Program Plan (1970 – 1990), von Braun envisioned a reusable NTR propulsion stage that would function as a "workhorse" space asset delivering cargo and crew to the Moon initially for lunar base construction, and then for sending human missions to Mars [7,8].

Despite the technological triumph of Apollo, the public's interest in the program waned and Apollo flights 18, 19 and 20 were cancelled, along with NASA's plans for a post-Apollo program that envisioned the construction of a lunar base and a human mission to Mars in the early 1980's. After the final Apollo 17 mission to Taurus-Littrow in December 1972 and cancellation of the Rover/NERVA programs in January 1973, short of flight demonstration, interest in human Moon/Mars missions and NTP development remained relatively dormant for more than a decade.

On July 20, 1989, the 20th anniversary of Apollo 11, President Bush proposed a Space Exploration Initiative (SEI) for the United States, which called for a return to the Moon "to stay", followed by a journey to Mars [9]. From 1989 – 1993, NASA conducted and funded both in-house and industry studies [10,11,12] that outlined a campaign of human exploration that included the establishment of a transportation node in LEO, a permanent base on the Moon, then human missions to Mars. During SEI, NASA's Glenn Research Center (GRC), then known as the Lewis Research Center, quantified the benefits of NTP for both lunar and Mars missions. With its high thrust and specific impulse capability, NTP enabled a reusable lunar transfer vehicle (LTV) that could return itself and a single stage LOX/LH$_2$ lunar excursion vehicle (LEV) to LEO for refueling, refurbishment and reuse [13]. By adding modular propellant tanks to the LTV's "core" propulsion stage [14], a reusable MTV was developed for the reference mission – a 434 day round trip / 30 day stay "opposition-class" mission to Mars in 2016. Both the reusable NTP LTV and MTV designs used a single 75 klb$_f$ NERVA-derived NTR engine with "composite fuel" (shown in Fig. 2).

Figure 2. GRC's Lunar and Mars Transfer Vehicle Designs Have Transitioned Away From Large Single to Small Clustered Engines, Some Capable of Generating Electrical Power

In NASA's First Lunar Outpost (FLO) study [15], an expendable NTP trans-lunar injection (TLI) stage powered by three 25 klb$_f$ "Pewee-class" engines [5] was analyzed for sending a large (~96 t) integrated lander and ascent stage to the Moon. The FLO study marked the first use of smaller clustered NTR engines to help improve packaging and overall mission reliability. In 1999, GRC introduced the "Bimodal" NTR (BNTR) and "Artificial Gravity" (AG) crewed MTV design (Fig. 2) in NASA's Design Reference Mission (DRM) 4.0 study [16,17]. The AG MTV's "core" propulsion stage featured three 15 klb$_f$ BNTR engines that produced up to 50 kW$_e$ of electrical power needed to run the crewed MTV's subsystems during the mission. Vehicle rotation out to Mars and back produced a centrifugal force and AG environment to help maintain crew health and fitness. Ten years later, in NASA's recent Mars Design Reference Architecture (DRA) 5.0 study [18,19], non-bimodal NTR engines and a zero-gravity (0-g$_E$) crewed MTV design were again selected to provide an "apples-to-apples" comparison with chemical propulsion. A sampling of lunar and Mars transfer vehicle concepts developed by GRC over the last 2 decades is shown in Fig. 2.

NASA's recent Mars DRA 5.0 study [18] examined both short and long surface stay landing missions. The "fast conjunction" long surface stay option was selected for the design reference because it provided adequate time at Mars (~540 days) for the crew to explore the planet's rich geological diversity while also reducing the crew "1-way" transit times to and from Mars to ~6 months, or ~1 year in deep space. Long surface stay missions also have lower energy requirements than the short round trip time, short surface stay "opposition-class" missions, and therefore require less propellant and less mass delivered to LEO.

The NTR was the propulsion system of choice in DRA 5.0 because of its high thrust (10's of klb$_f$) and high specific impulse (I$_{sp}$ ~900 –950 s), its increased tolerance to payload mass growth and architecture changes, and its lower IMLEO which is important for reducing the HLV launch count, overall mission cost and risk. With a 100% higher I$_{sp}$ than today's best chemical rockets, the use of NTP reduced the required launch mass by over 400 t – the equivalent mass of the International Space Station (ISS). More importantly, the NTR is a proven technology and the only advanced propulsion option to be successfully ground tested at the performance levels required for a human mission to Mars. No large technology or performance scale-ups are needed as with other propulsion options. In fact, the smallest and highest performing engine tested during the Rover / NERVA programs [5] – the 25 klb$_f$ "Pewee" engine is sufficient for a human mission to Mars when used in a clustered engine arrangement.

DRA 5.0 featured a "split mission" approach using separate cargo and crewed Mars transfer vehicles. Both vehicle types utilized a common "core" propulsion stage each with three 25 klb$_f$ "composite fuel" Pewee-class engines. Two cargo vehicles were used to pre-deploy surface and orbital assets to Mars ahead of the crew who arrived during the next mission opportunity (~26 months later). The crewed MTV called "Copernicus" [19,20] is a 0-g$_E$ vehicle design consisting of three basic components: (1) the NTR propulsion stage; (2) the crewed payload element; and (3) an integrated "saddle truss" and LH$_2$ propellant drop tank assembly that connects the two elements.

Although Copernicus was operated in an "expendable mission mode" to reduce IMLEO as mandated in DRA 5.0, substituting an "in-line" LH$_2$ tank and "star truss" assembly with modular drop tanks for the saddle truss / drop tank assembly, can provide the additional propellant needed to operate Copernicus in a "reuse mode" assuming of course that LEO infrastructure (e.g., a transportation node / propellant depot with refueling capability) is in place to support reusability. Although designed as an expendable spacecraft, Copernicus was sized to allow it to perform all of the fast-conjunction missions over the 15-year synodic cycle. It therefore has significant propellant capacity that can be utilized for a variety of other mission applications currently under study by NASA and the international space exploration community. Smaller, "scaled-down" versions of the Copernicus spacecraft can also be configured as reusable lunar cargo transports as envisioned by von Braun, or as a reusable crewed Asteroid Survey Vehicle (ASV) like the "Search Lite" concept illustrated in Fig. 3.

The Global Exploration Roadmap (GER), developed by the International Space Exploration Coordination Group (ISECG), reflects the initial efforts of NASA and 13 other space agencies to define feasible and sustainable pathways for future human space exploration that includes the Moon, near-Earth asteroids, and eventually Mars, the long-term goal of the GER. The initial GER [21], released in September 2011, identified two possible pathways for future human missions after ISS utilization. These pathways have been designated the *"Moon Next"* and *"Asteroid Next"* scenarios. Both approaches utilize a stepwise development and demonstration of capabilities that are required for the eventual human exploration of Mars.

The *"Moon Next"* pathway has a strong appeal to many countries that would like to see humans again walk on its surface and to whom the Apollo program has become a distant memory. Located just 3 days from Earth, the Moon is an entire world awaiting exploration, future settlement and potential commercialization. It is also an ideal location to test and demonstrate key technologies and systems (e.g., surface habitation, long-range pressurized rovers, surface power and resource extraction systems) that will allow people to explore, work, and live self-sufficiently on another planetary surface. Crewed NEA missions would follow that demonstrate the additional in-space capabilities needed to reach Mars (e.g., advanced propulsion). Efficient propulsion and an affordable in-space transportation system with reuse capability will also be required if initial lunar outposts are to evolve into eventual settlements capable of supporting commercial activities.

The *"Asteroid Next"* pathway has as its focus the development and demonstration of key in-space exploration technologies and capabilities (e.g., reliable life support systems, long duration habitation and cryogenic fluids management, and advanced propulsion) necessary for traveling through and living in deep space. In addition to the scientific knowledge gained by an "up close and personal" examination of these primordial objects, NEA missions can also provide a proving ground for validating the spacecraft systems that will be needed for sending astronauts to Mars. In the GER, the ISECG shows the first crewed mission to a NEA beginning in 2028. Deep space asteroid missions as precursors to a human Mars mission is also consistent with the current United States' National Space Policy [22] that states NASA shall: *By 2025, begin crewed missions beyond the Moon, including sending humans to an asteroid. By the mid-2030s, send humans to orbit Mars and return them safely to Earth.*
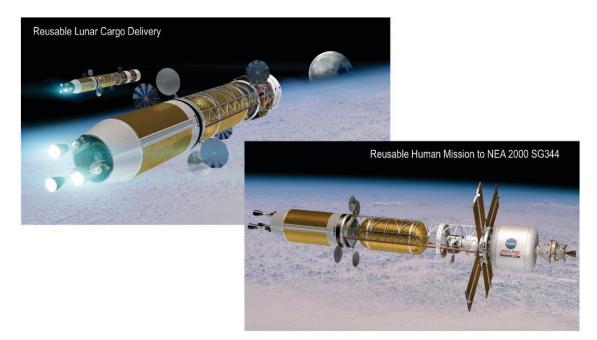
Figure 3. Reusable Lunar and NEA NTR Missions Possible using 70 t – 100 t-class SLS

Over the last several years, NASA's Human Architecture Team has pursued a strategy, referred to as a Capability Driven Framework (CDF), which assumes the utilization of evolving capabilities to pursue more demanding missions. With CDF, nearer-term technologies (chemical and solar electric propulsion) would be developed and demonstrated on less demanding missions (e.g., Earth-Moon L1, L2, Sun-Earth L2) first, before developing the real technologies and systems needed for more challenging lunar landing, NEA and Mars missions. Such an approach could be short-sighted and jeopardize NASA's ability to orbit Mars by 2035 by diverting resources away from proven technologies like NTP towards less capable systems that are large and operationally complex to use. Furthermore, a short round trip / short orbital stay mission to Mars is best performed in the 2033-2035 timeframe when the mission ΔV budgets are near their minimum values over the 15-year synodic cycle. After that, the ΔV budgets for successive short round trip missions increase significantly with the next minimum occurring in 2045.

This paper presents analysis that can support either the *"Asteroid Next"* or *"Moon Next"* pathways. It utilizes a "Technology Driven Framework" focused on developing and demonstrating the technologies and systems found in Copernicus' two key elements, its propulsion stage and integrated saddle truss/drop tank assembly, then validating them on NEA and lunar missions in the late 2020's/early 2030's in preparation for an orbital Mars mission in 2033. By focusing the resources of NASA and other space agencies on developing several key technologies and systems (the NTR, reverse turbo-Brayton refrigeration for zero-boiloff LH_2 storage, and large composite structures) and exploiting the technology synergies that exist between Copernicus, the HLV (e.g., large aluminum / lithium (Al/Li) LH_2 tanks) and existing flight-tested chemical rocket hardware (e.g., LH_2 turbopumps, regenerative- and radiation-cooled nozzles and skirt extensions), substantial savings in development time and cost are expected.

This paper examines alternative mission possibilities using the Copernicus MTV design and outlines a growth path using "modular" components that can increase its capability for more demanding missions. This same "modular growth" strategy can be applied to "scaled down" vehicle components configured for launch on the 70 t – 100 t SLS. The paper covers the following topic areas. First, the operational principles and performance characteristics of the baseline 25 klb$_f$ NTR engine used in this analysis are presented along with a summary of the Rover/NERVA programs' technical accomplishments. Mission and transportation system ground rules and assumptions used in the analysis are then presented along with a brief overview of the "7-Launch" Mars Mission Strategy for DRA 5.0 and a description of the Copernicus MTV design. Additional mission options for Copernicus along with increased capability made possible by using the modular growth strategy are discussed next. The paper's focus then turns to scaled down versions of Copernicus sized for reusable and expendable NEA missions, as well as, reusable cargo and crewed lunar missions and includes mission scenario descriptions, key vehicle features and operational characteristics. The paper ends with a brief discussion of NASA's current activities and future plans for developing NTP followed by a summary of our findings and some concluding remarks.

II. NTR System Description and Performance Characteristics

The NTR uses a compact fission reactor core containing 93% "enriched" Uranium (U)-235 fuel to generate 100's of megawatts of thermal power (MW_t) required to heat the LH_2 propellant to high exhaust temperatures for rocket thrust. In an "expander cycle" Rover/NERVA-type engine (Fig. 4), high pressure LH_2 flowing from twin turbopump assemblies (TPAs) cool the engine's nozzle, pressure vessel, neutron reflector, and control drums, and in the process picks up heat to drive the turbines. The turbine exhaust is then routed through the core support structure, internal radiation shield, and coolant channels in the reactor core's fuel elements where it absorbs energy produced from the fission of U-235 atoms, is superheated to high exhaust temperatures (T_{ex} ~2550 – 3000 K depending on fuel type and uranium loading), then expanded out a high area ratio nozzle (ε ~300:1 – 500:1) for thrust generation.

Controlling the NTR during its various operational phases (startup, full thrust and shutdown) is accomplished by matching the TPA-supplied LH_2 flow to the reactor power level. Multiple control drums, located in the reflector region surrounding the reactor core, regulate the neutron population and reactor power level over the NTR's operational lifetime. The internal neutron and gamma radiation shield, located within the engine's pressure vessel, contains its own interior coolant channels. It is placed between the reactor core and key engine components to prevent excessive radiation heating and material damage.
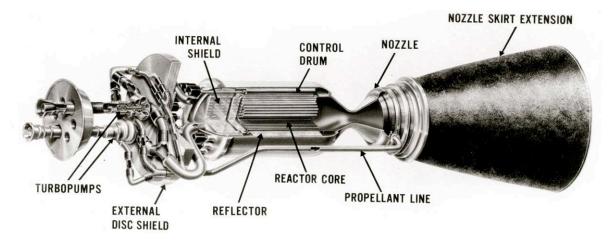

Figure 4. Schematic of "Expander Cycle" NTR Engine with Dual LH₂ Turbopumps

A Rover / NERVA-derived engine uses a "graphite matrix" material fuel element (FE) containing the U-235 fuel in the form of either coated particles of uranium carbide (UC_2) or as a dispersion of uranium and zirconium carbide (UC-ZrC) within the matrix material, referred to as "composite" fuel (shown in Fig. 5). The basic FE [5] has a hexagonal cross section (~0.75" across the flats), is 52" long and produces ~1 MW_t. Each FE has 19 axial coolant channels, which along with the element's exterior surfaces, are coated with ZrC using chemical vapor deposition (CVD) to reduce hydrogen erosion of the graphite. This basic shape was introduced in the KIWI-B4E and became the standard used in the 75 klb_f Phoebus-1B, 250 klb_f Phoebus-2A, 25 klb_f Pewee and the 55 klb_f NERVA NRX series of engines. These elements were bundled around and supported by cooled coaxial core support tie tubes. Six elements per tie tube were used in the higher power Phoebus and NRX reactor series. In the smaller Pewee engine, the ratio was reduced to three elements per tie tube. To provide sufficient neutron moderation and criticality in the smaller Pewee core, sleeves of zirconium hydride moderator material were added to the core support tie tubes (shown in Fig. 5).

The Rover program's 25 klb_f Pewee engine [5] was designed and built to evaluate higher temperature, longer life fuel elements with improved coatings, and in the process Pewee set several performance records. The Pewee full power test consisted of two 20-minute-long burns at the design power level of ~503 MW_t and an average fuel element exit gas temperature of ~2550 K, the highest achieved in the Rover/NERVA nuclear rocket programs. The peak fuel temperature also reached a record level of ~2750 K. Other performance records included average and peak power densities in the reactor core of ~2340 MW_t/m^3 and ~5200 MW_t/m^3, respectively. A new CVD coating of ZrC was also introduced and used in Pewee that showed performance superior to the niobium carbide (NbC) coating used in previous reactor tests.
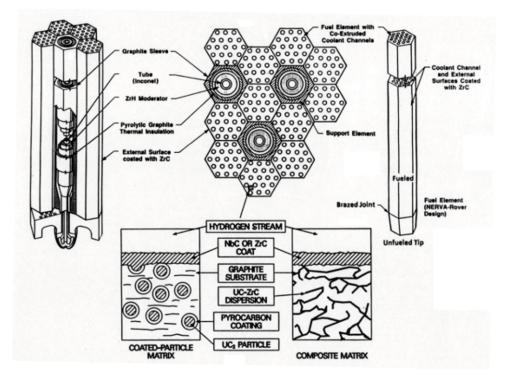
Figure 5. Coated Particle and Composite Rover / NERVA Fuel Element and Tie Tube Arrangement

In follow on tests in the "Nuclear Furnace" fuel element test reactor [5], higher temperature composite fuel elements with ZrC coating were evaluated. They withstood peak power densities of ~4500-5000 MW_t/m^3 and also demonstrated better corrosion resistance than the standard coated particle graphite matrix fuel element used in the previous Rover/NERVA reactor tests. Composite fuel's improved corrosion resistance is attributed to its higher coefficient of thermal expansion (CTE) that more closely matches that of the protective ZrC coating, thereby helping to reduce coating cracking. Electrical-heated composite fuel elements were also tested by Westinghouse in hot hydrogen at 2700 K for ~600 minutes – equivalent to ten 1-hour cycles. At the end of Rover / NERVA, composite fuel performance projections [23] were estimated at ~2-6 hours at full power for hydrogen exhaust temperatures of ~2500-2800 K and fuel loadings in the range of ~0.60 to 0.45 grams/cm^3. In addition to these carbide-based fuels, a ceramic-metallic or "cermet" fuel consisting of uranium dioxide (UO_2) in a tungsten (W) metal matrix was developed in the GE-710 and ANL nuclear rocket programs [24,25] as a backup to the Rover/NERVA fuel. While no integrated reactor/engine tests were conducted, a large number of fuel specimens were produced and exposed to non-nuclear hot H_2 and irradiation testing with promising results. Both fuel options are under development today.

The NTR engine baselined in DRA 5.0 and in this analysis is a 25 klb$_f$ "Pewee-class" expander cycle engine with the following performance parameters: T_{ex} ~2790 K, chamber pressure ~1000 psia, ε ~300:1, and I_{sp}~906 s. The LH$_2$ flow rate is ~12.5 kg/s and the engine thrust-to-weight ratio is ~3.50. The overall engine length is ~7.01 m, which includes an ~2.16 m long, retractable radiation-cooled nozzle skirt extension. The corresponding nozzle exit diameter is ~1.87 m. Recent Monte Carlo N-Particle (MCNP) transport modeling of the engine's reactor core [26], indicates that an I_{sp} range of ~894 s to 940 s is achievable by increasing the FE length from 0.89 m to 1.32 m and lowering the U-235 fuel loading in the core from ~0.45 to 0.25 grams/cm^3 which allows the peak fuel temperature to increase while still staying safely below the melt temperature.

Lastly, the state-of-the-art for NTP can be summarized as follows: It is a proven technology! A high technology readiness level (TRL~5-6) was demonstrated during the Rover / NERVA programs (1955-1972) [5]. Twenty rocket reactors were designed, built and ground tested in integrated reactor / engine tests that demonstrated: (1) a wide range of thrust levels (~25, 50, 75 and 250 klb$_f$); (2) high temperature carbide-based nuclear fuels that provided hydrogen exhaust temperatures up to 2550 K (*achieved in the Pewee engine*); (3) sustained engine operation (*over 62 minutes for a single burn on the NRX-A6*); as well as; (4) accumulated lifetime; and (5) restart capability (*>2 hours during 28 startup and shutdown cycles on the NRX-XE experimental engine*) – all the requirements needed for future human NEA and Mars exploration missions.

III. Mission and Transportation System Ground Rules and Assumptions

Specific mission and NTR transportation system ground rules and assumptions used in this paper are summarized in Tables 1 and 2, respectively. Table 1 provides information about the mission destinations, operational scenarios, and the assumed parking orbits at Earth, the Moon and Mars. Specific mission dates, trajectory details and ΔV budgets are provided within the appropriate sections of the paper. In addition to the large ΔV requirements for the primary propulsion maneuvers, like Earth orbit capture (EOC), smaller ΔV maneuvers are needed for rendezvous and docking (R&D) of vehicle components during the LEO assembly phase, for spacecraft attitude control during in-space transit, and for "split mission" Mars orbital operations involving R&D of the pre-deployed Earth Return Vehicle (ERV) with the "spent" crewed MTV for payload element transfer discussed in Section V.

Cargo delivery and crewed lunar landing missions are also considered in this paper. On lunar cargo flights, single or multiple habitat landers are delivered to LLO by a reusable NTR transport in a manner reminiscent of von Braun's reusable lunar NTR shuttle [7]. A single stage LOX/LH$_2$ LLV and Orion MPCV are transported to LLO on the crewed landing missions. The LLV is a "heritage" design [12] analyzed in considerable detail during SEI. It carried a crew of 4 plus 5 t of surface payload stored in two "swing-down" containers mounted on each side of the crew cab. The LLV mass breakdown with propellant loadings for a range of landed payload is shown in Table 1.

Table 1. Mission and Payload Ground Rules and Assumptions

Mission Destinations / Profiles:
• Moon – Cargo delivery and crewed lunar landing missions (starting in late 2020 timeframe)
• 2000 SG344 in 2028 (Small / Low Energy NEA)
• "Apophis" in 2028 (Large / High Energy NEA)
• Mars orbital mission in 2033 (Round trip time:
  - 545 days with 60 days in Mars orbit)
• Mars landing mission after 2033 (Round trip time:
  - 900 days with 540 days at Mars)
	• Lunar Missions: "All-Up" mission profile assumed for crewed mission; ~14 day stay time extendable with pre-deployed Habitat landers; "Reusable" scenario returns MPCV and LLV to 24-hr EEO
	• NEA Missions: "All-Up" mission profile is the baseline; no asset pre-deployment; ~1-year mission duration; "Reusable" scenario returns all payload elements (including the MMSEV) to an EEO
	• Mars Orbital Mission: "All-Up" mission profile; ~18 months mission duration; "Split mission" option pre-deploys ERV ahead of crew
	• Mars Landing Mission: Assumes DRA 5.0 mission profile
	• Orion capsule for crew recovery at Earth at mission end

Missions depart from low Earth orbit (LEO) and return to either a 6-hr or 24-hr elliptical Earth orbit (EEO) for reusable mission scenarios; capture and depart from either low lunar orbit (LLO) or a 24-hr elliptical Mars orbit (EMO)	• LEO: 407 km circular • 6-hr EEO: 500 km x 20,238 km • 24-hr EEO: 500 km x 71,136 km • LLO: 300 km circular • 24-hr EMO: 250 km x 33, 793 km
Lunar, NEA and Mars Mission ΔV Budgets:	• Mission dates, trip times & ΔV budget details provided in the paper
Additional ΔV Requirements: Advanced Material Bipropellant Rocket (AMBR) RCS thrusters used to perform non-primary propulsion maneuvers	• LEO R&D between orbital elements: ~15 – 100 m/s • Coast attitude control and mid – course correction: ~15 m/s and ~50 m/s, respectively • Mars orbit maintenance plus R&D: ~100 m/s
Lunar Cargo / Crewed Payload Masses: Reusable NTR cargo transports deliver Habitat landers to LLO; on crewed lunar landing missions, reusable NTR transport delivers Orion / MPCV and single stage LOX LH$_2$ Lunar Landing Vehicle (LLV) to LLO; LLV carries 4 crew and up to 5 t of payload to lunar surface; lunar missions return to a 24-hr EEO; crewed missions return the LLV, Orion MPCV and lunar samples	• Habitat lander: ~67 t – 81 t • LLV crew cab: 2.5 t • Crew (4) & EVA suits: 0.8 t • LLV descent /ascent stage: 6.1 t • LLV propellant load: 18.6 t – 20.9 t • LLV surface payload: 1.25 t – 5.0 t • Orion / MPCV: 13.5 t • Returned Samples: 0.1 t
Mars / NEA Crewed Payload Masses: Varies with crew size & mission duration; consumables based on a crew consumption rate of ~2.45 kg/person/day; payload also includes a short saddle truss (SST), a transfer tunnel with second docking module (TDM) an an exterior consumables container; for NEA missions, the exterior container is replaced with a MMSEV for "close up" inspection and sample gathering	• Transit Habitat: 22.7 t – 27.5 t (minus consumables) • SST/TDM/Container: 2.89 t – 5.08 t / 1.76 t / 23% of stored food • Crew (4-6): 0.4 t – 0.6 t • Total Consumables: 3.58 t –5.37 t (4 – 6 crew for 1-yr); extra consumables stored in exterior container • MMSEV: 6.7 t • Orion / MPCV: 10.0 t – 13.5 t • Returned Samples: 0.1 t (NEA); 0.25 t – 0.5 t (Mars)

For crewed NEA and Mars missions, the mass of some key payload elements, like the transit habitat, varies with crew size, mission destination and duration. With increasing crew size and mission duration, extra life support, food and accommodations are needed. For missions exceeding ~1 year in duration, like Mars, additional food supplies are carried in a consumables container attached to the primary TransHab module via a transfer tunnel. The container, along with unneeded mass, can then be jettisoned prior to the TEI maneuver to reduce propellant consumption. Both the container and transfer tunnel are enclosed within a short saddle truss that connects the transit habitat to the rest of the spacecraft. The short saddle truss has the same diameter as the long saddle truss that is sized by the diameter of the transfer vehicle's LH_2 drop tank. The mass of both truss segments therefore increases with tank diameter.

Fixed mass payload elements include the Orion Multi-Purpose Crew Vehicle (MPCV) and an auxiliary multi-mission space excursion vehicle (MMSEV) carried on NEA missions. The MMSEV provides a small livable volume for a crew of two for up to two weeks [27] and is attached to the TransHab module via the same transfer tunnel (shown in Fig. 6). The MMSEV provides ~200 - 300 m/s of translational ΔV, suitports for quick EVA capability, and remote manipulation capability for sample collection. For lunar and NEA missions analyzed here, it is assumed that the crew collects and returns ~100 kg of samples. For Mars DRA 5.0, ~250 kg – 500 kg of samples are returned.
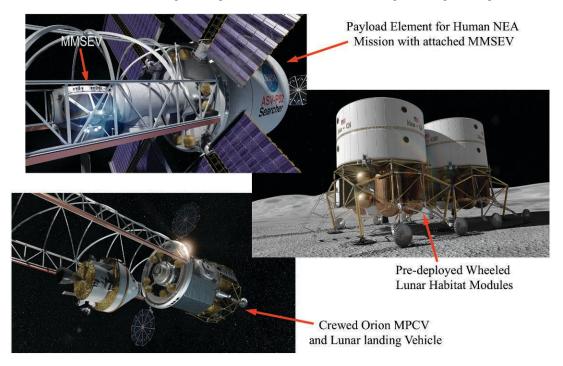

Figure 6. Payload Elements Delivered by NTP NEA and Lunar Transfer Vehicles

Table 2 lists the key transportation system ground rules and assumptions. The NTR engine and fuel type, operating characteristics, and thrust levels examined are summarized first. A three-engine cluster of 25 klb_f "Pewee-class" engines is baselined although lower thrust 15 klb_f engines (length ~5.3 m, engine thrust-to-weight ratio ~3.1) are also considered in the paper. All engines use composite fuel with a U-235 fuel loading of 0.25 g/cm^3. With a hydrogen exhaust temperature (T_{ex}) of ~2790 K, and a nozzle area expansion ratio of ~300:1, the I_{sp} is ~906 s with higher I_{sp} values achievable by increasing the fuel operating temperature. The total mission LH_2 propellant loading consists of the usable propellant plus performance reserve, post-burn engine cooldown, and tank-trapped residuals. For the smaller auxiliary maneuvers, a storable bipropellant RCS system is used. All transfer vehicle configurations utilize a "split RCS" with 16 of 32 AMBR thrusters and approximately half of the bipropellant mass located on the rear propulsion stage and the forward most saddle truss adaptor ring just behind the mission-specific payload.

The LH_2 propellant carried by the various vehicles is stored in the same "state-of-the-art" Al/Li LH_2 propellant tank being developed for the SLS / HLV that will support future human exploration missions. For this analysis, tank sizing assumes a 30 psi ullage pressure, 5 g_E axial / 2.5 g_E lateral launch loads, and a safety factor of 1.5. A 3% ullage factor is also assumed. All tanks use a combination foam / multilayer insulation (MLI) system for passive thermal protection. A zero boil-off (ZBO) "reverse turbo-Brayton" cryocooler system is used on the NTR propulsion

Table 2. NTR Transportation System Ground Rules and Assumptions

NTR System Characteristics	• Engine / Fuel Type: NERVA-derived / UC-ZrC "Composite" • Propellant: LH_2 • Thrust Level: 15 klb_f – 25 klb_f ("Pewee-class" is baseline) (3 engine cluster on "Core" Propulsion Stage) • Fuel Element Length: 0.89 m – 1.32 m (1.32 m FE is the baseline) • Exhaust Temp: T_{ex} ~2731 – 2940 K • Chamber Pressure: p_{ch} ~1000 psi • Nozzle Area Ratio: ε~300:1 • I_{sp} Range: 906 s (2790 K) – 941 s (2940 K)
Propellant Margins	• Cooldown: 3% of usable LH2 propellant • Performance reserve: 1% on ΔV • Tank trapped residuals: 2% of total tank capacity
Reaction Control System (LEO R&D, Settling, Attitude Coast Control, and Mid-course Correction Burns)	• Propulsion Type: AMBR 200 lb_f thrusters • Propellant: NTO / N_2H_2 • Nominal I_{sp}: 335 seconds
LH_2 Cryogenic Tanks and Passive Thermal Protection System (TPS)	• Material: Aluminum-Lithium (Al/Li) • Tank OD: 7.6 m – 10.0 m • Tank L: ~15.7 m – 19.7 m ("core" propulsion stage) ~15.7 m – 22.7 m ("in-line" drop tank) • Geometry: cylindrical with root 2/2 ellipsoidal domes • Insulation: 1" SOFI (~0.78 kg/m^2) + 60 layers of MLI (~0.90 kg/m^2)
Active Cryo-Fluid Management / Zero Boil-Off (ZBO) LH_2 Propellant System	• Reverse turbo-Brayton ZBO cryocooler system powered by PVAs • ZBO system mass and power requirements driven by core stage size; ~760 kg and ~5.26 kW$_e$ (7.6 m D) – 930 kg and ~8.87 kW$_e$ (10.0 m D)
Photovoltaic Array (PVA) Primary Power System	• Circular PVA sized for ~7 kW$_e$ at 1 A.U, two arrays provide power for ZBO cryocoolers on core stage and in-line tank if needed, PVA mass is ~455 kg & array area is ~25 m^2; to supply 1 kW$_e$ at Mars requires ~10 m^2 of array area • "Keep-alive" power supplied by lithium-ion battery system
Dry Weight Contingency Factors	• 30% on NTR system and composite structures (e.g., saddle truss) • 15% on established propulsion, propellant tanks, spacecraft systems
SLS / HLV Launch Requirements: – Lift Capability to LEO – Launch Shroud Size (D x L) – Cylindrical Payload (PL) Envelope	• ~70 t – 140 t; NTR propulsion stage with external crew radiation shields • 8.6 m D x 33 m L – 12 m D x 42.5 m L • 7.6 m –10 m D x ~25.1 m – 33.8 m L (DRA 5.0 crewed PL element)

stages and "in-line" LH_2 tanks (where required) to eliminate boil-off during LEO assembly and the remainder of the mission which can be months to years in duration. The propellant tank heat load is largest in LEO and sizes the ZBO cryocooler system. Solar photovoltaic arrays are baselined for supplying the primary electrical power needed for all key transfer vehicle subsystems. Because of the decreased solar radiation at Mars (~486 W/m^2), array areas can become quite large (~10 m^2/kW$_e$) necessitating multiple arrays for human Mars missions.

Table 2 also provides the assumed "dry weight contingency" (DWC) factors, along with HLV lift and shroud payload envelope requirements. A 30% DWC is used on the NTR system and advanced composite structures (e.g., stage adaptors, trusses) and 15% on heritage systems (e.g., Al/Li tanks, RCS, etc.). The transfer vehicle propulsion stage and mission payload elements drive the SLS / HLV lift capability and shroud size, respectively. For a human Mars mission, the crewed payload (PL) element includes the "packaged" TransHab module with PVA power system, the short saddle truss, consumables container and transfer tunnel with secondary docking module, and the Orion MPCV (see Figs. 7 and 8). On NEA missions, the consumables container is replaced with a MMSEV. Lunar payloads include single or multiple habitat landers delivered to LLO on cargo missions, or a single stage LLV and Orion MPCV delivered on crewed missions (depicted in Fig. 6). The PL envelope's diameter varies from ~7.6 m – 11 m and its length can be up to ~33.8 m (for DRA 5.0 PL including the attached MPCV). The propulsion stage mass drives the SLS / HLV lift requirement which can vary from ~70 – 140 t depending on the particular mission.

IV. Mars DRA 5.0: "7-Launch" NTR Mission Overview

The 7-Launch NTR Mars mission strategy [19] for a human landing mission is illustrated in Fig. 7 and is centered around the long surface stay, split cargo / piloted mission approach. Two cargo flights pre-deploy a cargo lander to the surface and a habitat lander into Mars orbit where it remains until the arrival of the crew on the next mission opportunity (~26 months later). The cargo flights utilize "1-way" minimum energy, long transit time trajectories. Four HLV flights carried out over 90 days (~30 days between launches), deliver the required components for the two cargo vehicles. The first two launches deliver the NTR propulsion stages each with three 25 klb$_f$ NTR engines. The next two launches deliver the cargo and habitat lander payload elements which are enclosed within a large triconic-shaped aeroshell that functions as a payload shroud during launch, then as an aerobrake and thermal protection system during Mars aerocapture (AC) and subsequent entry, descent and landing (EDL) on Mars. Vehicle assembly involves Earth orbit rendezvous and docking (R&D) between the propulsion stages and payload elements with the NTR stages functioning as the active element in the R&D maneuver.

Once the operational functions of the orbiting habitat and surface cargo landers are verified, and the Mars Ascent Vehicle (MAV) is supplied with ISRU-produced ascent propellant, the crewed MTV is readied and departs on the next mission opportunity. As mentioned previously, the Copernicus MTV is a 0-g$_E$ vehicle design that utilizes a fast conjunction trajectory with ~6 month "1-way" transit times to and from Mars. Like the cargo MTV, it is an "in-line" configuration that uses Earth orbit R&D to simplify vehicle assembly. It uses the same "common" NTR propulsion stage but includes additional external radiation shielding on each engine for crew protection during engine operation. Three HLV launches over 60 days are used to deliver the vehicle's key components which include: (1) the NTR propulsion stage; (2) integrated "saddle truss" and LH$_2$ drop tank assembly; and (3) supporting crewed payload. The crewed payload component includes the TransHab module with its six crew, a long-lived MPCV for vehicle-to-vehicle transfer and "end of mission" Earth entry, a secondary T-shaped docking module (DM), contingency consumables container and connecting structure. Four 12.5 kW$_e$ / 125 m^2 rectangular PVAs provide ~50 kW$_e$ of electrical power at Mars for crew life-support (~30 kW$_e$), a ZBO Brayton cryocooler system (~10 kW$_e$), and high data- rate communications (~10 kW$_e$) with Earth. When assembly is completed, the Mars crew is delivered
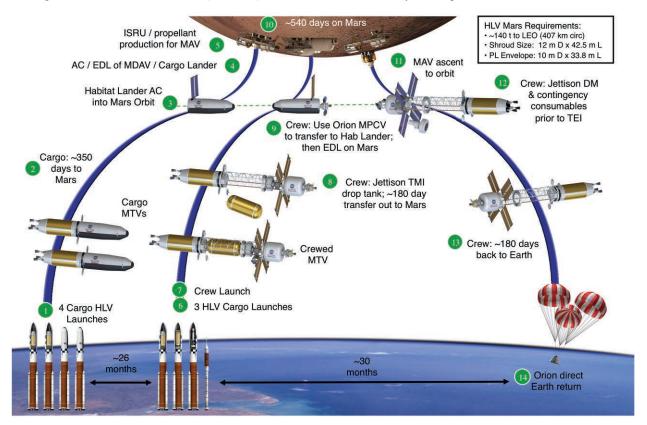

Figure 7. DRA 5.0 Long-Stay Mars Mission Overview: "7-Launch" NTR Strategy

to Copernicus and docks on its underside using the secondary DM that connects the TransHab crew module and contingency consumables container (shown in Figs. 8 and 9).

Following the TMI maneuver, the drained LH_2 drop tank, attached to the saddle truss, is jettisoned and the crewed MTV coasts to Mars under 0-g_E conditions with its four PVAs tracking the Sun. Attitude control and mid-course correction maneuvers are provided by Copernicus' split RCS that uses 200 lb_f storable bipropellant AMBR (Advanced Material Bipropellant Rocket) thrusters located on the rear NTR propulsion module and the short saddle truss forward adaptor ring just behind the TransHab module. After the MOC burn, Copernicus rendezvouses with the orbiting Hab lander using engine cooldown thrust and the vehicle's RCS. The crew then transfers over to the lander using the MPCV. After crew transfer, the MPCV returns and docks to the TransHab autonomously. The crew then initiates EDL near the cargo lander and begins the surface exploration phase of the mission. After ~533 days on the surface, the crew lifts off using the MAV and returns to Copernicus using its secondary DM (shown in Fig. 8). Following the transfer of the crew and samples, the MAV is jettisoned. The crew then begins a weeklong checkout and verification of all MTV systems, jettisons the DM and contingency consumables and performs the TEI burn to begin the journey back to Earth. After an ~6 month trip time, the crew enters the MPCV, separates from the MTV and subsequently re-enters the atmosphere while Copernicus flies by Earth at a "sufficiently high altitude" and is disposed of into heliocentric space. Although Copernicus was operated in an "expendable mission mode" in DRA 5.0 to reduce total IMLEO, it can readily be modified for operation in a "reusable mode" by providing the vehicle with additional propellant capacity as discussed in the next section.

Figure 8. Copernicus' Secondary DM Provides Access to the MPCV and MAV during the Mission

The "Copernicus" crewed MTV has an overall length of ~93.7 m (Fig. 9) and an IMLEO of ~336.5 t consisting of: (1) the NTP stage or NTPS (~138.1 t); (2) the saddle truss and LH_2 drop tank (133.4 t); and (3) the crew payload section (~65 t). The NTPS uses a three-engine cluster of 25 klb_f NTR engines and also carries additional external radiation shield mass (~7.3 t) for crew protection. The NTPS uses an Al/Li LH_2 tank size which has a diameter (D) and length (L) of 10 m D x 19.7 m L. The tank's LH_2 propellant capacity is ~87.2 t. The NTPS also carries avionics, RCS, auxiliary battery and PVA power, docking and Brayton-cycle ZBO refrigeration systems located in the forward cylindrical adaptor section. To remove ~78 watts of heat penetrating the 60 layer MLI system in LEO (where the highest tank heat flux occurs), the reverse turbo-Brayton cryocooler system needs ~8.9 kW_e for its operation (~114 W_e for each W_t removed). Twin circular PVAs on the NTPS provide the electrical power for the ZBO system in LEO until the four primary PVAs on the crewed PL section are deployed prior to TMI.

Copernicus' second major component is its saddle truss and LH_2 drop tank assembly. The saddle truss is a rigid, spine-like composite structure that wraps around the upper half of the LH_2 drop tank and connects the NTR stage to the forward payload section. It is ~27.7 m long and has a mass of ~9 t. The saddle truss is open underneath allowing the drained LH_2 drop tank to be jettisoned after the TMI burn is completed. The ~22.7 m long LH_2 drop tank has a mass of ~22 t and carries ~102.4 t of propellant.
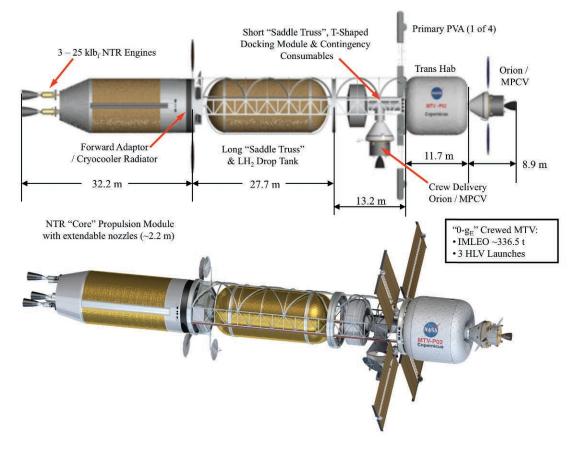

Figure 9. Key Features and Component Lengths of the Copernicus NTR Mars Transfer Vehicle

Copernicus' third and final component is its payload section. In DRA 5.0 it was designed for launch as a single integrated unit and thus determines the overall shroud size. The integrated payload element is ~33.8 m long and includes the short saddle truss, "T-shaped" docking module (DM) and transfer tunnel, consumables container, TransHab and Orion MPCV. The DM provides "secondary access" to Copernicus for the crew delivery MPCV and the MAV (see Figs. 8 and 9). Following the crew's return from Mars and MAV separation, the DM and attached consumables container are both jettisoned to reduce vehicle mass prior to TEI (see Fig. 7).

The total crewed payload mass at TMI is ~65 t distributed as follows: (1) short saddle truss (~5.1 t); (2) DM and transfer tunnel (~1.8 t); (3) contingency consumables and jettisonable container (~9.7 t); (4) TransHab with its primary PVAs (~27.5 t); (5) transit consumables (~5.3 t); (6) crew (~0.6 t); (7) MPCV (~10 t); and (8) forward RCS and propellant (~5 t). The crewed MTV's total RCS propellant loading is ~9.1 t with the "post-TMI" RCS propellant load split between the NTPS (~5.1 t) and the short saddle truss forward cylindrical adaptor ring (~4 t).

Lastly, for DRA 5.0, the performance requirements on operating time and restart for Copernicus' $3 - 25$ klb_f NTR engines are quite reasonable. For the round trip mission, there are 4 primary burns (3 restarts) that use ~178.4 t of LH_2 propellant. With 75 klb_f of total thrust and a I_{sp} of ~906 s, the total engine burn time for the mission is ~79.2 minutes (~55 minutes for the "2-perigee burn" TMI maneuver, ~14.5 minutes for MOC, and ~9.7 minutes for TEI), well under the ~2 hour accumulated engine burn time and 27 restarts demonstrated by the NERVA eXperimental Engine [5] – the NRX-XE in 1969.

V. Modular "Growth" Options and Alternate Mission Applications for Copernicus

In DRA 5.0, the Copernicus MTV was sized to allow it to perform all of the fast conjunction missions over the 15-year synodic cycle (~2030 – 2045 timeframe) with transit times to and from Mars ranging from ~150 – 210 days. The baseline mission trajectory, trip times and ΔV budget details for the DRA 5.0 "7-Launch" NTR strategy included the following: Earth departure C_3 ~18.40 km^2/s^2, ΔV_{TMI} ~3.992 km/s, outbound transit time ~180 days, arrival V_{inf} ~4.176 km/s, and ΔV_{MOC} ~1.771 km/s, stay time at Mars ~540 days, Mars departure C_3 ~14.80 km^2/s^2, ΔV_{TEI} ~1.562 km/s, inbound transit time ~180 days, and total mission duration ~900 days. For 180-day transit missions returning to a 24-hr EEO, the capture ΔV_{EOC} is ~1.855 km/s. (Gravity loss is also added to the ideal ΔV values shown). As mentioned previously, the "3-element" Copernicus spacecraft (Fig. 10 – Configuration 1) has significant propellant capacity that can be exploited for a variety of other mission applications. These include human missions to large, high energy NEAs, as well as, reusable lunar cargo delivery and crewed lunar landing missions. Mission details and vehicle characteristics for these applications are provided in later sections of this paper.
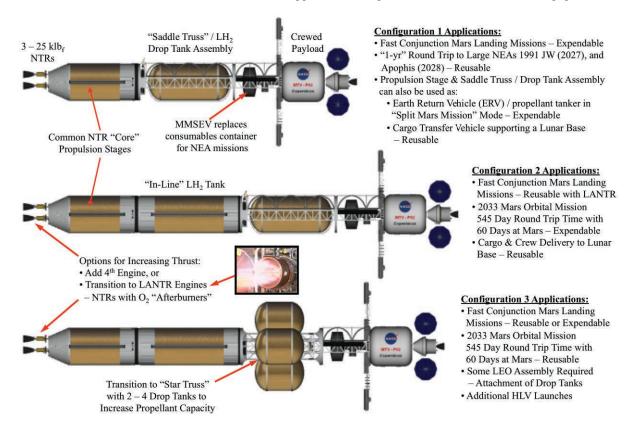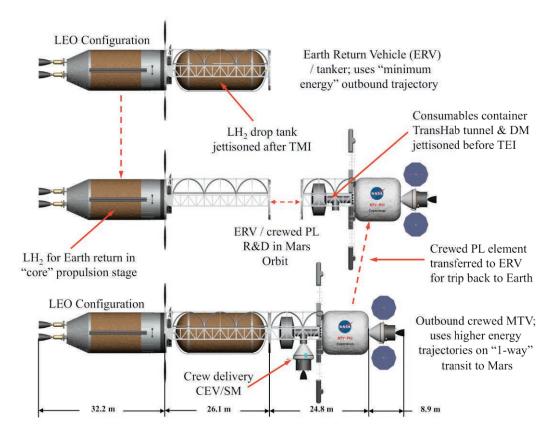

Figure 10. Growth Paths and Alternative Missions for Copernicus Spacecraft using Modular Components

Currently, the United States' National Space Policy states that NASA shall: *By the mid-2030s, send humans to orbit Mars and return them safely to Earth.* Short round trip / short orbital stay "opposition" missions to Mars have higher energy requirements than those in DRA 5.0 and are best performed in the 2033 – 2035 timeframe when the mission ΔV budgets are near their minimum values over the 15-year synodic cycle. After that the ΔV budgets for successive short round trip missions increase significantly necessitating larger quantities of propellant and extra spacecraft mass being launched in order to perform the mission. Using the Copernicus spacecraft and its two key components – the NTPS and integrated saddle truss / drop tank assembly – configured as an Earth Return Vehicle (ERV) / propellant tanker, a "split cargo / crewed mission" approach can be utilized for an initial Mars orbital survey mission before a human landing mission is attempted. The split mission option can also eliminate the additional time and cost to develop the extra vehicle components needed for an "all up" mission. Use of a pre-deployed ERV is not a new idea. In NASA's DRM 1.0 study in 1993 [14], the ERV was one of the key transportation system elements used in the mission architecture.

2033 Short Round Trip / Short Orbital Stay Expendable "Split Mars Mission" Option:

Using the Copernicus spacecraft, outfitted as a Mars Survey Vehicle (MSV), plus an ERV / propellant tanker, a 545-day round trip / 60-day stay crewed mission to Mars is possible in 2033 using the split mission approach outlined in Fig. 11. The ERV is pre-deployed to Mars orbit in advance of the crew. It departs from LEO in December 2030 (departure C_3 ~10.794 km^2/s^2, ΔV_{TMI} ~3.662 km/s), on a 283-day "minimum-energy" trajectory out to Mars. The ERV then arrives at Mars (arrival V_{inf} ~3.480 km/s) and propulsively captures into a "24-hour" elliptical parking orbit (~250 km x 33,793 km, ΔV_{MOC} ~1.34 km/s) in October 2031 where it remains until the crewed MSV, called "Searcher", arrives on the next opportunity ~2 years later.

Searcher departs LEO with its 6 crew in May 2033 (departure C_3 ~14.62 km^2/s^2, ΔV_{TMI} ~3.83 km/s). It uses a high energy, opposition trajectory out to Mars arriving ~159 days later (arrival V_{inf} ~3.79 km/s), then propulsively captures (ΔV_{MOC} ~1.53 km/s) into the same parking orbit as the ERV in October 2033. In the process, Searcher uses a substantial percentage of its available propellant. To return to Earth, Searcher rendezvouses with the ERV and the forward crewed PL element is switched over to the ERV (shown in Fig. 12). No propellant transfer is required just a R&D maneuver. Before the ERV performs the TEI maneuver (ΔV_{TEI} ~3.12 km/s), the exterior consumables container and connecting tunnel (~6.2 t) are jettisoned from the PL element to reduce propellant consumption. The ERV utilizes an inbound Venus swing-by during the 326-day transfer back to Earth. At mission end, the crew re-enters using the Orion MPCV capsule, while Searcher flies by Earth and is disposed of into heliocentric space.

Figure 11. Copernicus MTV & Components Configured for Short Round Trip "Split" Mars Orbital Mission

For its outbound Mars transit, Searcher has an IMLEO of ~251.1 t consisting of the NTPS (~107 t), the integrated saddle truss / LH$_2$ drop tank assembly (~84.3 t), and the crewed PL element (~59.8 t). The NTPS and ~21.1 m long LH$_2$ drop tank are substantially off-loaded in propellant (~ 66.8% and 59.6%, respectively), with ~114.5 t of LH$_2$ propellant carried on the outbound mission leg (~63% of the maximum available capacity of ~181.7 t). In addition to 3 restarts, the total burn time on Searcher's three 25 klb$_f$ engines is ~47.7 minutes, substantially lower than that needed for DRA 5.0, and well below the capabilities demonstrated on the NRX-XE.
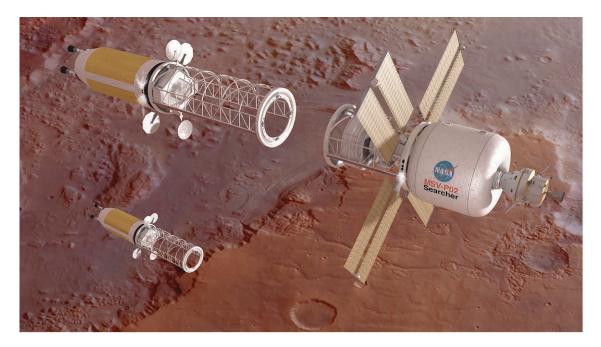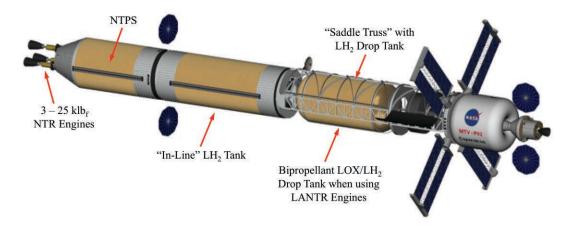
Figure 12.   "Switch-over" - Crewed PL Element Transfer to ERV in Mars Orbit for Trip Back to Earth.

The "round trip" ERV / tanker has an IMLEO of ~237.4 t consisting of NTR propulsion stage (~127.6 t) and the integrated saddle truss / LH_2 drop tank assembly (~109.8 t). The ERV carries a larger LH_2 propellant loading in its propulsion stage and forward drop tank totaling ~153.6 t (~84.5% of the maximum available capacity of ~181.7 t) which is needed to return Searcher's crewed PL back to Earth. In addition to 3 restarts, the total burn time on the ERV's three 25 klb$_f$ engines is ~64.2 minutes (~35.2 minutes for the "2-perigee" burn TMI maneuver, ~7.9 minutes for MOC, and ~21.1 minutes for TEI), again well below the capabilities demonstrated on the NRX-XE. The ERV's longer TEI burn duration is attributed to the addition of the Searcher's ~51.2 t crewed PL plus the higher TEI ΔV requirement (~3.12 km/s versus ~1.56 km/s for DRA 5.0). Lastly, the total mission IMLEO for the Searcher MSV and its ERV is ~488.5 t.

2033 Short Round Trip / Short Orbital Stay "All Up" Expendable Mars Mission Option:

The same 18 month round trip / 60-day stay Mars orbital mission discussed above can be performed using a single vehicle by positioning an "in-line" LH_2 tank between the propulsion stage and integrated saddle truss / drop tank assembly to provide additional propellant capacity (Fig. 10 – Configuration 2 and Fig. 13). This "all up" configuration has an IMLEO of ~429.4 t consisting of the "wet" NTPS (~134.4 t), in-line tank (~117.7 t), saddle truss / drop tank assembly (~117.7 t) and the crew PL section (~59.6 t). The overall vehicle length is ~117 m including the Orion MPCV at ~8.9 m. The LH_2 loading in the propulsion stage, in-line and drop tanks are ~87.1 t, ~88.9 t, and ~89.1 t, respectively. The LH_2 tank length in the NTPS is ~19.7 m. The in-line and drop tank lengths are the same at ~20 m. With these equal tank lengths, the masses of in-line tank element and the saddle truss / drop tank assembly are balanced at ~117.7 t. For this expendable "all up" Mars orbital mission, there are 4 primary burns (with 3 restarts) and the LH_2 propellant used for the mission is ~250 t. With 75 klb$_f$ of total thrust and a I_{sp} of 906 s, the total engine burn time for this mission is ~111 minutes. The first TMI perigee burn is the longest single burn at ~41.3 minutes. After this burn, the drop tank is drained and then jettisoned to reduce vehicle mass and propellant consumption during the second perigee burn (~27.6 minutes). The in-line tank and NTPS supply the propellant for the remaining MOC (~16.8 minutes) and TEI burns (~25.3 minutes). Compared to the split mission approach, the all up option requires one less HLV launch but it also requires development of the in-line tank element with its own ZBO cryocooler system. The total burn time requirement on the engines is also larger as is the maximum single burn duration which increases by ~64% (from ~25.2 minutes for Searcher's first perigee burn to ~41.3 minutes for this same maneuver). Adding a fourth engine would help reduce total mission burn time and gravity losses (~380 m/s) during the TMI maneuver but also increases NTPS inert mass at the expense of LH_2 propellant loading.

Figure 13. Configuration 2: Supports Expendable Short Stay or Reusable DRA 5.0 Missions using LANTR

Reusable DRA 5.0 Copernicus Option 1 (3 – LANTR Engines, 24-hr EEO capture $\Delta V_{EOC} \sim 1.855$ km/s):

There are two possible options for converting the Copernicus MTV from an expendable to a reusable spacecraft. In Option 1 (Fig. 13), the LH_2 drop tank is replaced by a bipropellant (LOX/LH_2) tank and oxygen "afterburner" nozzles are added to Copernicus' three NTR engines to help increase the vehicle's total thrust output and shorten the engine burn time during Earth departure. In the "LOX-Augmented" NTR (LANTR) option [28,29], oxygen is injected into the divergent section of the nozzle downstream of the sonic throat. Here it mixes with reactor-heated H_2 and undergoes supersonic combustion adding both mass and chemical energy to the engine's exhaust. By operating the LANTR engines with an oxygen-to-hydrogen mixture ratio (MR) = 1 during the first TMI perigee burn, the engine's thrust level is increased by over 62% – from 25 klb$_f$ to ~40.6 klb$_f$. The addition of oxygen at this MR, however, lowers the I_{sp} to ~726 s. To limit the number of HLV launches to 4, crew size is reduced from 6 to 4 thereby lowering the requirements on TransHab mass and mission consumables. The drained bipropellant drop tank is also jettisoned after the first perigee burn and all subsequent maneuvers use straight NTP and LH_2 propellant. The engines' composite fuel is also run at higher temperature to achieve a I_{sp} of ~935 s.

The IMLEO for Option 1 is ~466.6 t consisting of the "wet" NTPS (~136 t), in-line tank (~138 t), saddle truss / drop tank assembly (~138.6 t) and the crew PL section (~54 t). The overall vehicle length is ~116.3 m including the 8.9 m long Orion MPCV. The propellant loading in the propulsion stage, in-line and drop tanks are ~87.2 t, ~105.5 t, and ~111.6 t, respectively. The bipropellant drop tank includes ~58.1 t of LOX and ~53.5 t of LH_2. For this "all up" reusable Mars orbital mission, there are 5 primary burns (4 restarts). During the first perigee burn with LANTR "afterburners", the total vehicle thrust is increased to ~121.8 klb$_f$. With ~112.4 t of LOX/LH_2 propellant consumed and a I_{sp} of ~726 s, the burn duration is ~24.6 minutes. The duration of the remaining burns using straight NTP (total thrust ~75 klb$_f$, I_{sp} ~935 s) is as follows: second perigee burn (~31.9 minutes), MOC (~20.5 minutes), TEI (~14.4 minutes) and EOC (~13.8 minutes). The total engine burn time for the entire mission is ~105.2 minutes.

Reusable DRA 5.0 Copernicus Option 2 (4 – 25 klb$_f$ NTR Engines, 24-hr EEO capture $\Delta V_{EOC} \sim 1.855$ km/s):

In Option 2 (Fig. 10 – Configuration 3 and Fig. 14), Copernicus uses four conventional 25 klb$_f$ NTR engines cooled with LH_2 propellant, and carries the same 6-person crew and payload baselined in DRA 5.0. The IMLEO for Option 2 is ~542.6 t including the NTPS (~137.6 t), the in-line tank (~105.1 t), a 4-sided "star truss" with 4 modular LH_2 drop tanks (~233.6 t) and the crew PL section (~66.3 t). Two drop tanks (~112.9 t) carrying ~88.5 t of LH_2 (~44.2 t per tank) are delivered to LEO on a single HLV launch, then attached to the ~16 m long star truss (~7.8 t) during the LEO assembly phase. Each drop tank is ~11 m long. The overall vehicle length is ~102.1 m including the 8.9 m long Orion MPCV. The LH_2 loading in the propulsion stage, in-line and 4 drop tanks is ~85.3 t, ~76.9 t, and ~176.9 t, respectively, and the lengths of NTPS and in-line LH_2 tanks are ~19.7 m and ~17.6 m. For Option 2, there are 5 primary burns (4 restarts) and ~319.1 t of LH_2 propellant is used during the mission. With 100 klb$_f$ of total thrust and I_{sp} ~906 s, the total engine burn time for this mission is ~106.2 minutes. The first TMI perigee burn is the longest single burn at ~38.2 minutes. The subsequent burn durations are as follows: second perigee burn (~28.8 minutes), MOC (~16.7 minutes), TEI (~11.5 minutes) and EOC (~11 minutes). Four HLV launches are needed to deliver the major transportation systems elements (the NTPS, in-line tank and "twin" drop tank sets) to LEO. The SLS (70 t to LEO version) delivers the crewed payload element and the large but lightweight star truss.
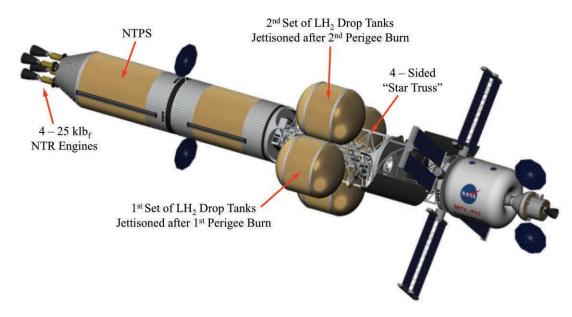
Figure 14. Configuration 3: Supports Reusable DRA 5.0 and 2033 Short Orbital Stay Mars Missions

Reusable 2033 Short Orbital Mars Mission (4 – 25 klb$_f$ Engines, 24-hr EEO capture ΔV_{EOC} ~1.557 km/s):

Configuration 3 can also support a reusable 18 month round trip Mars orbital mission in 2033 with a crew of 4 by increasing the lengths of the in-line and drop tanks (to ~21.1 m and ~12.4 m, respectively) to carry more LH$_2$. The vehicle has an IMLEO of ~581.3 t, which includes the NTPS (~140 t), the in-line tank (~121.4 t), the "star truss" with 4 modular LH$_2$ drop tanks (~270.8 t) and the crew PL section (~49.1 t). The overall vehicle length is ~112 m. The LH$_2$ loading in the propulsion stage, in-line and 4 drop tanks is ~84.3 t, ~94.5 t, and ~200.3 t, respectively, and the total usable propellant is ~356.2 t. As illustrated in Fig. 14, the first set of LH$_2$ drop tanks are jettisoned after the first perigee burn to reduce vehicle mass and propellant consumption during the second perigee burn. With 100 klb$_f$ of thrust and I$_{sp}$ ~906 s, the total engine burn time is ~118.6 minutes, close to the ~2 hours demonstrated on the NRX-XE engine. The duration of the individual burns are as follows: first perigee burn (~41.6 minutes), second perigee burn (~28.1 minutes), MOC (~15.7 minutes), TEI (~24.1 minutes) and EOC (~9.1 minutes). Again, four HLV and two SLV launches would be required to deliver the vehicle elements to LEO. Although reusable short orbital stay missions are possible using the modular approach, significant propellant and hardware mass (jettisoned drop tanks) is expended in conducting a mission that has a limited orbital stay time at Mars (60 days) compared to the total mission duration (545 days). It is therefore logical to ask the following question, "Should long surface stay expendable missions with faster transit times be considered instead and what is the impact on vehicle design?"

Options for Faster "1-Way" Transit Times in DRA 5.0:

The key to achieving shorter "1-way" transit times in fast-conjunction Mars missions like that used in DRA 5.0 is more propellant. For the 2033 opportunity, the total mission ΔV budget (TMI, MOC and TEI minus gravity losses) is ~6.05 km/s for 180-day transit times to and from Mars. It increases to ~8.83 km/s for 120-day transit times and even more dramatically to ~15.2 km/s for 90-day transit times to and from Mars. By reconfiguring Copernicus with an in-line and twin drop tanks to increase its propellant capacity, along with a fourth NTR engine (shown in Fig. 15) to increase vehicle thrust and reduce gravity losses, transit times to and from Mars can be cut by ~33% to 120 days each way. The vehicle IMLEO is ~551.3 t including the NTPS (~139.9 t), the in-line tank (~133.8 t), the "star truss" with drop tanks (~210.2 t) and the crew PL section (~67.3 t). The overall vehicle length is ~116 m. The LH$_2$ loading in the propulsion stage, in-line and 2 drop tanks is ~83.1 t, ~98.6 t, and ~159.3 t, respectively, and the total usable propellant is ~321 t. With 100 klb$_f$ of thrust and I$_{sp}$ ~906 s, the total engine burn time is ~106.9 minutes. After a long first perigee burn (~52.3 minutes) is completed, both drop tanks are jettisoned. The remaining burn durations are as follows: second perigee burn (~20.8 minutes), MOC (~22.3 minutes) and TEI (~11.5 minutes). Four HLV and two SLV launches are again required to deliver the vehicle elements to LEO.
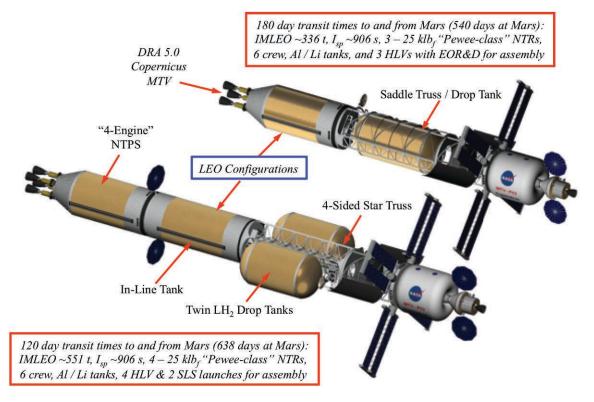
180 day transit times to and from Mars (540 days at Mars): IMLEO ~336 t, I_{sp} ~906 s, 3 – 25 klb_f "Pewee-class" NTRs, 6 crew, Al / Li tanks, and 3 HLVs with EOR&D for assembly

DRA 5.0 Copernicus MTV

Saddle Truss / Drop Tank

"4-Engine" NTPS

LEO Configurations

4-Sided Star Truss

In-Line Tank

Twin LH_2 Drop Tanks

120 day transit times to and from Mars (638 days at Mars): IMLEO ~551 t, I_{sp} ~906 s, 4 – 25 klb_f "Pewee-class" NTRs, 6 crew, Al / Li tanks, 4 HLV & 2 SLS launches for assembly

Figure 15. Adding In-Line & Twin Drop Tanks to Copernicus Enables 120-Day Transit Times for DRA 5.0

VI. Human NEA Mission Possibilities Using NTP

The benefits of using NTP for human missions to both small, low energy and large, high energy NEA targets are examined in this section. The small NEA selected is 2000 SG344. It has a 2028 launch date and a round trip time of ~327 days that includes a 7-day NEA stay time. Specific mission ΔV budget details include trans-NEA injection (TNI) ΔV_{TNI} ~3.254 km/s, $\Delta V_{Arrival}$ ~0.144 km/s, ΔV_{TEI} ~0.392 km/s and 6-hr EEO capture ΔV_{EOC} ~1.203 km/s (based on an arrival V-infinity at Earth of ~0.855 km/s). The 2028 mission date is consistent with both the ISECG's GER that shows a first crewed NEA mission beginning in 2028, as well as, "preliminary" development plans that envision an Initial Operational Capability (IOC) for a crewed NTP transportation system in this same timeframe.

Small asteroids, like 2000 SG344 (~35 – 60 m), are likely to be fast rotating and have a monolithic composition with less surface regolith. Large asteroids – 100 m or larger, tend to rotate more slowly and have a high probability of being rubble piles of rock offering a greater diversity of surface terrain and material composition desired by the scientific community [27]. The large, high energy NEA target analyzed in this paper is Apophis (~270 – 350 m). Like 2000 SG344, it too has a 2028 launch date but the round trip time is longer (~344 days) for the same 7-day stay time. The ΔV budget for the Apophis mission is also larger than that for 2000 SG344 with ΔV_{TNI} ~3.783 km/s, $\Delta V_{Arrival}$ ~1.542 km/s, ΔV_{TEI} ~0.342 km/s, and ΔV_{EOC} ~1.950 km/s (assuming capture into a 24-hr EEO with an arrival V-infinity of ~5.882 km/s). Apophis is of particular interest to NASA because on Friday, April 13, 2029, it will pass Earth's surface at an altitude of ~18,300 miles – within the orbits of geosynchronous communications satellites [30]. It will return for another close Earth approach in 2036.

Two human mission architectures have been examined [31] – one is "fully reusable" and the other "expendable". The reusable mission scenario is shown in Fig. 16. Three SLS / HLV launches (with lift capabilities ranging from ~70 t – 140 t) deliver the components for the Asteroid Survey Vehicle (ASV) to LEO over a 60 day period (30 day launch centers are assumed). The crewed ASV, shown in Fig. 17, is a "Copernicus-class" vehicle called "Searcher". Like Copernicus, Searcher is a 0-g_E, in-line vehicle design that uses automated R&D for assembly and has three key elements: (1) the "core" NTPS; (2) the integrated "saddle truss" and LH_2 drop tank assembly; and (3) the crewed NEA payload element. The crew would be launched on either a commercial crew delivery system or atop the SLS in the Orion MPCV that would then dock with the orbiting ASV at the front end of the TransHab crew module.
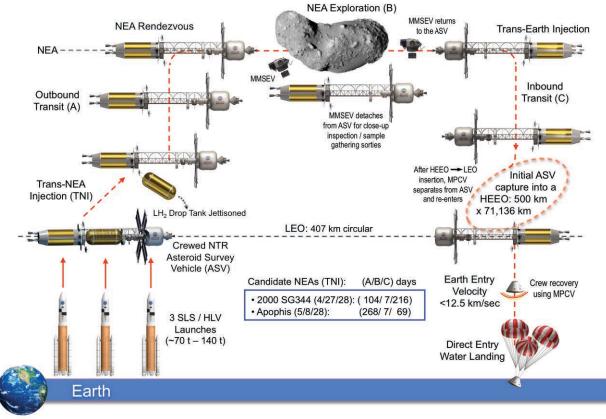
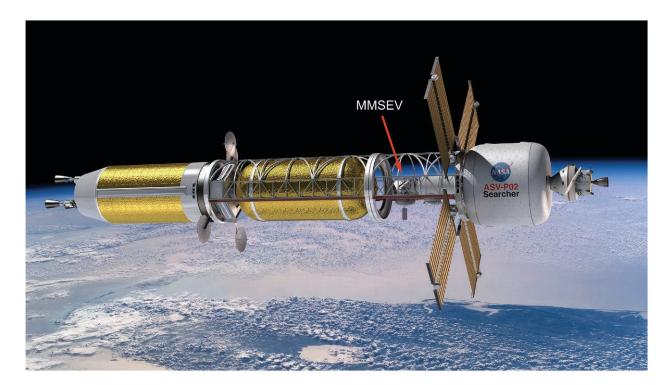**Figure 16. "Fully Reusable" Mission Scenario – ASV and MMSEV Returned to Earth Orbit**

Figure 17. DRA 5.0 Copernicus Crewed MTV Outfitted as Asteroid Survey Vehicle – "Searcher"

Following a "2-perigee burn" TNI maneuver, the drained LH_2 drop tank is jettisoned from the saddle truss and the ASV coasts to the target NEA under $0\text{-}g_E$ conditions with its PVAs tracking the Sun. Attitude control, mid-course correction and vehicle orientation maneuvers are again provided by Searcher's split RCS with bipropellant AMBR thrusters located on the NTPS and on the short saddle truss forward adaptor ring just behind the TransHab module. After propulsively braking near the target NEA, Searcher uses the post burn "cool-down thrust" provided by its three NTR engines, together with its RCS, to rendezvous with the NEA. Two crewmembers then transfer over to the MMSEV, undock from the transfer tunnel, and travel over to the NEA to begin the surface exploration and sample-gathering phase of the mission (Fig. 18). With Searcher at an appropriate standoff distance from the target NEA, multiple MMSEV sorties with rotating crews are flown to different NEA locations to gather a diverse sampling of materials.

As the 7-day stay at the target NEA draws to a close, the crew begins a period of vehicle checkout and systems verification before performing the TEI burn to begin the journey back to Earth. In the fully reusable architecture, the MMSEV is returned to Earth along with ASV. On final Earth approach, Searcher performs a braking burn and captures into a 24-hour EEO (~500 km x 71,136 km) like the reusable Mars MTV options discussed above. Its post burn engine cool-down thrust is then used to assist in orbit lowering. An auxiliary tanker vehicle, operating from a LEO servicing node/propellant depot, supplies the additional LH_2 needed by Searcher for final orbit lowering and rendezvous with the LEO transportation node where it is refurbished and resupplied before its next mission. The crew then enters the Orion MPCV, separates from Searcher and does a direct entry and landing on Earth. The departure dates, outbound, stay and return times for 2000 SG344 and Apophis are also shown in Fig. 16.
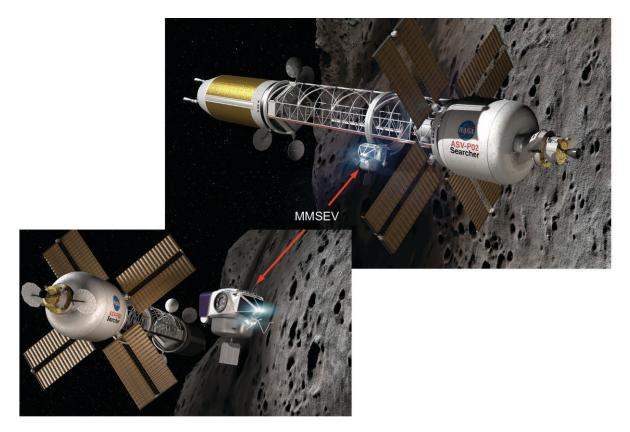

Figure 18. MMSEV Separates from ASV for Up Close NEA Examination and Sample Gathering

In the expendable architecture, all of the mission elements are disposed of in order to keep the total mission IMLEO and the size of the transportation system components as low as possible. The MMSEV is not returned to Earth but is left at the target NEA for continued teleoperated exploration after the crew has departed. To reduce vehicle mass and propellant requirements further, the transfer tunnel is also jettisoned before the TEI maneuver. On the final approach to Earth, the crew enters the Orion MPCV, separates from the ASV and does a direct entry and landing while the ASV flies by Earth at a "sufficiently high altitude" and is disposed of into heliocentric space.

VII.   Asteroid Survey Vehicle (ASV) Configuration Options for Candidate NEA Missions

The "Copernicus / Searcher" spacecraft design illustrated in Figs. 8 and 17 is used as the baseline configuration for assessing the benefits of using NTP for human NEA missions. Two target NEAs – both with departure dates in 2028 – have been selected for analysis in this paper. They are 2000 SG344 (a small asteroid with a low energy / ΔV requirement) and Apophis (a large asteroid with a high energy / ΔV requirement). The impact of key mission variables (like crew size, the assumed mission architecture, and launch vehicle performance – specifically lift capability and PL volume) on vehicle size and mass has been assessed and is discussed below. Key features and component lengths for three different size ASV options examined for our target NEA missions are shown in Fig. 19.
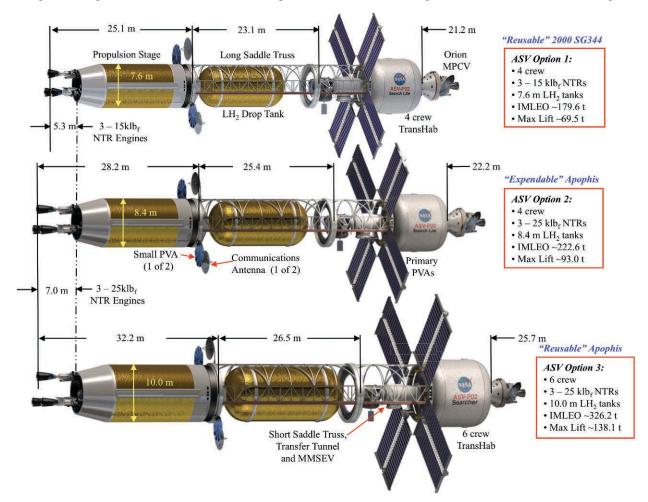

Figure 19. ASV Configuration Options for Reusable and Expendable Human NEA Missions

ASV Option 1 (Reusable Mission to 2000 SG344):

Because of its low energy requirements, a human mission to 2000 SG344 can readily be accomplished with a "scaled-down" version of the larger 10 m diameter Searcher ASV, called "Search Lite". Search Lite Option 1 uses three smaller 15 klb$_f$ NTR engines on its propulsion stage rather than the baseline 25 klb$_f$ Pewee-class engines. It also uses 7.6 m D LH$_2$ tanks and carries a smaller TransHab module to accommodate a crew of 4. Housed within the forward cylindrical adaptor section of all the propulsion stage options shown in Fig. 19 is the RCS, avionics, batteries, deployable twin Orion-type circular PVAs, and docking system, along with a reverse turbo-Brayton cryocooler system for ZBO LH$_2$ storage. The Brayton cryocooler system mass and power requirements increase with tank diameter and are sized to remove the heat load penetrating the 60 layer MLI system while the stage is in LEO where the highest tank heat flux occurs (see Table 2). The small circular PVAs on the propulsion stage provide the electrical power for the ZBO system in LEO.

With Option 1, a human mission to 2000 SG344 can be performed using three 70 t-class SLS launches. The vehicle IMLEO is ~179.6 t which includes the NTPS (~69.5 t), the saddle truss and drop tank assembly (~54.8 t) and the crew PL section (~55.3 t). The overall vehicle length is ~78.3 m including the 8.9 m long Orion MPCV. The LH_2 tank lengths for the NTPS and the drop tank are identical at ~15.7 m with each tank carrying ~38.9 t of LH_2 propellant (~98% of tank's maximum capacity of ~39.7 t). The NTPS used for Option 1 is the same as that used for a reusable, crewed lunar landing mission discussed later in the paper. It is this lunar mission application that actually determines the NTPS's physical dimensions and characteristics. Maximizing the use of common hardware elements (e.g., same size NTPS, propellant tanks) for different mission applications is an important consideration that can help reduce vehicle development and recurring costs and is utilized throughout this paper.

The long "saddle truss" connecting the propulsion stage and PL sections is a composite structure whose mass scales with tank diameter and length and varies from ~4.2 t – 8.9 t for 7.6 m – 10.0 m diameter tanks. (The short saddle truss included in the PL section uses the same composite structure.) The crewed PL section also includes deployable rectangular PVAs used for primary power. The four PVAs shown in Fig. 19 are appropriate for use at Mars because of the decreased solar radiation (~486 W/m^2). For NEA missions that are flown near 1 A.U. (solar radiation ~1368 W/m^2), two smaller panels producing ~15 – 25 kW_e should be adequate.

The reusable crewed mission to 2000 SG344 requires 5 primary burns (4 restarts) that use ~73.7 t of LH_2 propellant. With 45 klb_f of total thrust and a I_{sp} of ~906 s, the total engine burn time is ~54.5 minutes. The first of the two TNI perigee burns is the longest single burn at ~29.6 minutes after which the vehicle's drop tank is drained and jettisoned. The NTPS provides the LH_2 propellant needed for the remaining propulsive maneuvers: the second perigee burn (~10.5 minutes), braking at 2000 SG344 (~1.3 minutes), TEI (~3.4 minutes), and EOC (9.7 minutes). With the vehicle's available propellant capacity, it is also capable of capturing into a lower apogee, higher energy 6-hr EEO at the end of the mission.

ASV Option 2 (Expendable Mission to Apophis):

Search Lite Option 2 shown in Fig. 19 is sized for a 344-day expendable mission to Apophis with a crew of 4. With an estimated diameter of ~270 – 350 m, Apophis is ~5 – 10 times larger than 2000 SG344. A size comparison between a notional "Apophis-like" NEA and Search Lite Option 2 is shown in Fig. 20. Option 2 uses three clustered 25 klb_f engines and 8.4 m diameter Al/Li LH_2 propellant tanks on both the propulsion stage and drop tank. (The Al/Li LH_2 tank on the SLS "core stage" currently has this same diameter.) Even with its higher ΔV requirement, a crewed mission to Apophis is still possible, in an expendable mode, using three 100 t-class SLS launches and ASV Option 2 (illustrated in Fig. 21). As with Option 1, the drop tank is drained and jettisoned after the first perigee burn and the NTPS supplies the propellant for the second perigee burn, NEA braking and TEI maneuvers that follows.

The vehicle IMLEO is ~222.6 t including the NTPS (~93 t), the saddle truss / drop tank assembly (~73.3 t) and the crew PL section (~56.3 t). The overall vehicle length is ~84.7 m with the Orion MPCV. The LH_2 tank lengths are the same for both the propulsion stage and the drop tank at ~16.9 m and each contains ~52.5 t of LH_2 propellant.
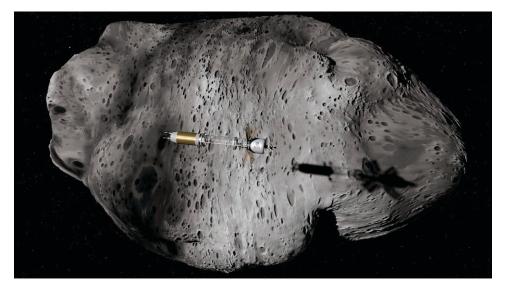

Figure 20. Size Comparison of ASV Option 2 and a Notional 300 m Long NEA

For this expendable mission to Apophis, there are 4 primary burns and 3 engine restarts. The total LH_2 propellant used in the mission is ~98.7 t and the total engine burn time is ~43.8 minutes. The first perigee burn is the longest at ~25 minutes and provides ~2/3rd of total ΔV required for the TNI maneuver. The durations of the remaining burns are as follows: the second perigee burn (~8.4 minutes), braking at Apophis (~8.8 minutes), and TEI (~1.6 minutes).

With its smaller diameter tanks, Search Lite Option 2 does not have sufficient propellant capacity to be reusable. This deficiency can be corrected through the addition of an "in-line" LH_2 tank as discussed previously. The resulting "4-element" version of Option 2 described elsewhere [31] has an IMLEO of ~347.9 t including the NTPS (~98.6 t), the in-line tank (98.2 t), the saddle truss and LH_2 drop tank (~91.3 t), and the crewed PL element (~59.8 t). The total engine burn time is ~84 minutes, and the longest single burn is again the first perigee burn at ~40.9 minutes. Increasing the SLS/HLV lift capability and usable PL volume allows a reusable "3-element" ASV discussed below.

Figure 21. "Search Lite" ASV used in Expendable Human Mission to Apophis in 2028

ASV Option 3 ("Searcher" Reusable Mission to Apophis):

Option 3 is the Copernicus spacecraft outfitted with an MMSEV for a reusable mission to Apophis in 2028 (shown in Figs. 17, 18, and 19). Searcher uses three 25 klb$_f$ NTR engines, has 10 m diameter LH_2 tanks and carries a crew of 6. The Apophis mission has a total ΔV requirement of ~7.617 km/s (including the 24-hr EEO capture maneuver) which is ~4% larger than that needed for the expendable Mars DRA 5.0 mission. For a reusable mission scenario to this difficult target, Searcher's engines are operated at a higher temperature and I_{sp} (~940 s) during the two TNI perigee burns to reduce propellant consumption and its drop tank length is increased by ~0.4 m to ~21.5 m to accommodate the required propellant. Searcher has an IMLEO of ~326.2 t which includes its propulsion stage (~138.1 t), saddle truss and LH_2 drop tank (~125.9 t), and crewed PL element (~62.3 t). Searcher's overall length is ~93.3 m including the 8.9 m Orion MPCV. During launch, the engines have a portion of their nozzle (~2.2 m) retracted so the propulsion stage length does not exceed 30 m. Once in orbit, the nozzles are extended and the propulsion stage length increases to ~32.2 m. The tank length for Searcher's propulsion stage is the same as that on Copernicus at ~19.7 m but its drop tank is slightly longer at ~21.5 m. The corresponding propellant loads are ~87.2 t and ~96.4 t for a maximum LH_2 capacity of ~183.6 t. For Apophis, the total LH_2 propellant used is ~174.2 t and the total engine burn time is ~77.3 minutes. The first perigee burn again provides ~2/3rd of total ΔV required for TNI and is the longest single burn at ~37.2 minutes. The durations of the remaining burns are as follows: the second perigee burn (~12 minutes), braking at Apophis (~13 minutes), TEI (~2.5 minutes) and EOC (~12.6 minutes).

Other options for the Apophis mission include returning just the Searcher ASV to EEO and leaving the MMSEV at Apophis for continued autonomous exploration, or reducing the TransHab mass and associated consumables to accommodate 4 crewmembers (NASA's baseline for NEA missions). This later option reduces IMLEO to ~313.7 t and requires no increases in engine I_{sp} or drop tank length.

VIII. "Search Lite" Vehicle Utilization for Lunar Cargo and Crewed Landing Missions

The Search Lite vehicle configurations described above can also play an important role in returning humans to the Moon "to stay" by providing an affordable in-space transportation system with reuse capability that could allow initial lunar outposts to evolve into eventual settlements capable of supporting commercial activities. Utilization of efficient NTP for lunar cargo delivery and crewed lunar landing missions is also consistent with the *"Asteroid Next"* pathway that includes human missions to the Moon to test out key surface systems (e.g., habitats, power systems, and long-range pressurized rovers) needed for an eventual human landing on Mars.

Using the SLS lift capability of ~70 t – 100 t to LEO, two classes of Search Lite vehicles are examined. The Class-I vehicle uses a NTPS with $3 - 15$ klb$_f$ engines, 7.6 m D propellant tanks and has a mass limit of ~70 t. The Class-II vehicle uses $3 - 25$ klb$_f$ engines, 8.4 m D propellant tanks and has a mass limit of ~100 t.

For reusable cargo delivery missions, three SLS launches are used to deliver the vehicle and payload elements to LEO. The NTP cargo transport then departs from LEO (ΔV_{TLI} ~3.3 km/s including g-losses of ~200 m/s) and captures into a circular LLO (~300 km, ΔV_{LOC} ~915 m/s) approximately 72 hours later. Key phases of the cargo delivery mission are illustrated in Fig. 22. Once in orbit, the habitat lander(s) separate from the lunar NTR (LNTR) transport and descends to the surface, landing autonomously at a predetermined location on the Moon. It is assumed that the habitat landers use LOX/LH$_2$ chemical engines and are also equipped with either deployable wheels or articulated landing gear allowing movement in both the vertical and horizontal directions so that the landers can either "drive or walk" short distances from the landing site. Connecting several "functionally different" lander modules together (for habitation, science, etc) would form a large contiguous pressurized volume for the crew and also provide a "building block" approach to establishing an initial lunar base.
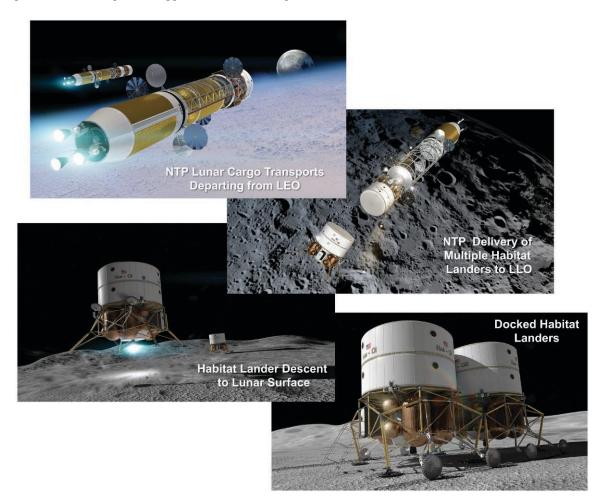

Figure 22. Reusable NTP Lunar Cargo Delivery Mission Phases

After payload separation, the LNTR transport departs LLO (ΔV_{TEI} ~915 m/s) and returns to Earth capturing into a 24-hr EEO (ΔV_{EOC} ~0.355 km/s). Post burn engine cool-down thrust and an auxiliary tanker vehicle are again used to return the LNTR vehicle to the LEO transportation node for refurbishment and resupply before its next mission.

The key phases of the crewed NTR landing mission are illustrated in Fig. 23. Again, three SLS launches – two for the LNTR vehicle and one for the crewed payload element – are used for this reusable lunar mission that returns the LNTR transport vehicle, MPCV and LLV to EEO. After capture and rendezvous with the LEO propellant tanker, the crew separates the MPCV from the LNTR and re-enters using the Orion capsule.

After R&D of the LNTR elements, the crewed payload element is launched then attached to the LNTR vehicle. The crewed element includes the Orion MPCV plus a "single stage" LOX/LH$_2$ LLV that carries a crew of 4 and 5 t of surface payload stored in two "swing-down" containers mounted on each side of the crew cab. After the "2-perigee burn" TLI maneuver (ΔV_{TLI} ~3.3 km/s including g-losses), the drop tank is jettisoned from the saddle truss at which time the Orion MPCV separates from the front of the LNTR vehicle. It then repositions itself inside the saddle truss where it docks with the LLV using a common "docking port" mounted to the forward saddle truss ring (shown in Fig. 23). After a 3-day coast to the Moon, the LNTR vehicle performs the LOC burn (ΔV_{LOC} ~915 m/s) inserting itself and its payload into LLO. The crew then enters the LLV, separates from the LNTR transport and prepares to land.
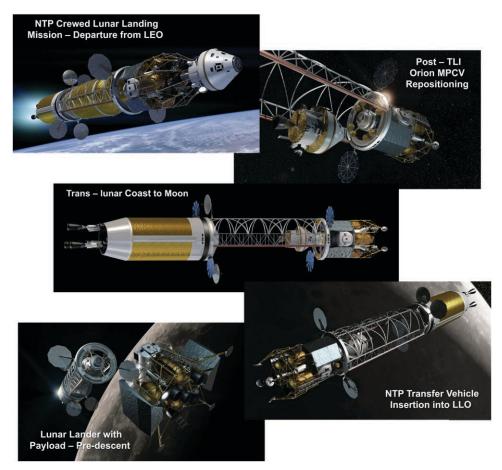

Figure 23. Reusable NTP Crewed Lunar Landing Mission – Outbound Mission Leg

The LLV has total "wet" mass of ~35.3 t (Table 1) that includes the crew cab (~2.5 t), the descent / ascent stage (~6.1 t) and its LOX/LH$_2$ propellant (~20.9 t), surface payload (~5 t), plus the 4 crew and their suits (~0.8 t). After separating from LNTR, the LLV's two payload containers are rotated 180 degrees and lowered into their landing position in preparation for descent to the lunar surface. The ΔV budget for the LLV includes the following [12]: ΔV_{des} ~2.115 km/s and ΔV_{asc} ~1.985 km/s. The LLV uses five RL 10A-4 engines that operate with a I$_{sp}$ ~450 s consistent with the Martin Marietta design [12]. It expends ~13.4 t of LOX/LH$_2$ propellant during the descent to the
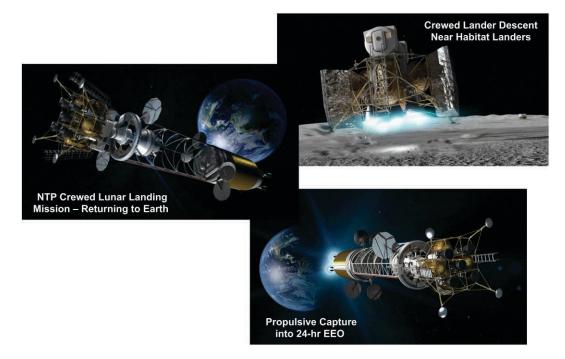
Figure 24. Reusable NTP Crewed Lunar Landing Mission – Landing and Return Mission Leg

surface (shown in Fig. 24). After lunar touchdown, the crew can operate out of the LLV for ~14 days using its surface landed payload or longer (~180 days) using the pre-deployed habitat landers.
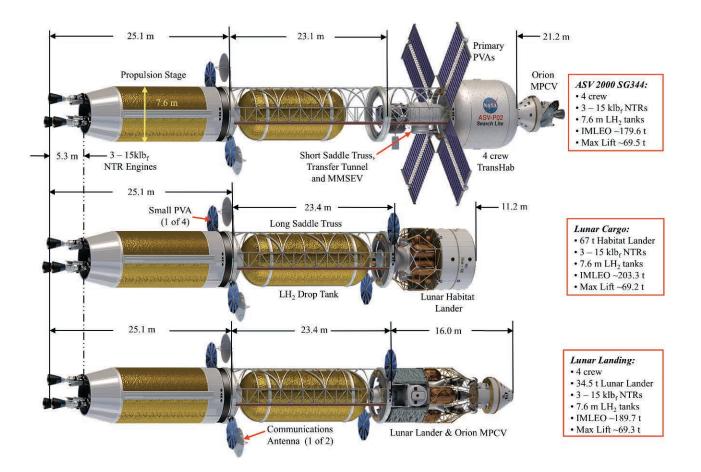
As the "surface systems test and exploration" phase of the mission nears its completion, the crew prepares the LLV for departure. At liftoff, the LLV mass is ~15.1 t and ~5.5 t of propellant is used during the ascent to LLO. The LLV, with 100 kg of lunar samples, then rendezvous with the LNTR vehicle and preparations for the TEI maneuver begin. After completing the TEI burn (ΔV_{TEI} ~915 m/s), the crew spends the next 3 days readying the LNTR for the final phase of the mission – capture into a 24-hr EEO with its MPCV and LLV payload (depicted in Fig. 24) followed by MPCV separation and capsule re-entry of the crew.

IX. Lunar Mission Capabilities using "Search Lite" Class-I and -II Vehicle Configurations

The key features, component lengths, and launch mass requirements for lunar cargo and crewed landing missions using the smaller Class-I ASV elements are shown in Fig. 25. The Class-I Search Lite ASV used for the reusable, crewed mission to 2000 SG344 is also included for comparison. As mentioned previously, Class-I vehicles use a NTPS with 3 – 15 klb$_f$ engines, have 7.6 m D propellant tanks and have a mass limit on vehicle elements of ~70 t. The LNTR cargo transport shown in Fig. 25 has an IMLEO of ~203.3 t consisting of the NTPS (~69.2 t), the saddle truss and drop tank assembly (~64.5 t) and the habitat lander (~67 t) with its connecting structure (~2.6 t). The overall vehicle length is ~59.7 m. The LH$_2$ propellant loads in the NTPS and drop tank are ~39.7 t and ~44.8 t. The corresponding tank lengths are ~15.7 m and ~18.4 m – lengths sized by the crewed lunar landing mission.

For the reusable cargo delivery mission, the 5 primary engine burns use ~79.4 t of LH$_2$ propellant. The drop tank is jettisoned after the second perigee burn. With 45 klb$_f$ of total thrust and I$_{sp}$ ~906 s, the total engine burn time is ~58.7 minutes. The first of the two TLI perigee burns is the longest at ~25.4 minutes. The duration of the remaining burns are as follows: second perigee burn (~20.1 minutes), LOC (~8.9 minutes), TEI (~3.2 minutes) and EEO capture (~1.1 minutes). Given projected full power operational lifetimes of ~6 – 10 hours for NERVA-derived engines using composite fuel [23], cargo transport vehicles with multi-mission capability appear viable.

The crewed lunar landing mission has an IMLEO of ~189.7 t including the NTPS (~69.3 t), the saddle truss and drop tank assembly (~67.6 t), the wet LLV (~29.5 t) with its surface payload (~5 t), the Orion MPCV (~13.5 t), connecting structure (~3.5 t), consumables (~0.5 t), and 4 crewmembers (~0.8 t includes lunar EVA suits). The overall vehicle length is ~64.5 m. For the crewed mission, the LH$_2$ propellant loads in the NTPS and drop tank are at their maximum capacity of ~39.7 t and ~53.4 t for the specified tank lengths of ~15.7 m and ~18.4 m.

Figure 25. Class-I Vehicle Configurations for Reusable NEA, Lunar Cargo & Crewed Landing Missions

The common NTPS used for all three Class-I vehicles shown in Fig. 25 has an ~15.7 m long LH_2 propellant tank. The ASV uses a drop tank of this same length as well. It is positioned near the aft end of the saddle truss to minimize the length and mass of the propellant lines as well as transfer losses. All three vehicles also carry a saddle truss of the same approximate length – a little over 23 m to further maximize component commonality.

The crewed landing mission also requires 5 primary burns and jettisons its drop tank after the second perigee burn. With 45 klb_f of total thrust, I_{sp} ~906 s, and ~82.1 t of LH_2 propellant used during the mission, the total engine burn time is ~60.9 minutes, again well under the capabilities demonstrated on the NRX-XE. The first perigee burn is the longest at ~24.3 minutes and the remaining maneuvers having the following burn durations: second perigee burn (~18.1 minutes), LOC (~8.3 minutes), TEI (~7.5 minutes) and EOC (~2.7 minutes).

The component lengths and launch mass requirements for the larger Class-II lunar vehicles are shown in Fig. 26. The Class-II Search Lite ASV (Figs. 19 and 21) used in the expendable, crewed Apophis mission is also shown for comparison. Class-II vehicles use 3 – 25 klb_f engines, 8.4 m D propellant tanks, and require a 100 t-class SLS for component delivery. The Apophis mission discussed previously determines the dimensions and mass of the NTPS, saddle truss and drop tank, and these same components are used for the lunar cargo and crewed lunar landing missions. With larger diameter tanks and mass allowance for the vehicle components, two lighter "mini-hab" landers can also be delivered to LLO by the LNTR cargo transport as shown in Fig. 26.

The Class-II lunar cargo vehicle has an IMLEO of ~250.8 t including the NTPS (~91.6 t), the saddle truss and drop tank assembly (~75.2 t) and the twin habitat landers (~81 t) with their connecting structure (~3.0 t). The LH_2 tank lengths are the same for both the propulsion stage and the drop tank at ~16.9 m and each contains ~52.5 t of LH_2 propellant. At launch, the Class-II NTPS length is ~26 m with the nozzle skirt extensions on the larger 25 klb_f engines retracted ~2.2 m. At TLI, with the skirts fully extended, the overall length of the cargo transport is ~76 m.

The cargo mission jettisons its drop tank after the second perigee burn, has 5 primary burns and uses ~98.8 t of LH_2 propellant. With a higher total thrust (75 klb_f) on the Class-II vehicles and I_{sp} of ~906 s, the total engine burn time is shortened to ~43.9 minutes. The first perigee burn is again the longest single mission burn at ~19.5 minutes.
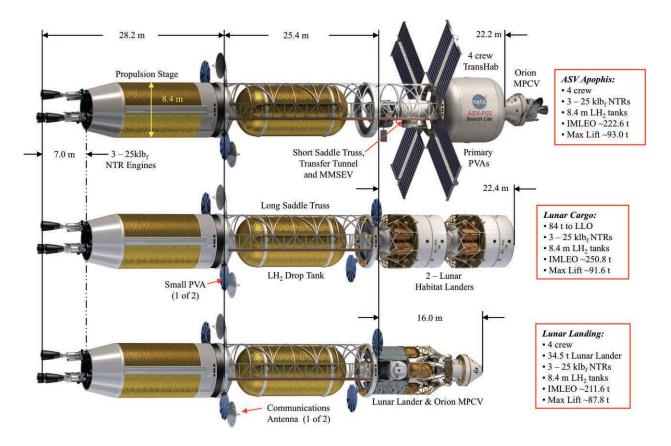
Figure 26. Class-II Vehicles for Expendable Apophis / Reusable Lunar Cargo & Crewed Landing Missions

The remaining propulsive maneuvers have the following burn durations: second perigee burn (~14.3 minutes), LOC (~6.7 minutes), TEI (~2.5 minutes) and EOC (~0.9 minutes).

Like the cargo transport, the LNTR vehicle for the crewed landing mission (shown in Fig. 26) uses the same NTPS, saddle truss and drop tank to maximize hardware commonality. It has an IMLEO of ~211.6 t that includes the NTPS (~87.8 t), the saddle truss and drop tank assembly (~71.7 t), the LLV (~29.5 t) and surface payload (~5 t), the Orion MPCV (~13.5 t), connecting structure (~2.8 t), consumables (~0.5 t), plus the 4 crew and their EVA suits (~0.8 t). The overall vehicle length is ~69.6 m. The NTPS and drop tank are also slightly off-loaded with each tank carrying ~48.7 t of LH$_2$ propellant (~93% of maximum capacity).

The reusable crewed landing mission uses ~91.7 t of LH$_2$ propellant. With the NTPS's higher thrust level, the total engine burn time is ~40.7 minutes. The duration of the 2-perigee departure burn is ~28.4 minutes with the first burn again the longest at ~16.4 minutes. After the second perigee burn, the drop tank is again jettisoned. The remaining burn durations include LOC (~5.5 minutes), TEI (~5.0 minutes) and EOC (~1.8 minutes).

X. Notional Plans for NTP Technology Development and Demonstration

In FY'11, NASA started a technology development effort in NTP under the Advanced In-Space Propulsion (AISP) component of its Exploration Technology Development and Demonstration (ETDD) program. The NTP effort included two key tracks: "Foundational Technology Development" followed by "Technology Demonstration" projects (details shown in Fig. 27). Near-term activities initiated under Foundational Technology Development (now part of NASA's Advanced Exploration Systems' Nuclear Cryogenic Propulsion Stage (NCPS) project [32]), included five key tasks and objectives:

Task 1. *Mission Analysis, Engine/Stage System Characterization and Requirements Definition* to help establish performance goals for fuel development and guide concept designs for small, scalable demonstration engines and the full size engines needed for future human Moon, NEA and Mars missions;

Task 2. *NTP Fuels and Coatings Assessment and Technology Development* aimed at recapturing fabrication techniques, maturing and testing fuel, then selecting between the two primary fuel forms previously identified by the

DOE and NASA – NERVA "composite" and UO$_2$ in tungsten "cermet" fuel [33]. Partial, then full-length fuel elements will be tested in the NTR Element Environmental Simulator (NTREES) [34] at the MSFC using up to ~1.2 MW of RF heating to simulate the NTP thermal environment that includes exposure to hot H$_2$. Candidate fuels and fuel element designs will be screened in NTREES prior to irradiation testing and final selections;

Task 3. Engine Conceptual Design, Analysis, and Modeling aimed at developing conceptual designs of small demonstration engines and the full size 25 klb$_f$-class engines utilizing the candidate fuels discussed above. State-of-the-art numerical models are being used to determine reactor core criticality, detailed energy deposition and control rod worth within the reactor subsystem [35], provide thermal, fluid and stress analysis of fuel element geometries [36], and predict engine operating characteristics and overall mass [37];

Task 4. Demonstration of Affordable Ground Testing focused on "proof-of-concept" validation of the SAFE (Subsurface Active Filtration of Exhaust) [38] or "bore-hole" test option at the Nevada Test Site (NTS). Non-nuclear, subscale hot gas injection tests, some with a radioactive tracer gas, will be conducted in a vertical bore-hole to obtain valuable test data on the effectiveness of the porous rock (alluvium) to capture, holdup and filter the engine exhaust. The data will also help calibrate design codes needed by DOE to design the SAFE test facility and support infrastructure needed for the small engine ground technology demonstration tests and the larger 25 klb$_f$-class engine tests to follow; and

Task 5. Formulation of an Affordable and Sustainable NTP Development Strategy aimed at outlining the content of an affordable development plan that utilizes separate effects tests (e.g., NTREES and irradiation tests), existing assets and innovative SAFE testing at the NTS, and small scalable engines for ground and flight technology demonstrations.
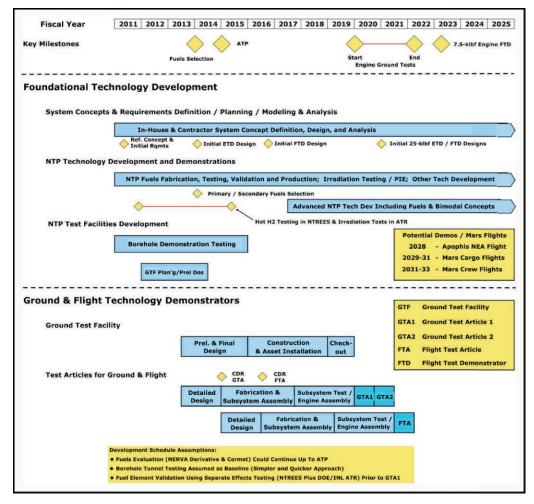

Figure 27. Notional NTP Development Plan includes Foundational, Ground and Flight Technology Demonstrations.

The above tasks, successfully carried out over the next 3 years under the NCPS project, could provide the basis for "authority to proceed" (ATP) in ~2015 with ground technology demonstration (GTD) tests at the NTS in late 2019, followed by a flight technology demonstration (FTD) mission in 2023. To reduce development costs, the GTD and FTD tests would use a small, low thrust (~6.5 - 7.5 klb$_f$) engine with a "common" fuel element design that is scalable to higher thrust levels by increasing the number of elements in a larger diameter core producing a greater thermal power output. The GTD effort would test two ground test articles (GTA1, GTA2) and one flight test article (FTA) that provide system-level technology demonstration and design validation for a follow-on FTD mission.

The small engine ground and flight demonstration tests would also maximize the use of existing and proven liquid rocket components to further ensure affordability. A small NTP FTD could fit within the 5-meter fairing of the Delta 4 M (5,4) launch system (shown in Fig. 28) and leverage a lot of DCSS components like the hydrogen tank, systems for pressurization, attitude control, avionics and power, plus inter-stage and thrust structure [39, 40]. The hydrogen tank's cylindrical barrel section would be increased to accommodate the propellant needed for a particular mission. A candidate FTD mission for a small NTP stage is a robotic flight to 2000 SG344. With a 7 klb$_f$ engine, I$_{sp}$ of ~905 s and ~6.2 t of LH$_2$ propellant, the total engine burn time is ~29.5 minutes.
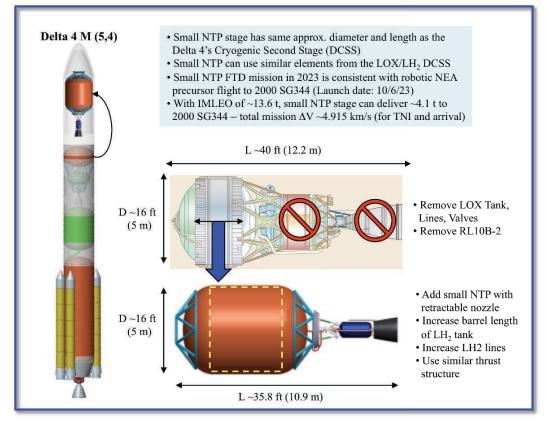

Figure 28. Small NTP Stage Launched on Delta 4 M (5,4) Could Validate NTR / Stage Hardware and Support a Robotic Precursor Flight to 2000 SG344 in Late 2023 Timeframe.

A single small engine stage can be used for a variety of robotic science missions, or a 2 – 3 engine cluster can be arranged on a larger capacity propellant tank for higher payload cargo delivery and modest-size crewed missions in cislunar space (e.g., Lagrange points). The FTD will also provide the technical foundation for an "accelerated approach" to design, fabrication, ground and flight testing of the fill-size 25 klb$_f$-class engine by ~2026. The Rover program used a common fuel element/tie tube design and similar approach to test the 50 klb$_f$ Kiwi-B4E, the 75 klb$_f$ Phoebus-1B, the 250 klb$_f$ Phoebus-2A, and 25 klb$_f$ Pewee engines, in that order, between 1964 and 1968. Flight testing a stage with clustered 15 klb$_f$ – 25 klb$_f$ engines would follow next in time to support human NEA then lunar missions in the late 2020's followed by short round trip / short orbital stay Mars missions in the early 2030's.

XI. Summary and Conclusions

The Global Exploration Roadmap developed by the ISECG, on behalf of NASA and the other participating space agencies, has identified two possible options for future human exploration known as the *"Asteroid Next"* and the *"Moon Next"* pathways. The *"Asteroid Next"* path has as its focus a first human NEA mission in 2028 necessitating the development and demonstration of key in-space exploration technologies and capabilities needed for traveling through and living in deep space. Advanced propulsion is one of these key technologies. The *"Moon Next"* path is focused on using the Moon to test and demonstrate key surface systems (e.g., habitats, power systems, and long-range pressurized rovers) required to support an eventual human landing on Mars. This paper shows the benefits of using NTP for human exploration missions to the Moon, NEAs and Mars that is consistent with either exploration pathway. It also outlines a growth path using "modular" components that can increase a vehicle's capability to support more demanding missions assuming a range of SLS / HLV lift capability varying from ~70 t – 140 t.

This "modular growth strategy" is first applied to the Copernicus crewed MTV design developed for DRA 5.0. To support short round trip/short orbital stay high energy opposition-class missions in the 2033 – 2035 timeframe, a "split mission" approach using the basic Copernicus MTV and its two key components configured as an ERV can be considered. For an "all-up" vehicle, the addition of an "in-line" LH_2 tank, positioned between the propulsion stage and integrated saddle truss / drop tank assembly, can provide the extra propellant capacity needed for this option.

Although Copernicus was operated in an "expendable mission mode" in DRA 5.0 to reduce total IMLEO, the addition of an in-line LH_2 tank and "star truss" assembly with 4 modular drop tanks, in place of the integrated saddle truss/drop tank assembly, can provide the additional propellant needed to operate Copernicus in a "reuse mode". Increasing the length and propellant capacity of in-line and drop tanks even further allows reusability for the 2033 Mars orbital mission as well. Reusability, however, is only possible if the necessary LEO infrastructure is in place (e.g., a transportation node / propellant depot with auxiliary tanker refueling capability) to support it.

Modular components can also be used to achieve even shorter "1-way" transit times to and from Mars using fast-conjunction trajectories like that used in a DRA 5.0. Again, the key requirement is more propellant. By increasing the length of the in-line tank and adding two large drop tanks to Copernicus to increase its propellant capacity along with a fourth NTR engine to increase vehicle thrust and reduce gravity losses, transit times to and from Mars can be cut by ~33% – from 180 days down to 120 days each way but at the cost of larger launch mass to LEO.

Since Copernicus was sized to perform fast-conjunction Mars missions over the entire 15-year synodic cycle, it has significant capability that can be used to conduct human missions to difficult, high energy NEAs, like Apophis. This asteroid is of particular interest because of its large size and its close approach to Earth in 2029. Outfitted with a MMSEV instead of an external consumables container, Copernicus can be transformed into an ASV, called Searcher, capable of carrying a crew of 6 on a 344-day round trip mission to Apophis, then returning to a 24-hour EEO for tanker servicing and return to LEO.

Scaled-down versions of the larger Copernicus/Searcher-class ASV, called Search Lite, have also been examined. Using three 70 t-class SLS launches, $3 – 15$ klb$_f$ engines, and 7.6 m diameter LH_2 propellant tanks, a Class-I Search Lite ASV can readily support a reusable, 327-day round trip low-energy mission to the small NEA, 2000 SG344, carrying a crew of 4. With three 100 t-class SLS launches, $3 – 25$ klb$_f$ engines, and 8.4 m diameter propellant tanks, a Class-II ASV can perform the more difficult mission to Apophis with 4 crew in an expendable mode. A reusable mission to Apophis is also possible by adding an in-line tank.

Using the same two size classes of "Search Lite" vehicles, reusable cargo delivery and crewed lunar landing missions are also possible using a 70 t – 100 t SLS. Search Lite vehicles can play an important role in returning humans to the Moon "this time to stay" by providing an affordable in-space transportation system that is reusable and can allow initial lunar outposts to evolve into eventual settlements capable of supporting commercial activities.

Lastly, NASA has initiated Foundational Technology Development work on NTP in a number of key areas under the NCPS project. If successful, this effort could be followed by major system-level Technology Demonstrations that include ground testing a small, scalable NTR by 2020, followed by a flight test of a small NTPS in 2023. A single small engine stage can support a variety of robotic science missions, or a cluster of $2 – 3$ small engines can be arranged on a larger capacity propellant tank to support higher payload cargo delivery and modest-size crewed missions in cislunar space (e.g., Lagrange points).

Acknowledgments

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