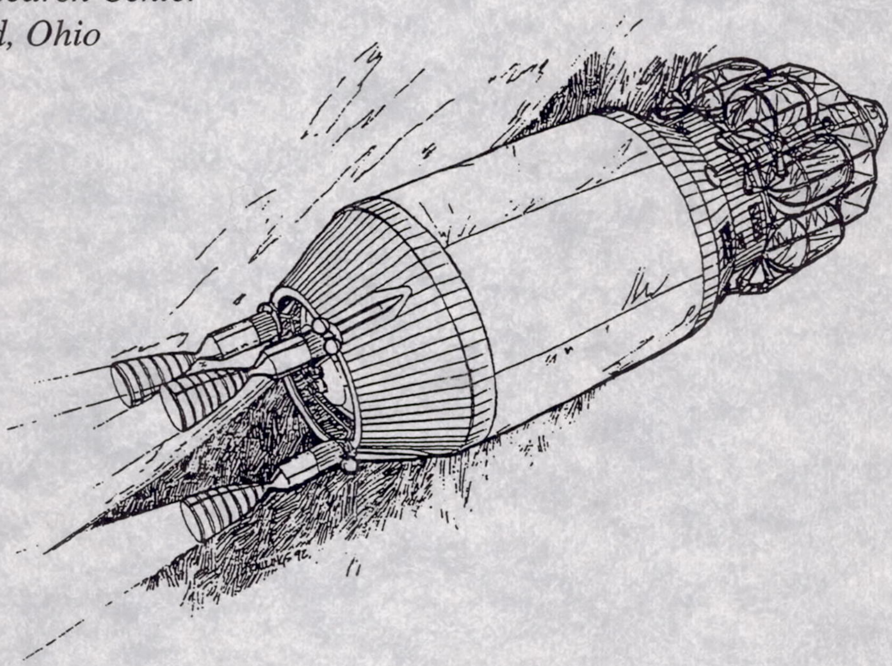
NASA Technical Memorandum 106748
AIAA–92–3812

"Fast Track" NTR Systems Assessment for NASA's First Lunar Outpost Scenario

Stanley K. Borowski and Stephen W. Alexander
Lewis Research Center
Cleveland, Ohio



Prepared for the
28th Joint Propulsion Conference and Exhibit
cosponsored by AIAA, SAE, ASME, and ASEE
Nashville, Tennessee, July 6–8, 1992

National Aeronautics and
Space Administration

"FAST TRACK" NTR SYSTEMS ASSESSMENT FOR NASA'S FIRST LUNAR OUTPOST SCENARIO

Stanley K. Borowski*
Nuclear Propulsion Office
NASA/Lewis Research Center
21000 Brookpark Road
Cleveland, OH 44135

Stephen W. Alexander**
Advanced Space Analysis Office
NASA/Lewis Research Center
21000 Brookpark Road
Cleveland, OH 44135

ABSTRACT

Integrated systems and mission study results are presented which quantify the rationale and benefits for developing and using nuclear thermal rocket (NTR) technology for returning humans to the Moon in the early 2000's. At present, the Exploration Program Office (ExPO) is considering chemical propulsion for its "First Lunar Outpost" (FLO) mission, and NTR propulsion for the more demanding Mars missions to follow. The use of an NTR-based lunar transfer stage, capable of evolving to Mars mission applications, could result in an accelerated schedule, reduced cost approach to Moon/Mars exploration. Lunar mission applications would also provide valuable operational experience and serve as a "proving ground" for NTR engine and stage technologies. In terms of performance benefits, studies indicate that an expendable NTR stage powered by two 50 klbf engines can deliver ~96 metric tons (t) to trans-lunar injection (TLI) conditions for an initial mass in low Earth orbit (IMLEO) of ~199 t compared to 250 t for a cryogenic chemical TLI stage. The NTR stage liquid hydrogen (LH_2) tank has a 10 m diameter, 14.8 m length, and 68 t LH_2 capacity. The NTR utilizes a "graphite" fuel form, consisting of coated UC_2 particles in a graphite substrate, and has a specific impulse (Isp) capability of ~870 s, and an engine thrust-to-weight ratio of ~4.8. The NTR stage and its piloted FLO lander has a total length of ~38 m and can be launched by a single Saturn V-derived heavy lift launch vehicle (HLLV) in the 200 to 250 t-class range. The paper summarizes NASA's First Lunar Outpost scenario, describes characteristics for representative engine/stage configurations, and examines the impact on engine selection and vehicle design resulting from a consideration of alternative NTR fuel forms and lunar mission profiles.

INTRODUCTION

The Space Exploration Initiative (SEI) outlined by President Bush on July 20, 1989, the 20th anniversary of Apollo 11, calls for a return to the Moon "to stay" early in the next century, followed by a journey to Mars using systems "space tested" in the lunar environment. Initial assessments of the space transportation system elements and infrastructures required to move humans and support equipment (e.g., habitats, supplies, and science and exploration equipment) from Earth to the surfaces of the Moon and Mars were outlined by the National Aeronautics and Space Administration (NASA) in its "90-Day Study Report"[1] and in an internal set of four White Papers. These NASA efforts were followed by the Synthesis Group report[2] which proposed four different architectural strategies for lunar/Mars exploration, identified key technology development areas and included recommendations for effectively implementing SEI.

The Synthesis Group also specified several important technical strategies common to its four architectures that affect space transportation systems design. These included use of (1) a heavy lift launch vehicle (HLLV) to limit on-orbit assembly; (2) a split mission strategy (where cargo and crew fly on separate missions); (3) pre-deployed and verified "turn-

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*Ph.D./Nuclear Engineering, Member AIAA
**Aerospace Engineer

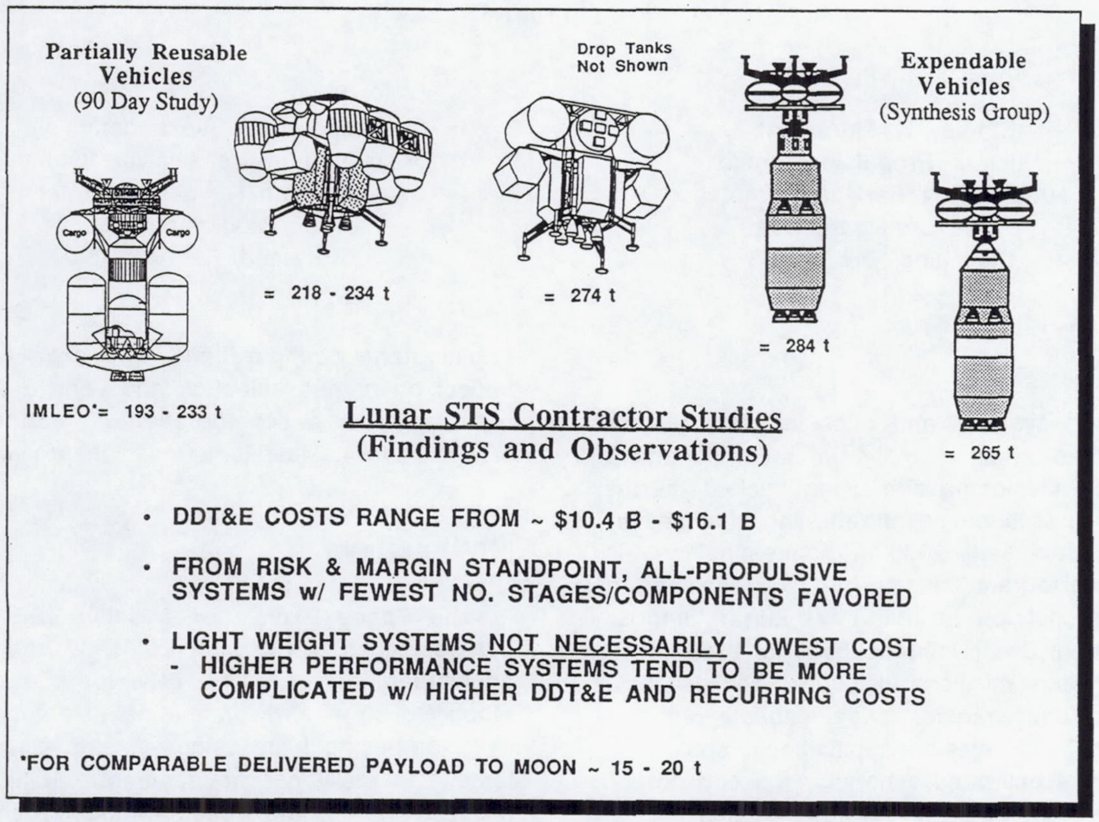


Fig. 1. Sampling of "Aerobraked/All-Propulsive" Chemical Lunar Transportation System Concepts

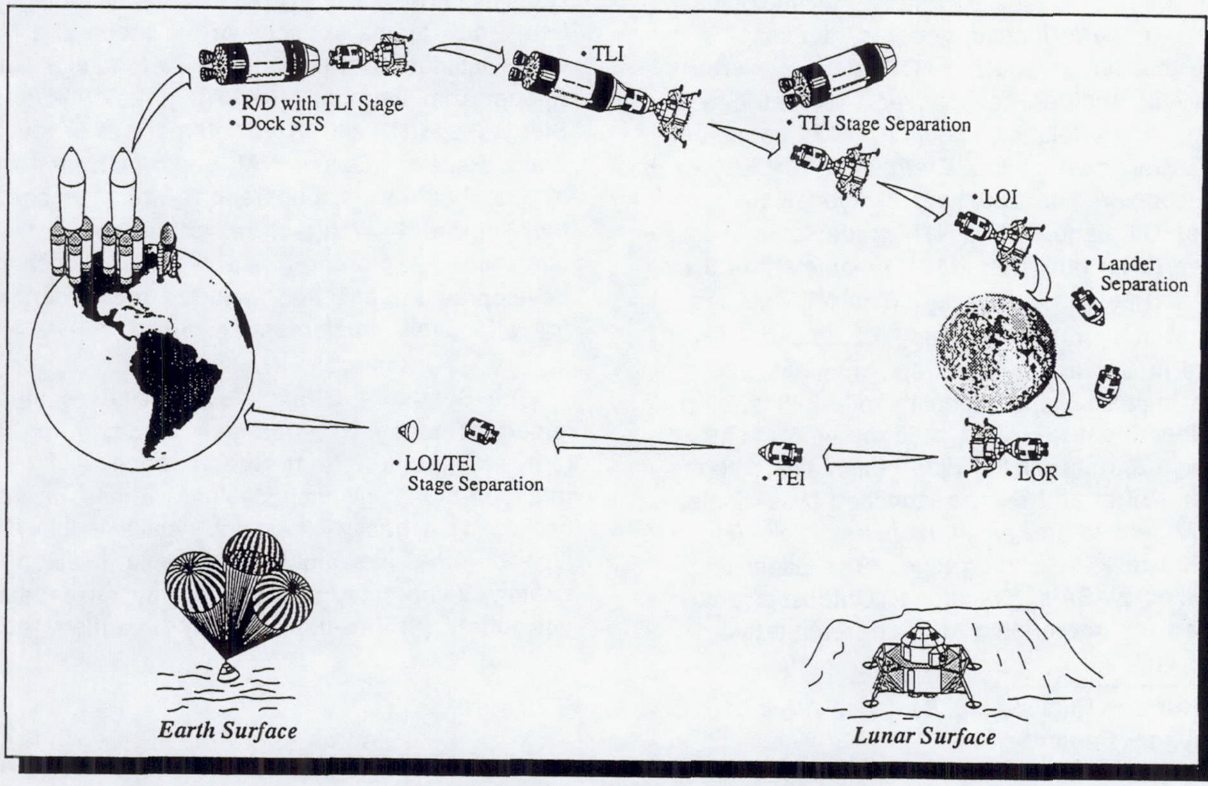


Fig. 2. Dual Launch Earth Orbit Rendezvous Lunar Mission Scenario

key" habitats; (4) chemical and nuclear thermal propulsion for lunar and Mars missions, respectively; (5) direct entry of returning crews to Earth's surface; (6) lunar missions as a "testbed" for Mars, and (7) to the extent possible, common systems for lunar and Mars missions.

As a result of the different ground rules and assumptions utilized in the NASA and Synthesis Group assessments, a spectrum of lunar space transportation system (LTS) concepts have been configured (see Figure 1). The 90-Day Study LTS consisted of two separate vehicles -- a "space-based" lunar transfer vehicle (LTV) operating between low Earth orbit (LEO) and low lunar orbit (LLO), and a lunar excursion vehicle (LEV) providing transportation between LLO and the lunar surface. The partially reusable LTV employed aerobraking for Earth orbit capture (EOC). This initial concept was followed by integrated, single crew module LTV/LEV configurations using either aerodynamic braking or propulsive braking for EOC. A transition occurred during the Synthesis Group activity away from reusable aerobrake concepts to more "Apollo-like" vehicle configurations operating in an "all propulsive" expendable mission mode. Both minimal capability single launch and higher performing dual launch Earth orbit rendezvous mission scenarios (see Figure 2) were studied assuming a 150 metric ton (t) HLLV capability and "direct capsule entry" for Earth return.

While chemical propulsion was baselined for lunar missions, the Synthesis Group recommended the NTR as the "only prudent propulsion system for Mars transit."[3] Because the time and cost to develop two separate transportation systems for SEI could be substantial, the Nuclear Propulsion Office (NPO) has been examining[4,5] the rationale and benefits of developing a "fully reusable" NTR-based lunar space transportation system and then evolving it to Mars mission applications through the use of modular engine/stage components (see Figures 3 and 4). In addition to enabling significant performance enhancements on its lunar missions (both in terms of reduced IMLEO and vehicle reusability), such an approach would allow NASA to make a significant down payment during its initial lunar program on key

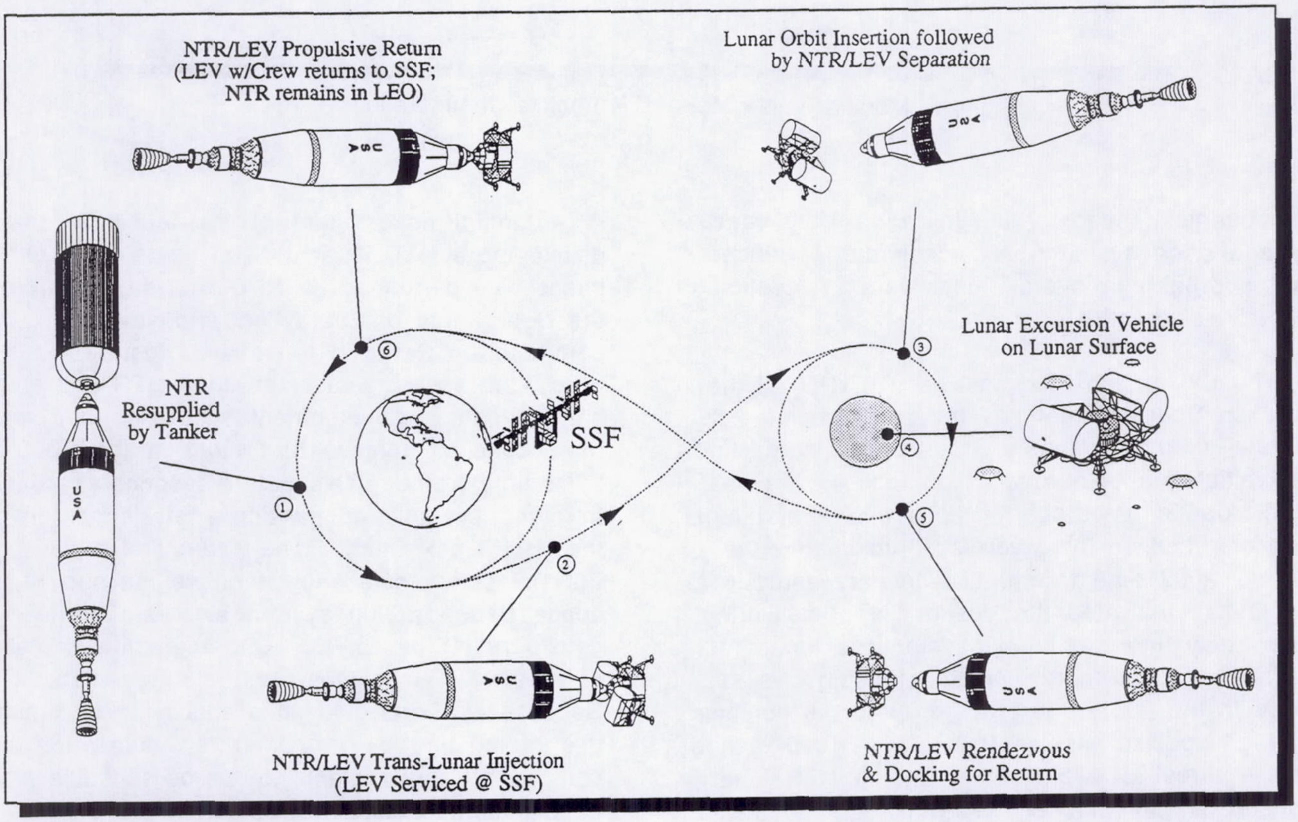


Fig. 3. "Fully Reusable" NTR Lunar Scenario

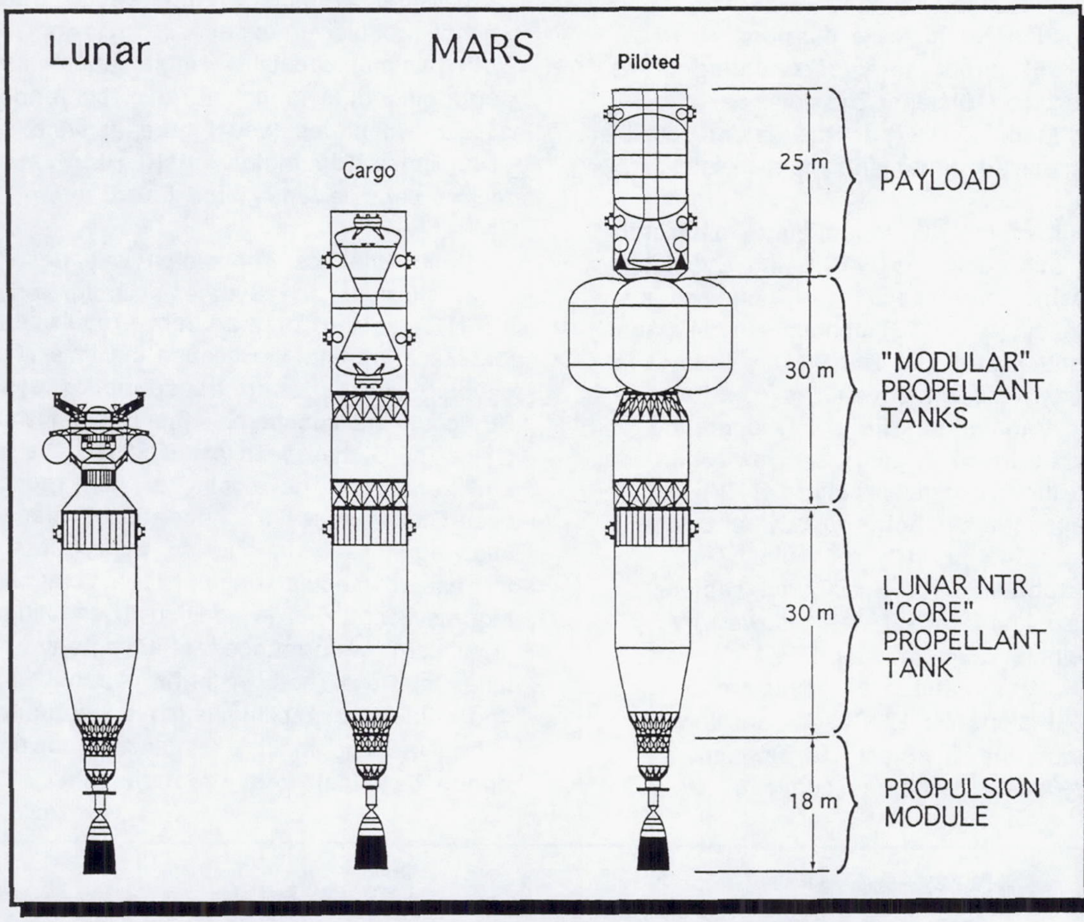


Fig. 4. Modular Lunar/Mars NTR Vehicle Configurations

components needed for the follow-on Mars space transportation system. An accelerated, reduced cost approach to overall lunar/Mars exploration is therefore expected.[6]

The Exploration Program Office (ExPO) at the Johnson Space Center has recently completed its review[7] of the Synthesis Group architectures and has initiated a course of action focusing on near-term, robotic precursor missions and a first lunar outpost (FLO) on the Moon in approximately the 1999 - 2002 time frame. Preliminary analysis at ExPO has indicated the desirability of delivering large, fully integrated payloads to the lunar surface (e.g., "turn-key" habitats) using a single HLLV in the 200 - 250 t range. With its potential for high specific impulse (Isp ~850 - 1000 s) and engine thrust-to-weight (~3 to 10), a NTR lunar transfer stage could significantly enhance the payload delivered to trans-lunar injection (TLI) conditions for a given HLLV capability.

An initial assessment of the feasibility of developing a NTR lunar transfer stage for FLO usage was performed by NPO with support from the Department of Energy and industry contractors. Referred to as the "Fast Track" Study, the assessment established NTR and stage characteristics, development schedules, and cost projections to achieve first flight in the 2000 - 2002 time frame. This paper describes results from the system and mission analysis portion of the Fast Track Study. The paper first reviews the FLO mission profile and describes the current space transportation system elements under consideration by ExPO. Characteristics of "state-of-the-art" NTR engines are then presented. Because of ExPO guidelines specifying maximum use of existing or "demonstrated" hardware components and systems to reduce schedule and development costs, NPO selected "proven" Rover/NERVA technology for its "reference" system in these initial assessments. Mission and transportation system ground rules and assumptions are presented next. These are used in determining attractive engine and stage characteristics which are subsequently compared with the present chemical FLO baseline. The impact on engine selection and vehicle design resulting from a consideration of alternative lunar mission profiles and NTR fuel forms is also discussed. Finally, a summary of the technical results and the conclusions reached in the study are presented.

FIRST LUNAR OUTPOST MISSION/SYSTEM OVERVIEW

Since 1987, NASA has spent considerable time assessing the human operations and surface support requirements needed to return humans to the Moon at levels ranging from short duration expeditionary landings to human-tended outposts, and ultimately to centralized bases supporting a substantial permanent human presence. The Synthesis Group also considered a spectrum of initial lunar operational capability in its four architectures. These varied in regard to their emphasis on exploration and science, human presence, space resource utilization, and Moon versus Mars focus.

Following its review of the Synthesis Group architectures, the ExPO has adopted a "lunar campsite" strategy[8] for FLO. Designed to provide facilities to support a crew of four for 45 Earth-days (i.e., a lunar day, night, day cycle), FLO consists of a pre-integrated, reusable habitat module delivered intact on a cargo lander. The outpost would be autonomously landed and its operational functions verified prior to crew arrival on a separate piloted flight. This predeployment of surface infrastructure via the split cargo and piloted mission approach is expected to improve overall mission success and reduce the amount of EVA required by the crew to prepare the outpost for initial occupancy. Because FLO is intended to be reusable, return visits to the same campsite are possible with resupply, or the outpost can be expanded to support larger crew and/or surface activities by landing additional surface assets. Alternatively, the campsite strategy also allows a second human-tended outpost to be established at other sites of interest.

Lunar Mission Profile Options

The selection of a particular lunar mission profile is strongly influenced by the HLLV assumptions, mission design requirements, and orbital mechanics constraints (see Figure 5). At the lower range of HLLV capability (~150 t), a single launch approach would most efficiently utilize the lunar orbit rendezvous (LOR) mission profile (see Figure 6). Here the piloted vehicle is separated into two spacecraft -- a LTV and a two stage LEV. The LTV contains the heavy crew transfer cab, and the propellant requirements for lunar orbit insertion (LOI) and trans-Earth injection (TEI). The LEV carries only the propellant necessary for lunar descent and ascent. By leaving the LTV in lunar orbit, a larger crew module and payload can be delivered to the lunar surface by the LEV for a given IMLEO. A single launch/LOR strategy was successfully utilized during the "Apollo Program" with the Saturn V HLLV delivering ~130 t to LEO and ~43 t (the combined weight of the Apollo command, service, and lunar excursion modules) to TLI conditions.

With a 150 t launch limit, a dual launch, Earth-orbit rendezvous and dock (EOR&D)/LOR approach (see Figure 2) can be used to assemble larger cargo and piloted vehicles having increased payload delivery and/or surface stay capability. Figure 7 shows the relative size and characteristics of the Apollo lunar excursion module (LEM), the 90-Day Study LEV, and the current FLO piloted lander concept. The reusable LEV (Figure 7b) was designed to have a significantly greater performance capability than the LEM. It could deliver ~15 t of cargo to the lunar surface and support a crew of 4 for up to 30 days. By contrast, the expendable LEM (Figure 7a) delivered less than a ton of cargo and supported a crew of 2 for a little more than 3 days on the lunar surface. A multiple launch, EOR&D/LOR strategy was baselined by NASA during its 90-Day Study and although it allows larger payloads to be delivered to the Moon, this mission scenario requires mastering a number of operational and technical challenges. Included among these are the need for (1) autonomous rendezvous and docking (already demonstrated by the former Soviet Union during its Mir/Progress resupply missions), (2) long term cryofluid management (involving both storage and

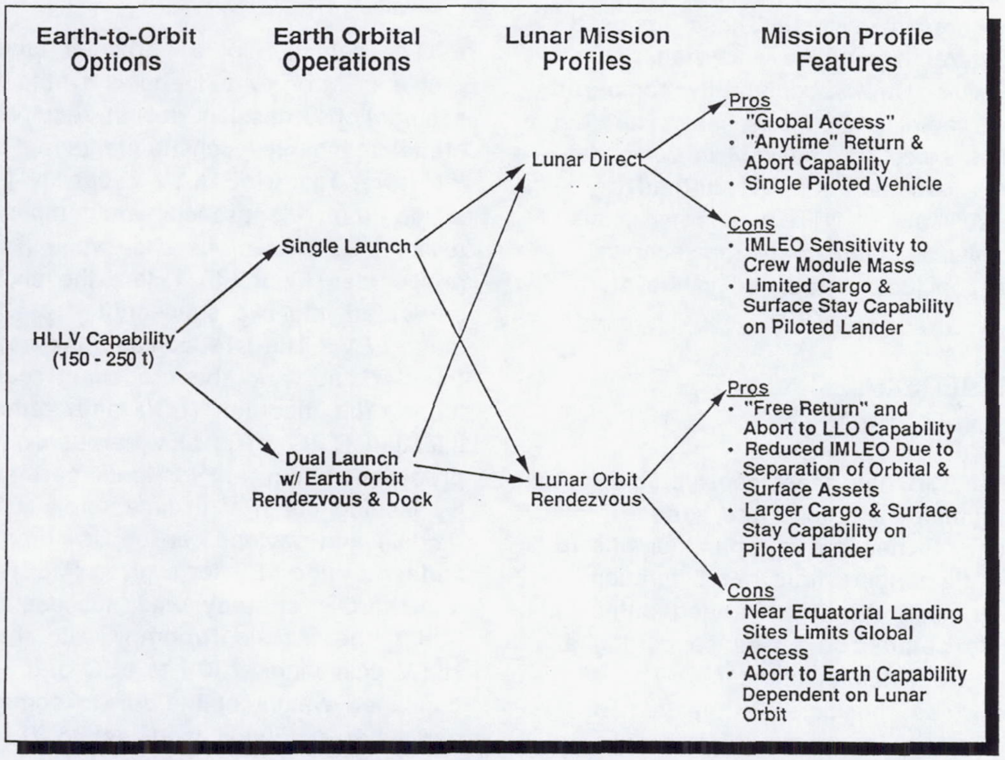


Fig. 5. Heavy Lift Launch Vehicle/Lunar Mission Profile Options

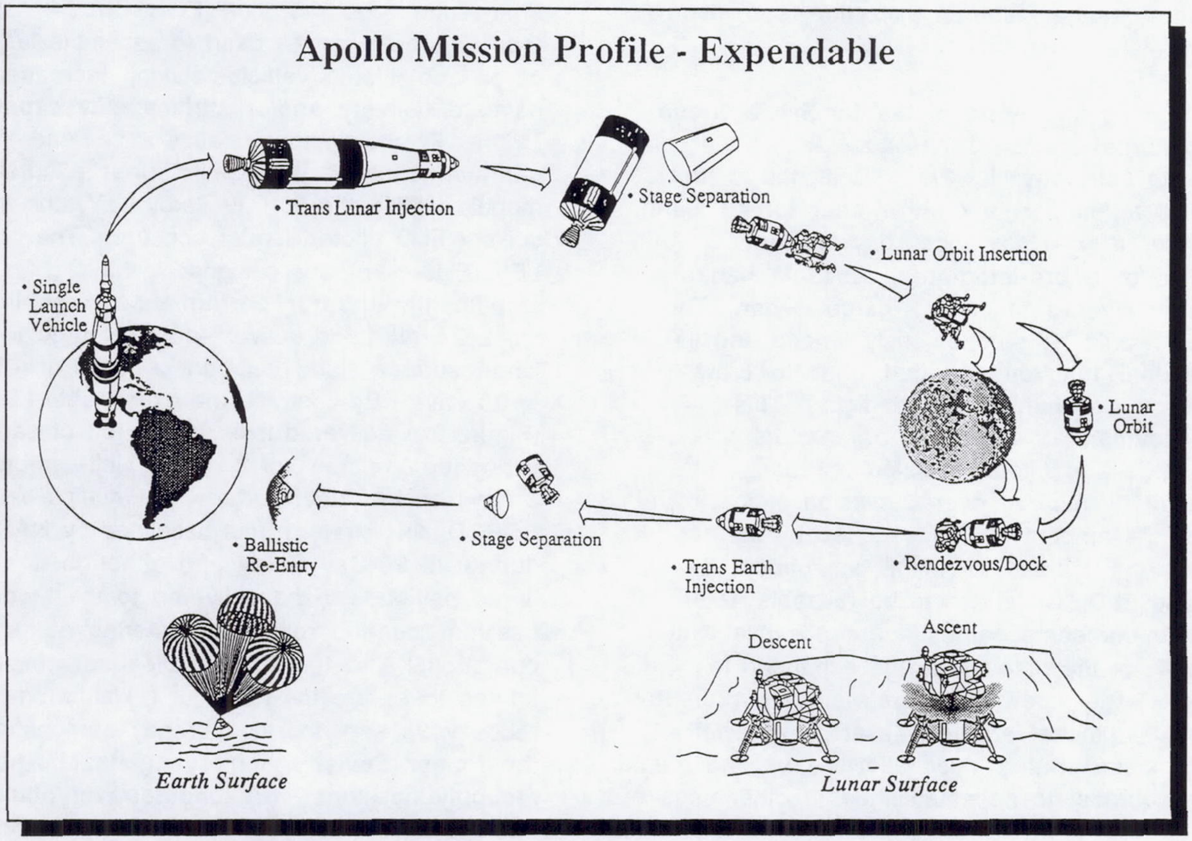


Fig. 6. Single Launch Lunar Orbit Rendezvous Mission Scenario

transfer), (3) micrometeoroid/debris protection and countermeasures, and (4) long term orbital maintenance and stationkeeping. While the above challenges may be viewed as potential unnecessary risks for the initial FLO mission, it should be remembered that each of these features is inherent in a piloted Mars mission along with the need for multiple HLLV launches, EOR&D of vehicle components, and Mars orbital rendezvous maneuvers between the primary interplanetary spacecraft and the Mars excursion vehicle.

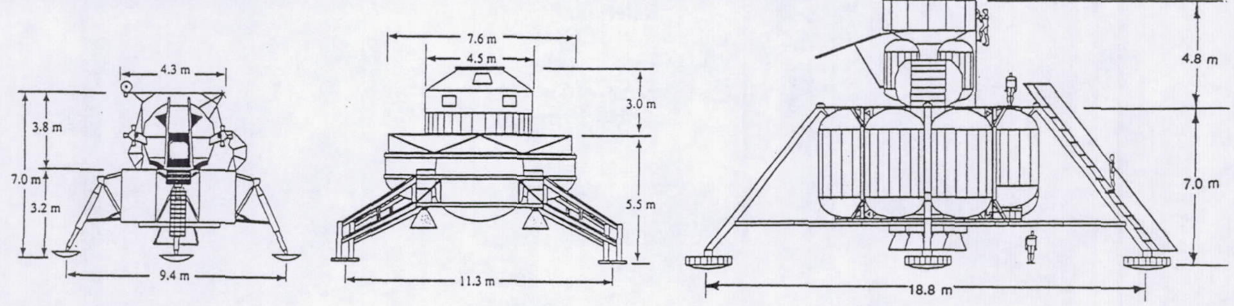
Besides using LOR, a dual launch, EOR&D strategy can also proceed using a "lunar direct" mission profile. In the lunar direct mode, a single integrated LTV/LEV design (see Figure 7c) is used for "in-space" transfer and lunar landing. Because the entire piloted vehicle is transported to the lunar surface, the lunar direct mode is very sensitive to variations in crew module mass and is also limited in the amount of cargo that can be transported with the crew. (Some of the pros and cons associated with each mission profile are identified in Figure 5).

At the upper HLLV range (~200 - 250 t), a single launch, lunar direct mission profile becomes possible. The ExPO is presently adopting this approach to provide a framework for its initial assessment of FLO. It is felt that the dual launch scenario would require increased launch costs and operational complexity both in terms of ground processing and in-space technology/systems requirements. Furthermore, because the short TLI window (~1 day per month for optimal conditions) must be closely synchronized with the second HLLV launch, a launch delay at the Kennedy Space Center could result in a costly one-month-long mission delay. The single launch strategy is expected to provide improved mission design flexibility (e.g., two daily TLI windows ~3 hours in duration) and reduced operational costs and risks.

FLO Transportation System/Mission Scenario Description

The FLO mission scenario assumes separate cargo and piloted missions with each vehicle requiring the launch of a single 200 - 250 t class HLLV. The sizing of the lunar transportation system elements for FLO was driven by several key requirements and assumptions levied by ExPO. These included: (1) a "global access" and "anytime return" capability, (2) a 45 day surface stay on the Moon with a crew of 4, (3) a 5 t resupply capability on the piloted missions, and the use of (4) cryogenic propellants for TLI, lunar orbit insertion (LOI) and descent, and (5) storable

Mass (t)	Apollo LEM		LEV	FLO Piloted Vehicle	
	Descent	Ascent	Ind. Crew Module	Lander	Return Stage
Dry	1.9	2.0	8.6	10.7	10.8
Propellant	8.2	2.4	22.3	45.7	18.6
RCS & Fluids		0.3	Uses Main Propellant	0.2	1.3
Gross	10.1	4.7	30.9	56.6	30.7



a.) **Apollo Lunar Module**
- Propellant: Storable
- Two Stage
- Crew Size: 2
- Surface Staytime: 3.25 days

b.) **Lunar Excursion Module**
- Propellant: LOX/LH2
- Single Stage
- Crew Size: 4
- Surface Staytime: 30 days (w/o support)

c.) **FLO Piloted Vehicle**
- Propellant: LOX/LH2 (Lander) Storable (Ascent)
- Two Stage
- Crew Size: 4
- Surface Staytime: 45 days (w/ Hab module)

Fig. 7. Relative Size/Characteristics of Lunar Landing Vehicles

propellants for lunar ascent and TEI.

The lunar transportation system elements for FLO are shown in Figure 8. They consist of a TLI stage, a common lunar lander, an Earth return stage, and a crew module all of which are expended during the course of the mission. The TLI stage uses a single J-2S engine (Isp = 436 s) with a thrust of 265 klbf for primary propulsion, and a monopropellant hydrazine (Isp = 237 s) reaction control system (RCS) for attitude control and stabilization. Aluminum alloy is utilized for structures and tankage. The stage contains ~133.5 t of liquid oxygen/liquid hydrogen (LOX/LH_2) propellant and has an inert mass of ~21.5 t. It is capable of injecting 96 t of payload to the Moon.

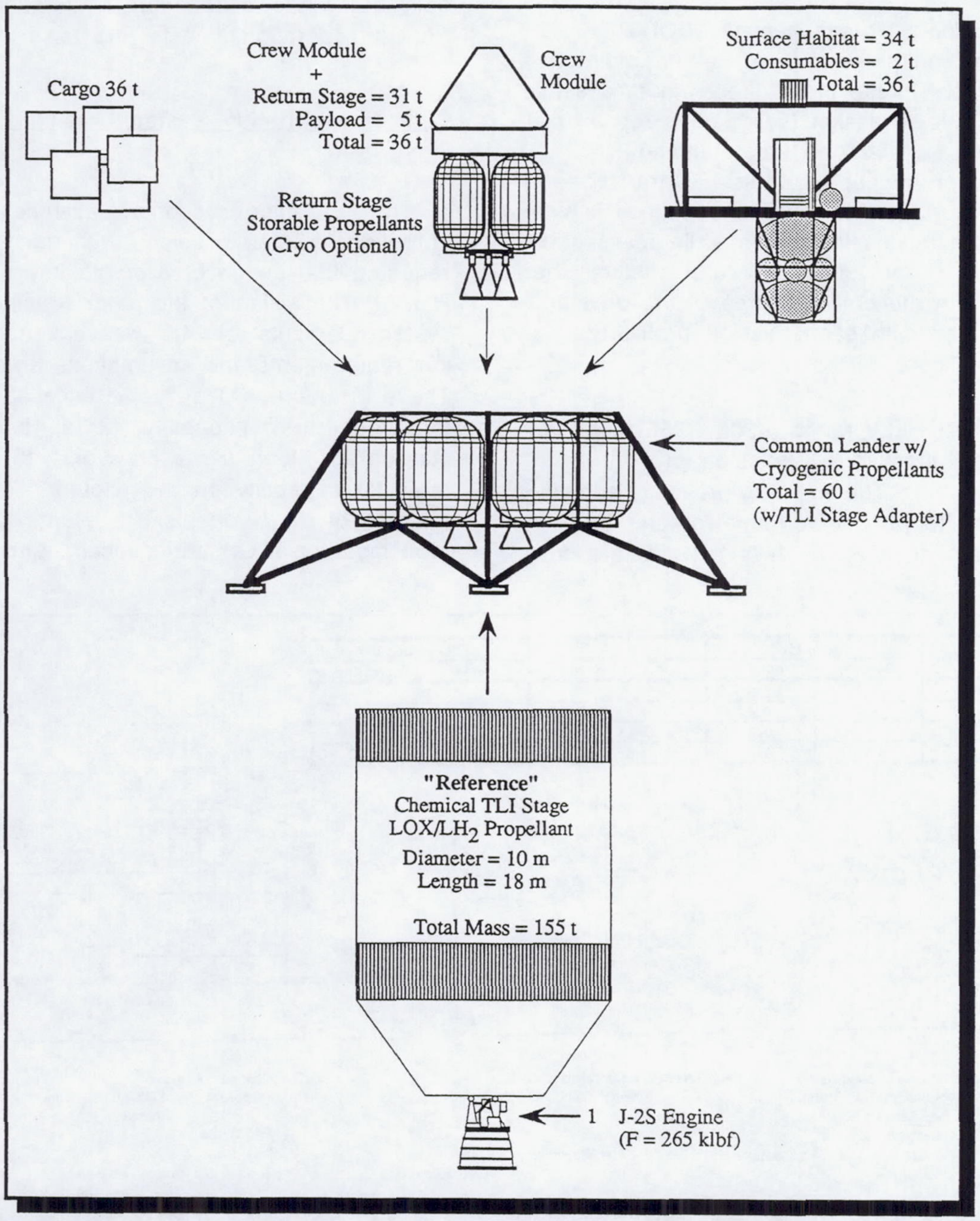


Fig. 8. FLO Space Transportation System Elements

The common lunar lander uses four RL-10 derivative engines (Isp = 444 s) which produce a combined thrust of ~80 klbf. With ~45.7 t of LOX/LH_2 propellant contained in its eight main propellant tanks and ~2.8t attributed to the TLI stage adaptor, the lander's gross mass is ~60 t. The return stage uses three Delta second stage engines (Isp = 320 s) having a combined thrust of ~30 klbf. The stage has a gross mass of ~24.1 t of which ~18.1 t is storable propellant contained in four main tanks.

The final element, the crew module, is an Apollo-shaped capsule upscaled by ~5% for the larger crew. The crew module is occupied for ~10.5 days (~8.5 days in space and 2 days on the lunar surface). During the remaining 43 days of the FLO surface stay the crew occupies the outpost while the return stage and crew module exists in a powered-down, dormant state. Air, water and power for the crew module during its occupied periods are provided by the return stage. During the Apollo program, the service module provided similar life-support functions to the Apollo command module.

The transportation elements shown in Figure 8 can be configured to fly in either a "cargo-only" or "piloted-plus-cargo" mode. In the cargo-only mode, the return stage and crew module would be replaced by an equivalent amount of payload which could include such items as surface habitats, crew consumable, rovers and science equipment.

Figure 9 depicts some of the key phases of the piloted mission scenario. The mission begins with the launch of a single 200 - 250 t HLLV to a circular 100 nautical mile (185 km) Earth staging orbit. Here the vehicle systems are checked out and verified prior to Earth departure. The expendable TLI stage is then fired placing the piloted vehicle on a 4-day trajectory to the Moon. After transfer to the Moon is complete, the lunar lander is used to propulsively capture the piloted vehicle into a temporary 100 km parking orbit. Pausing here allows time for navigational updates and phasing alignment over the desired landing site prior to final descent to the lunar surface. When the surface mission is completed, the crew reenters the return stage and ascends to its earlier parking orbit prior to initiating TEI. Nearing Earth, the crew module separates from the return stage and performs a direct Earth entry while the return stage is expended in cislunar space via an Earth fly-by.

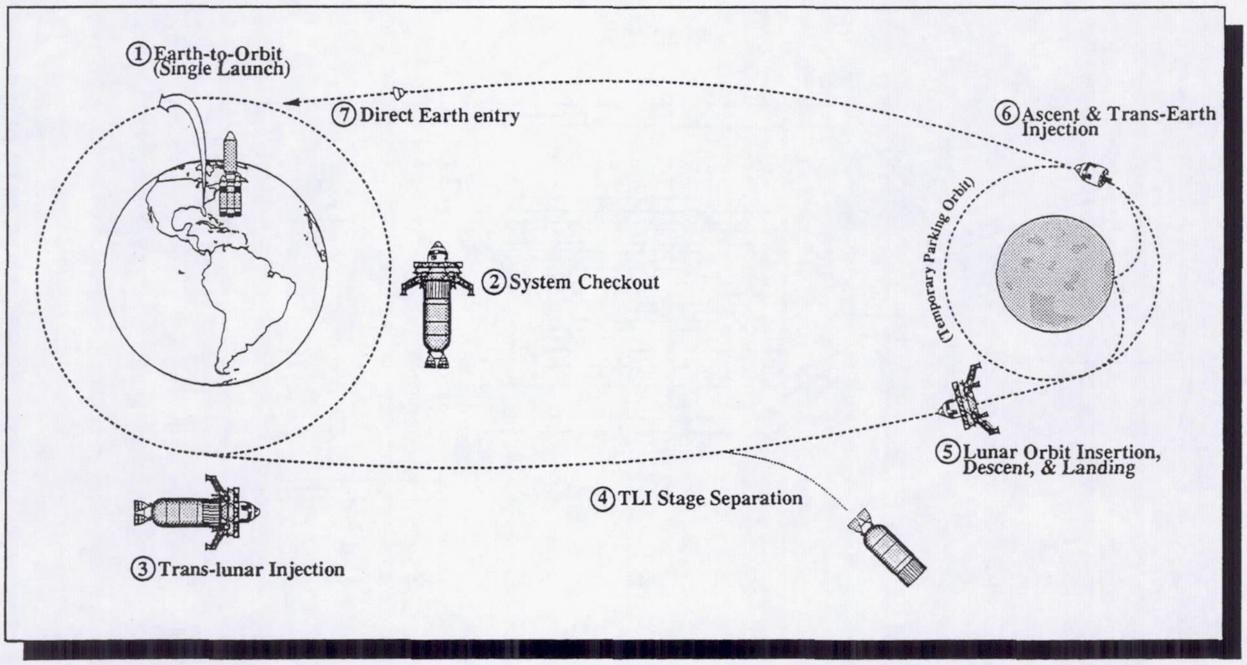


Fig. 9. "First Lunar Outpost" Piloted Mission Scenario

NUCLEAR THERMAL ROCKET SYSTEM DESCRIPTION

The nuclear thermal rocket represents the next major evolutionary step in propulsion technology and is expected to be an important complement to chemical propulsion for NASA's SEI missions. Conceptually, NTR systems are relatively simple (see Figure 10). They function by raising hydrogen propellant to high pressure in a turbopump assembly, passing it through a high power reactor where it is heated to high temperatures, and then exhausting it through a nozzle at high speeds to generate thrust. Because a fission reactor, rather than chemical reactants, provides the heat source, the NTR can use low molecular weight liquid hydrogen as both the reactor coolant and propellant and achieve specific impulse values nearly twice that of conventional LOX/LH_2 fueled chemical rockets at comparable exhaust temperatures.

In the "expander cycle" engine shown in Figure 10, the turbine drive gas is routed to twin turbopumps (used for redundancy and improved system reliability) and then through the reactor core allowing the entire propellant flow to be heated to design conditions. Hydrogen flowing from the pumps would be split with a portion being used to cool the nozzle, reflector, control rods and internal dome shield, and the remainder going to the core support tie tubes (not shown in Figure 10) for cooling and providing the necessary turbine drive power.

A workshop was conducted by NASA, DOE and DOD in July 1990 to identify and evaluate candidate NTR concepts.[9] Over seventeen concepts were presented including solid, liquid, and gaseous core systems. The solid core concepts are considered to be lower technical risk, and are presently being evaluated by NASA.[10] In keeping with ExPO design guidelines specifying maximum use of existing or near term hardware to reduce schedule and system development costs, the demonstrated technology base of the Rover/NERVA programs was chosen for the Fast Track Study.

Rover/NERVA Technology Overview

The feasibility of a hydrogen-cooled, graphite moderated NTR was demonstrated by the Rover nuclear rocket program[11] begun at Los Alamos in 1955. The promising early results from this effort led to the formation in 1960 of a joint

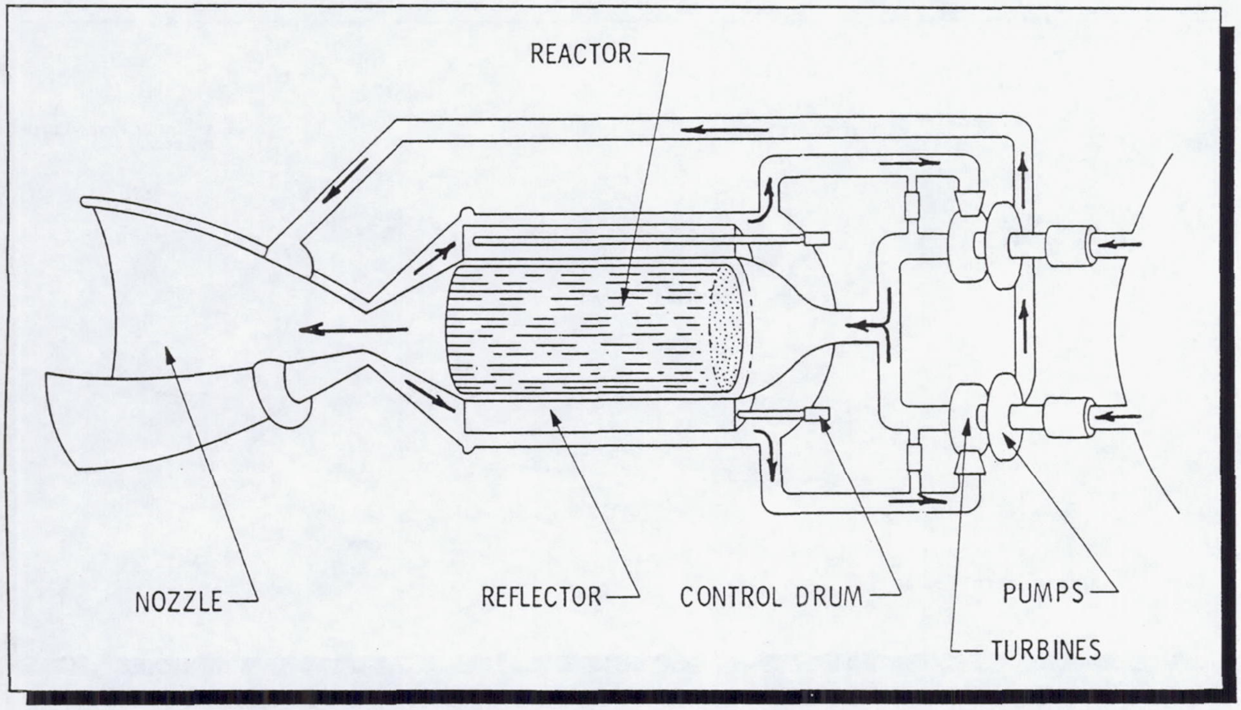


Fig. 10. Schematic of Dual Turbopump Expander Cycle NTR

program between NASA and the Atomic Energy Commission (AEC) to develop a Nuclear Engine for Rocket Vehicle Application (NERVA).[12] From 1955 until the program was stopped in 1973, a total of twenty reactors were designed, built and tested at a cost of ~\$1.4 billion. Escalated to 1992 dollars, Rover/NERVA technology represents and investment of ~\$10 billion. The accumulated experience of the Rover/NERVA programs can be seen in the results achieved in the last 6 reactor tests conducted between 1967 and 1972 (see Table 1).

At the heart of the NERVA reactor design is a 52" long hexagonally-shaped fuel element (0.75" across the flats) which is capable of producing approximately 0.9 to 1.2 megawatts of thermal power (MWt) (see Figure 11). Each fuel element has 19 axial coolant channels which along with the outer element surfaces, are coated with zirconium carbide (ZrC) to reduce hydrogen/graphite reactions. A "2-pass" regeneratively-cooled, tie-tube assembly supports from 3 to 6 fuel elements forming a fuel bundle (shown in Figure 11). Specifying the engine thrust level, hydrogen exhaust temperature (or equivalent Isp), and the fuel element power density determines the reactor power output and sets the core diameter and number of fuel bundles required in the engine. For lower thrust engines criticality can be achieved with reduced core diameters and acceptable thrust-to-weight ratios by augmenting the moderating capability of the graphite core with additional zirconium hydride (ZrH) neutron moderator. The ZrH is contained in the tie-tube support elements which are increased in number for lower thrust engines by decreasing the fuel-to-support element ratio (from ~6 to 1 for engine thrust levels greater than 50 klbf down to ~3 to 1 for a 25 klbf-class engine).

Two fuel forms were tested[11] during the Rover/NERVA programs which have the potential

Table 1. Accumulated Experience Base from Rover/NERVA Reactor Tests

Last 6 Rover/NERVA Program Reactor Tests

• Phoebus-1B : (1967)	1500 MWt/75 klbf 30 min burn duration @ full power
• NRX-A6 : (1967 - 68)	1100 MWt/55 klbf 62 min burn duration @ full power
• Phoebus-2A : (1968)	4100 MWt/200 klbf 12 min burn duration @ full power Demonstrated regeneratively cooled support elements
• Pewee : (1968)	500 MWt/25 klbf 20 min burn duration @ full power 1.2 MWt/fuel element @ T_{ex} = 2550°K Demonstrated use of ZrH on support elements
• NRX-XE : (1968 -69)	1100 MWt/55 klbf 28 start-up/shutdown cycles with 115 minutes of operation @ partial and full power
• NF-1 : (1972)	Fuel Element Test Reactor/examined "composite" and "carbide" fuel forms 109 minutes accumulated (4 tests) at 44 MWt

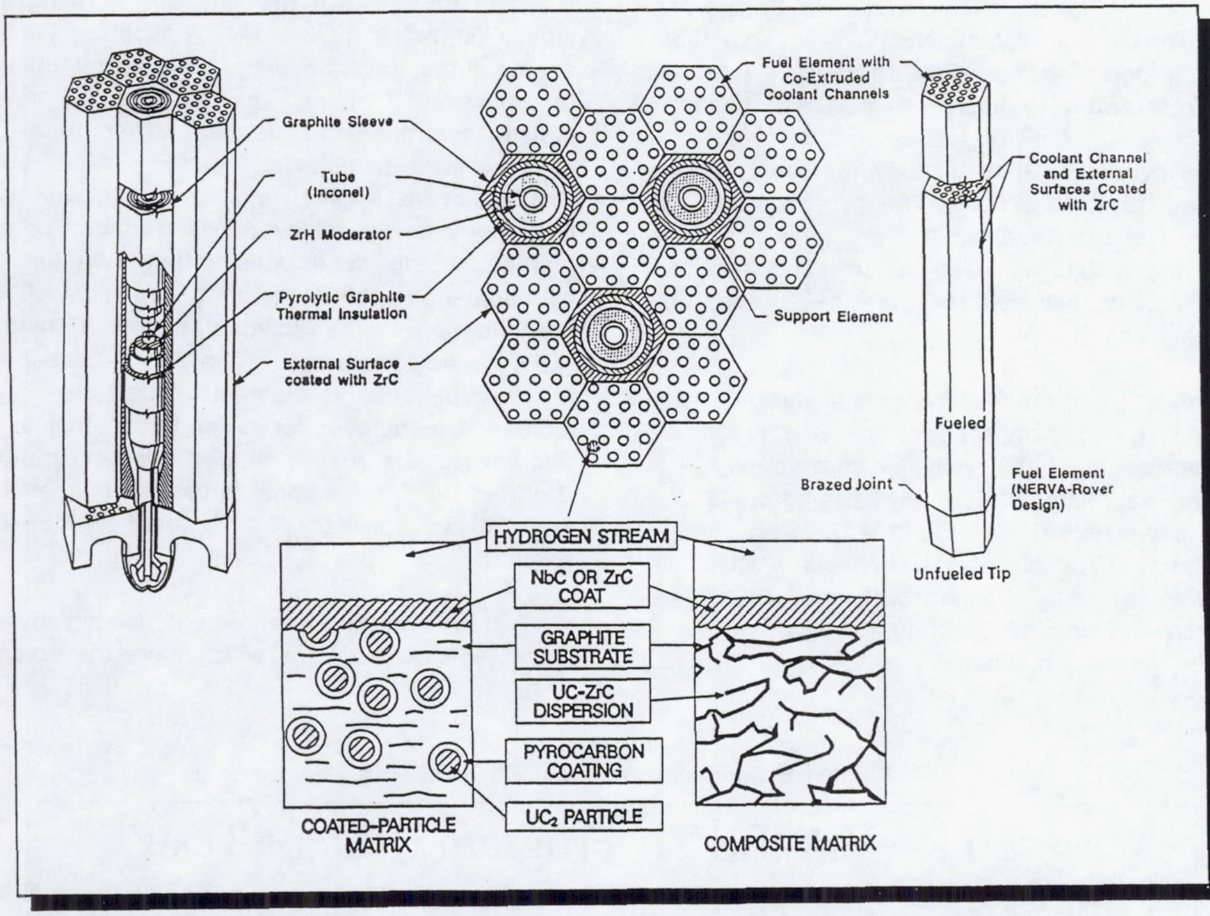


Fig. 11. Rover/NERVA Fuel Element Configuration

for near term applications in a Fast Track NTR development scenario. The vast majority of experimental data was obtained using a "graphite fuel" form. It consisted of pyrocarbon coated uranium carbide (UC_2) fuel particles which were dispersed in a graphite substrate (see Figure 11). This fuel was operated at hydrogen exhaust temperatures as high as 2550 K. The second fuel form was a "composite fuel" which consisted of a UC-ZrC dispersion in the graphite substrate. Although the composite fuel received only limited nuclear testing in the Nuclear Furnace (NF-1), it has the prospect of potentially providing exhaust temperatures as high as 2700 K.

Fast Track Engine Design Strategy

The goal of the Fast Track Study was to determine the feasibility of developing a NTR-powered lunar transfer stage in the 2000 - 2002 time frame to support cargo and piloted missions of the type envisioned for FLO. In determining the characteristics of NTR engines, a design approach was adopted which (1) stressed maximum use of demonstrated systems and technologies, and (2) emphasized safety, reliability and modest performance gains rather than focusing on achieving the highest fuel temperature, specific impulse or engine thrust-to-weight ratio. These criteria led the NPO study team to the selection of Rover/ NERVA-derived technology and to coated UC_2 particles in graphite as the reference configuration. Composite fuel was specified as a backup or follow-on fuel form.

To anchor the Fast Track engine designs to demonstrated operating conditions, the 500 MWt, 25 klbf-class Pewee reactor system[13] provided an initial starting point. The Pewee reactor used a 52" long graphite fuel element capable of producing ~1.2 MWt and of operating at hydrogen exhaust temperatures of 2550 K (a Rover/ NERVA program performance record). The Pewee fuel element and operating temperature was recommended as a reference point for subsequent reactor analysis and engine design work by the industry contractor team of Rocketdyne and Westinghouse who participated in the Fast Track Study. A broad range of single and multi-engine stage configurations and engine thrust levels (extending from 10 klbf to 125 klbf) were examined by the NPO. Because of the deterioration in engine thrust-to-weight ratio at the lower thrust levels resulting from criticality considerations, engine configurations with and without ZrH moderator augmentation in the tie-tube supports elements were examined. Finally, modest performance and design targets of Isp ~870 s and engine thrust-to-weight ≥ 3 (with internal shield) were specified in keeping with the "Model T"-type NTR design philosophy assumed in this study.

Engine Sizing Results

Figure 12 shows engine weight scaling data for NERVA-derived NTR engines operating with and without ZrH moderator augmentation. Achieving the specific impulse design goal of 870 s and satisfying an initial engine length limit of ~6 m for a 25 klbf-class engine resulted in a chamber pressure of ~785 psia, a nozzle area expansion ratio of 200 to 1 and a 110% length optimum contour Rao nozzle. An expander cycle was baselined in this study with turbine drive gas provided by the reactor tie-tube support elements. These same pressure and nozzle conditions were maintained for engine point designs at the 50 and 75 klbf thrust levels. The relative size of these three NERVA-derived NTR engines is shown in Figure 13.

To assess the performance of lower thrust

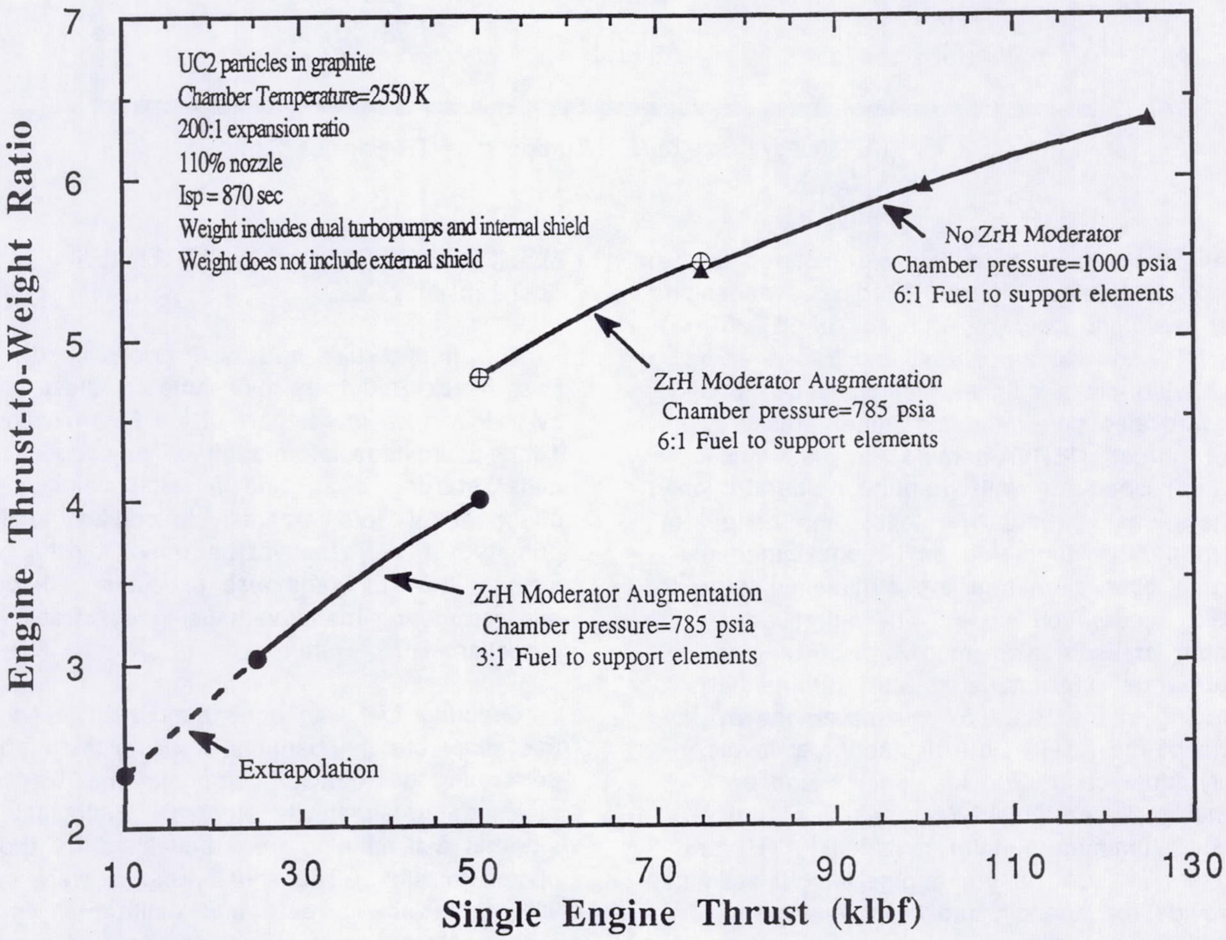


Fig. 12. NERVA - Derivative Engine Weight Scaling

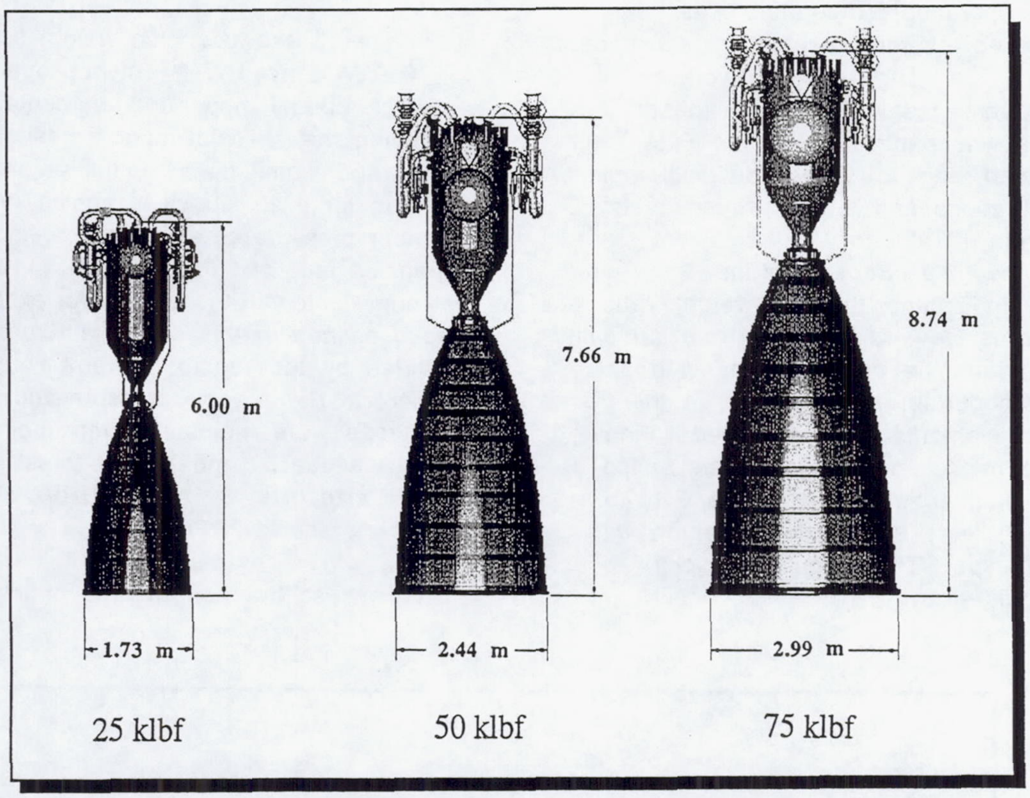


Fig. 13. Relative Size of Dual Turbopump NTR Engines

engines the scaling data was extrapolated to the appropriate levels. Figure 12 indicates an engine thrust-to-weight ratio of ~2.3 for a 10 klbf-class NTR and values of 3, 4, and 5 for the 25, 50, and 75 klbf-class engines, respectively. Scaling data was also generated for higher pressure, higher thrust NERVA-derived engines (up to 125 klbf) operating with graphite moderator only. At the higher thrust/power levels the benefit of ZrH moderator augmentation becomes marginal because core diameters are sufficiently large for graphite moderation alone. The increased chamber pressure also improves both engine performance characteristics and engine/stage packaging in the HLLV by minimizing the overall growth of the NTR at the higher thrust levels. Finally, dual centrifugal turbopumps and an internal radiation shield (comprised of boron-carbide aluminum-titanium hydride (BATH) and lead) are included in our engine weight estimates to provide redundancy, and improve engine reliability and safety.

MISSION/TRANSPORTATION SYSTEM GROUND RULES AND ASSUMPTIONS

The ground rules and assumptions used in the Fast Track Study are the same as those used by ExPO in its assessment of the FLO mission. Table 2 provides information on payload masses, initial starting orbit, and mission velocity changes (ΔV) requirements. In addition to the primary TLI ΔV maneuver performed by the NTR system, the TLI stage also executes mid-course and retargeting maneuvers using a storable propellant RCS system.

Graphite fuel was used almost exclusively in this study but the benefits of using the higher performing but heavier composite fuel was also assessed in sensitivity studies. Biological external disk shields were baselined for the piloted mission. The shield weights were scaled with thrust/power level and calibrated with earlier NASA contractor studies [14,15] of lunar NTR

Table 2. FLO Mission Ground Rules and Assumptions

"One Burn" Lunar Scenario

•TLI Payload	96 t (piloted vehicle & TLI stage adaptor)	
•TLI Maneuver	ΔV	3200 m/s + gravity losses
	Initial orbit	100 n. mi. circular LEO (185 km)
•NTR System	Propellant	Cryogenic hydrogen
	Isp	870 sec (graphite) / 900 sec (composite)
	External Shield Mass	≈ 60 kg/ klbf thrust
	Burn Duration	≤ 30 minutes
	Flight Performance Reserve	1% of usable propellant
	Cooldown (effective)	3% of usable propellant
	Residual	1.5% of total tank capacity
•RCS System	Propellant	Hydrazine
	Isp	237 sec
	TLI burnout ΔV	60 m/s (30 m/s for trailing edge lunar flyby)
•Tankage	Material	2219-T87 Al
	Diameter	10 meters
	Geometry	Cylindrical tank with √2/2 domes
	Insulation	2 " MLI + micrometeoroid shield (3.97 kg/m^2)
	Boiloff	12.40 kg / day
•Contingency	Engine & external shields	15%
	All other dry masses	10%

stages conducted in the 1960's and the early 1970's. Allowances for flight performance reserve, post-burn reactor cool down and tank trapped propellant residuals were also accounted for in estimating the total propellant requirements for the mission.

"Off-the-shelf" aluminum alloy was specified by ExPO for structure and cryogenic tank construction. In this study aluminum alloy 2219-T87 (F_{tu}=62 ksi, ρ= 2821 kg/m^3) was utilized for structure and the LH_2 propellant tank(s). This selection is due to its favorable properties at cryogenic temperatures and its extensive use in cryogenic tank construction. It has a relatively high strength-to-density ratio, good toughness and availability, is weldable and low in cost. Alloy 2219-T87 plate is also presently used for the LOX/LH_2 external tank used on NASA's Space Shuttle. Tank thicknesses were calculated assuming a maximum internal pressure of 35 psi (241.3 kPa) and included hydrostatic loads using a "4-g" load factor along with a safety factor of 1.5. A 2.5 percent ullage was also assumed.

A two inch helium-purged, multilayer insulation (MLI) system (at 50 layers per inch) was assumed for thermal protection of the NTR stage LH_2 tank. This insulation thickness exceeds the requirements for the short duration (≤ 8 hrs), "one burn" FLO mission, as well as, the "ground hold" thermal protection requirements for "wet-launched" LH_2 tanks (a minimum of 1.5 inches of helium-purged insulation).[16] Its use in this study ensures extra margin and also provides the capability for longer duration lunar missions (~30 - 180 days in lunar orbit). The installed density of the "2 inch MLI system" is ~2.62 kg/m^2 and the resulting boiloff rate is ~0.77 kg/m^2/month (based on an estimated heat flux of ~0.129 W/m^2). Finally, one 0.5 mm sheet of aluminum (corresponding to ~1.35 kg/m^2) was assumed for micrometeoroid protection on the stage's LH_2 tank.

LUNAR NTR MISSION DESCRIPTION

A mission profile analogous to FLO was used to identify attractive engine/stage configurations. As illustrated in Figure 14, the mission begins with a single HLLV launch that delivers the lunar NTR stage and piloted lander to a 185 km circular Earth orbit. Following a systems checkout and verification period which can last up to ~8 hours, the NTR stage performs the TLI maneuver placing both it and the piloted vehicle on a trans-lunar trajectory. Although a single engine burn in excess of one hour was demonstrated by the NRX-A6 reactor during the NERVA program (see Table 1), a maximum single burn duration of 30 minutes was assumed in this study to provide margin and enhanced mission success probability.

Following an appropriate cool down period, the piloted vehicle and NTR stage separate with the piloted vehicle continuing on its nominal mission while the NTR stage executes a retargeting maneuver with its RCS system to perform a "trailing edge" lunar swingby. The resulting lunar gravity assist is used to deliver the "spent" NTR stage to a long-lived (~10^5 year) heliocentric orbit with minimal risk of Earth reencounter.

ENGINE/STAGE SIZING ANALYSIS

Determining attractive engine/stage configurations for FLO was one of the principle activities in the Fast Track Study. Figure 15 shows the IMLEO required to deliver 96 t (the mass of the current FLO piloted vehicle) to TLI conditions, as a function of engine thrust level for single and multi-engine stage designs. Each curve represents a "family of vehicles" which are similar in terms of the number of engines and the stage geometry (e.g., all LH_2 tanks are cylindrical with 10 m diameters and $\sqrt{2}/2$ ellipsoidal upper and lower domes). The configurations vary, however, with regard to the total length of the LH_2 tank and the physical dimensions of the engine(s) used.

Figure 15 also shows that, for a given "total" thrust level, multiple engine configurations have a higher IMLEO. This is due in part to the buildup of inert weight from multiple engine components (e.g., pumps, lines and valves, shielding, etc.) in a "clustered" configuration, and also to the deterioration in the engine thrust-to-weight ratio for lower thrust NTR systems (shown in Figure 12). For example, assuming a total thrust

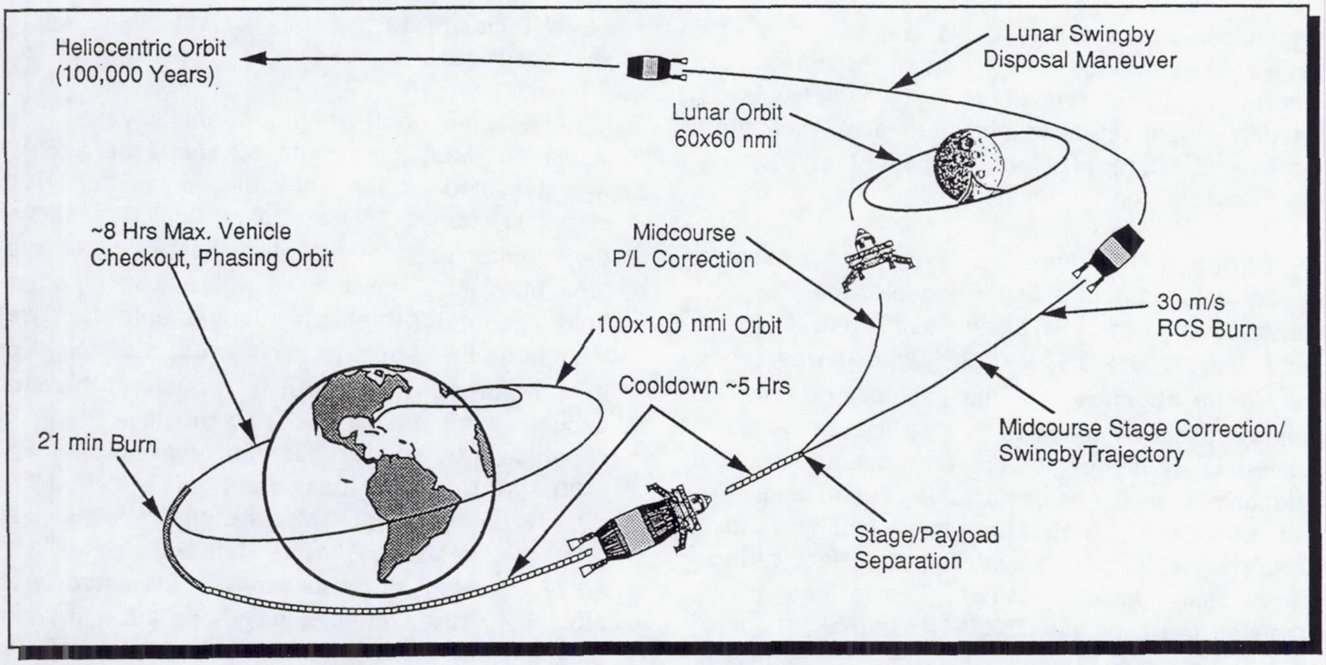


Fig. 14. NTR - Based FLO Mission Scenario

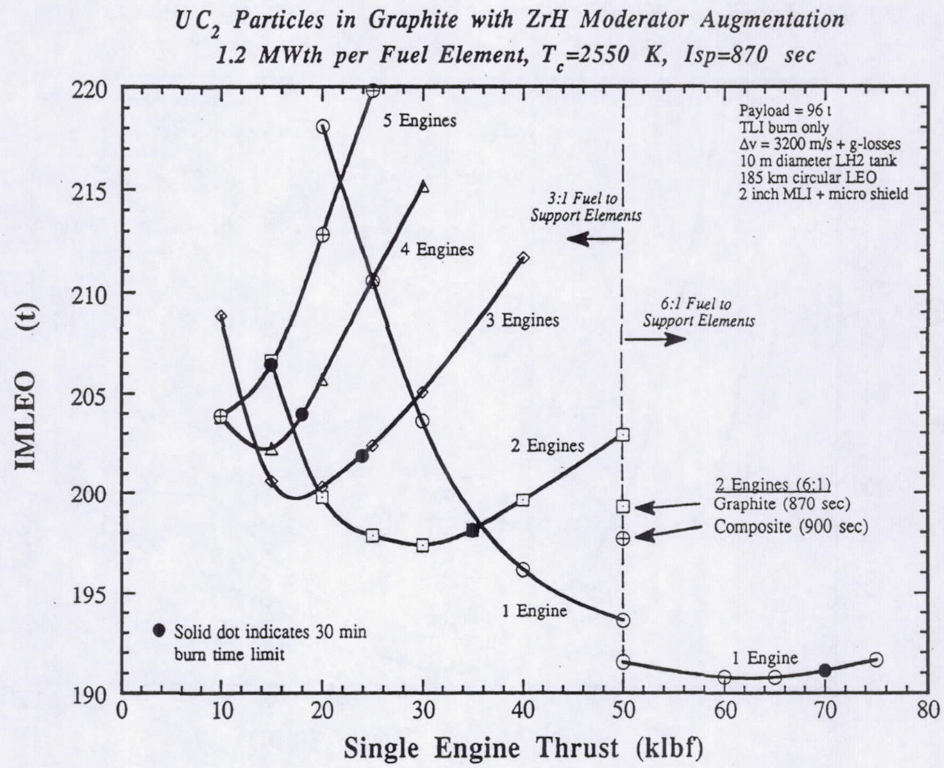


Fig. 15. "First Lunar Outpost," IMLEO Sensitivity to Single Engine Thrust Level

level of 75 klbf, the five 15 klbf engine configuration has the largest IMLEO at ~206.4 t, followed by the three 25 klbf engine vehicle at ~202.4 t. The single 75 klbf vehicle has the lowest IMLEO at ~192.9 t.

Each curve in Figure 15 also exhibits a distinct minimum in IMLEO. It is at this point that the optimum engine thrust level (with respect to IMLEO) is found. At higher thrust levels, or to the right of the optimum engine size, the propulsion system mass is excessive and leads to an increase in IMLEO despite the mass savings resulting from reduced gravity losses. Conversely, at the lower thrust levels, or to the left of the minimum IMLEO, reductions in propulsion system mass due to lower total thrust are offset by the additional propellant and tankage mass associated with the higher gravity losses.

To prevent the TLI burn times from becoming excessive and to provide margin for the remaining engine(s) in case of an "engine out" occurrence, a "30 minute limit" on burn time (represented by the solid dot on each curve) was specified. This burn time constraint is violated to the left of the dots (e.g., four 10 klbf engines and three 15 klbf engine configurations require burn times of 59.2 and 50.6 minutes, respectively), while points to the right of the solid dots have burn times less than 30 minutes. As points of comparison, the single J-2S engine used on the chemical TLI stage burns for ~7.9 minutes, while the three 25 klbf, two 50 klbf and single 75 klbf engine configurations have burn times of 28.7, 20.8 and 27.3 minutes, respectively. In all the curves shown the optimal thrust levels corresponding to the minimum IMLEO exceeded the burn time constraint and were not considered further. The "constrained minimum IMLEO" for each curve is at the 30 minute burn time limit.

Figure 16 depicts the single, two, and three engine curves from the previous figure (with ZrH moderator augmentation) along with a curve portraying single engine configurations without ZrH moderator augmentation. The 100 and 125 klbf-class single engine stage configurations have IMLEO and burn time values of 195.2 t and 199.7 t, and 20.3 and 16.5 minutes, respectively. Although a single 75 klbf engine stage design has

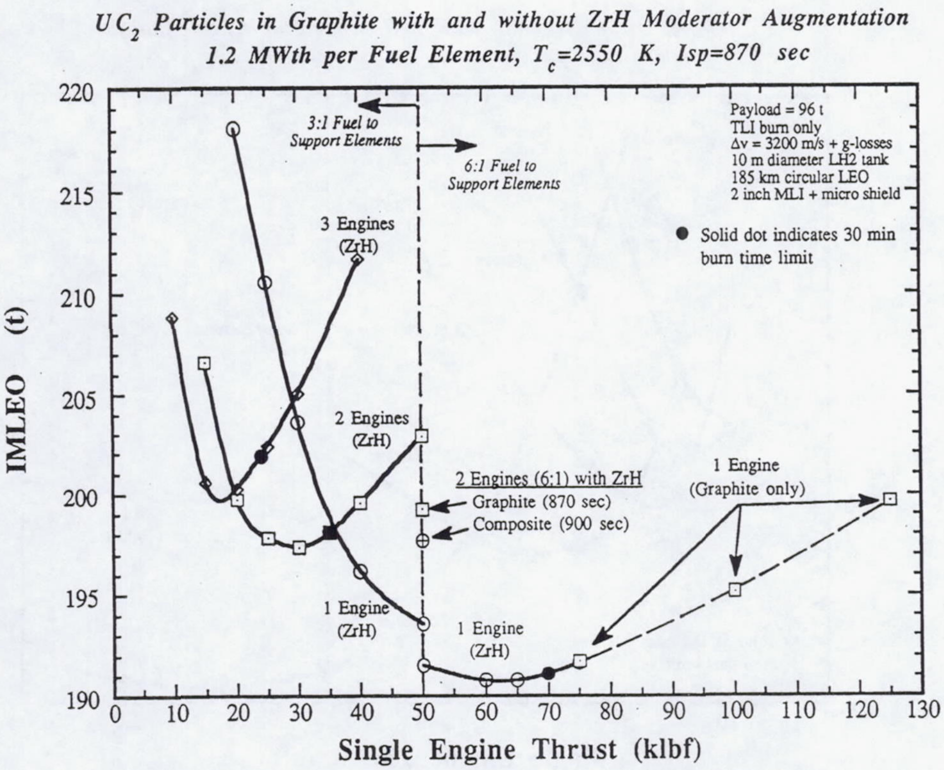


Fig. 16. FLO IMLEO Sensitivity to Single Engine Thrust Level with/without ZrH Moderator Augmentation

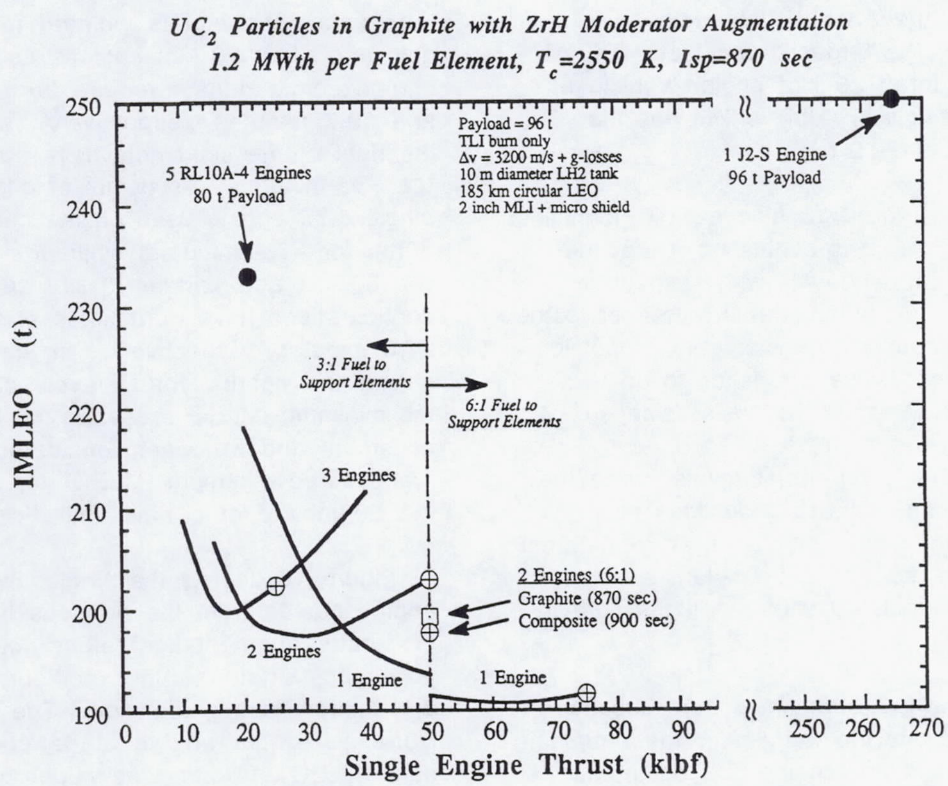


Fig. 17. Benefits of NTR Propulsion for "First Lunar Outpost"

the best performance in terms of IMLEO, a two engine configuration using 50 klbf NTRs has been chosen as the reference system because of its "engine out" capability and the attractiveness of the clustered 50 klbf vehicle configuration for Mars cargo and piloted missions[6]. A large experimental database also exists on 50 klbf-class engines (the KIWI-B and NRX reactor series) from the earlier Rover/NERVA programs.

Figure 17 compares the IMLEO for FLO using NTR and chemical propulsion TLI stages. All of the NTR stages considered have a lower IMLEO than the current chemical reference system which uses a single J-2S engine producing ~265 klbf of thrust. A clustered engine configuration using five RL10 A-4 engines (but delivering only 80 t to TLI conditions) is also indicated for comparison. Figure 17 illustrates quite dramatically that NTR propulsion can significantly enhance the performance capability for the FLO mission.

LUNAR NTR STAGE DESCRIPTION

A representative NTR-powered lunar transfer stage using three 25 klbf-class NERVA-derived NTRs is illustrated in Figures 18 and 19. The "reference " NTR stage for FLO is shown in Figure 20 and its mass properties are provided in Table 3. The main LH_2 propellant tank has a 10 m diameter, ~14.8 m length and √2/2 ellipsoidal domes. The tank is constructed of 2219-T87 Al, has a LH_2 propellant capacity of ~67.9 t (with an assumed 2.5% ullage), and is designed to handle "4 g" launch loads under fully-fueled and loaded conditions. Avionics, power and RCS are located in the stage forward adaptor section. During launch, loads from the lander and TLI stage are transferred to the HLLV through a cylindrical ring or "skirt" located at the aft end of the tank. Fairings for the lander and tank MLI protection carry only aerodynamic loads and are expended before TLI. In-space thrust loads from the two 50 klbf NTRs are transferred to the vehicle through the rear conical adaptor or "thrust structure". The propellant feed system includes two boost pumps to supply the pressure differential, and to allow a restart capability.

An external disk shield for crew radiation protection is also assumed on each engine at present. Because of the substantial quantities of cryogenic and storable propellant between the crew and engines, it may be possible to reduce or even eliminate the need for external shielding. Analysis is presently on-going with the DOE to determine actual shielding requirements for the FLO stage.

With regard to size and mass difference between the NTR and chemical TLI systems, the

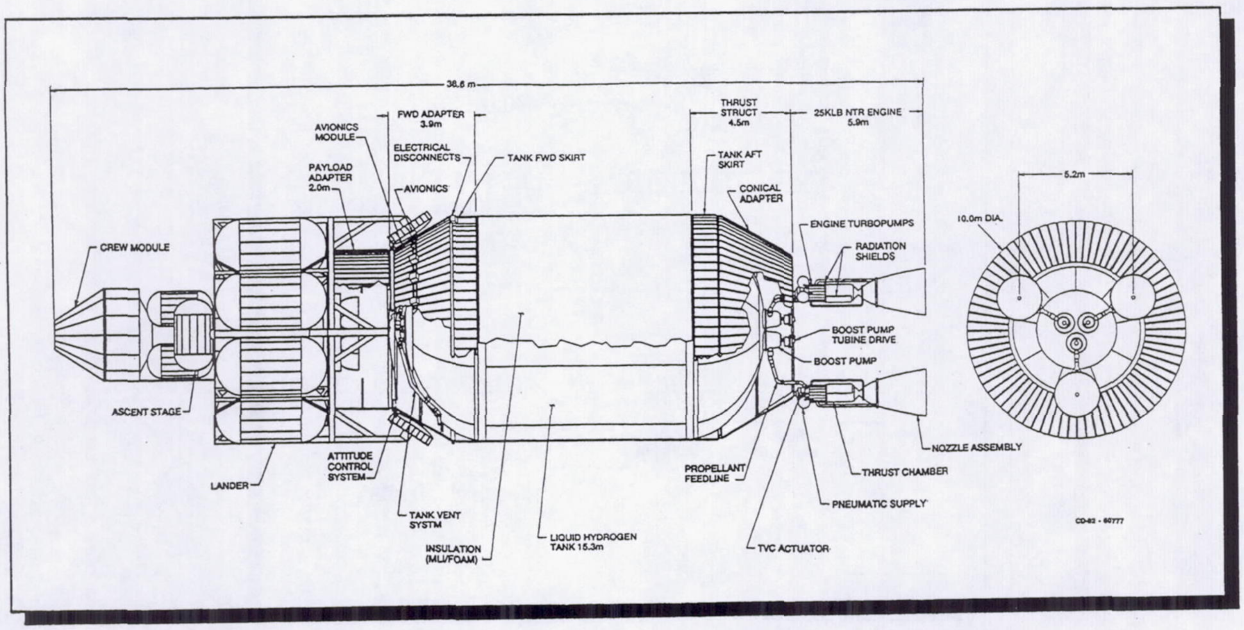


Fig. 18. Three Engine NTR Transfer Stage for FLO

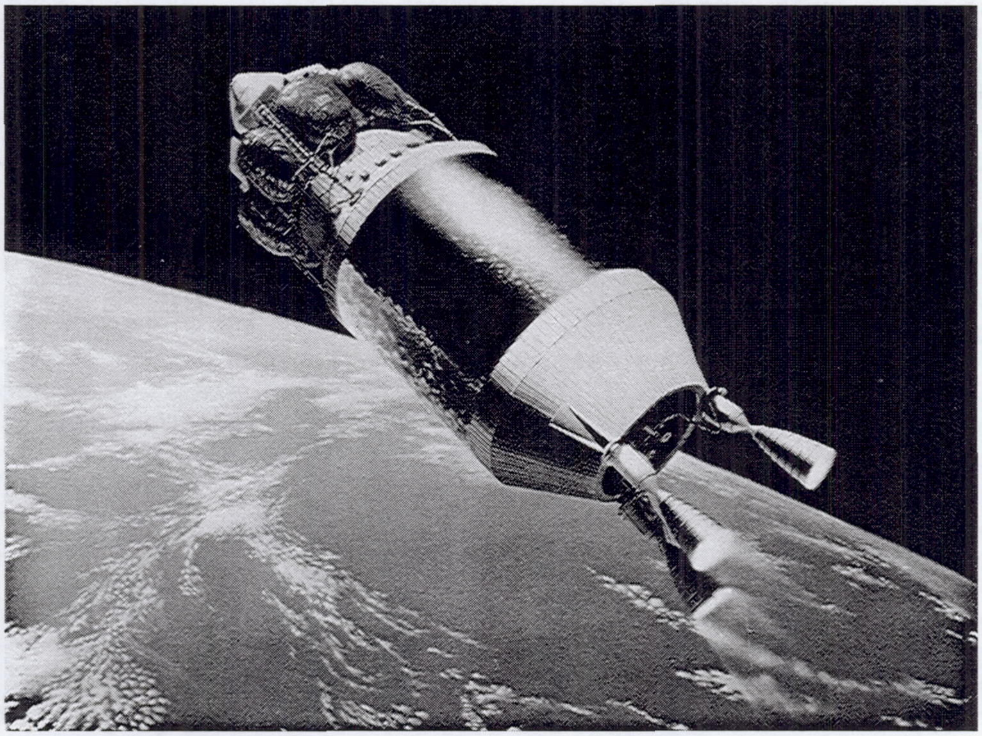


Fig. 19. Artist's Illustration of NTR Lunar Transfer Stage

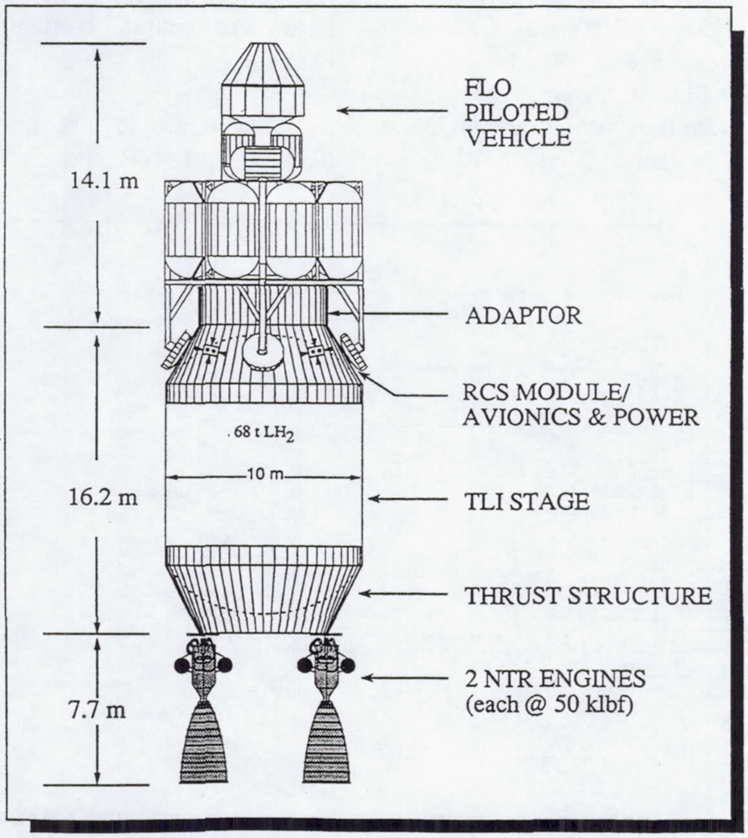


Fig. 20. Reference NTR Vehicle Configuration for FLO

Expendable FLO TLI Stage

Two 50 klbf NTR engines

		Mass in kilograms	
Structure		12955	
Tank Forward Adaptor	670		
Tank Forward Skirt	494		
Tank Aft Skirt	1674		
Conical Adaptor	486		
Main Propellant Tank	7646		
TPS + Micrometeoroid Shield	1985		
Feed System		207	
Feed Lines and Valves	62		
Manifolds, gimbal joints, insulation	54		
Boost pumps	91		
He System		29	
Base Heat Shield		156	
TVC		181	
Avionics and Power		998	
RCS Hardware		454	
NTR Assemblies		15526	
Engines (2)	9526		
External Shields (2)	6000		
Contingency		3827	
Dry Mass			**34333**
H2 Propellant Load		67878	
RCS Propellant		1038	
He		13	
Stage Mass			**103262**
Payload		96000	
Payload Fairing		10823	
Tank MLI Protective Fairing		2251	
Booster Adaptor		7411	
Booster Payload			**219747**

Table 3. Reference Vehicle Weight Statement

total length of the NTR TLI stage along with the FLO lander is ~38 m compared to ~32 m for the chemical system. From a mass standpoint however, the NTR system is ~51 t lighter than its chemical counterpart. Although the NTR stage is assumed to depart from the same 185 km altitude as the chemical system, higher LEO starting altitudes are readily achieved. Sensitivity analysis[17] conducted on the stretched Saturn V-derived HLLV indicate a payload versus altitude tradeoff of -0.2 t per each additional nautical mile the payload is lifted. This result indicates that the lunar NTR stage and its payload could also be delivered to a substantially higher starting altitude (~500 km) if desired from an overall safety or public acceptance standpoint.

ALTERNATIVE FUEL FORMS/MISSION PROFILES

"Composite" fuel is also a potential candidate for the FLO mission. Although it received only limited nuclear testing in the Nuclear Furnace (NF-1)[11], it also underwent extensive electrical furnace testing[18] (~10 hours at 2750 K with 64 temperature cycles) which demonstrated the potential to provide hydrogen exhaust temperatures and equivalent Isp values of ~2700 K and 900 s, respectively. Table 4 shows the IMLEO sensitivity to fuel form and different engine configurations for the FLO mission. Because of the higher density and mass of the composite-fueled system, and the "limited use" mission application ("1 Burn" TLI maneuver), the IMLEO savings resulting from the use of the composite fuel is only ~2 t. For more demanding "multi-burn" lunar missions, and the following Mars missions, the use of composite fuel shows definite performance advantages.

In addition to fuel form, a variety of alternative lunar mission profiles for both cargo and piloted flights have been examined[4] (see Figure 21). The fully reusable, piloted NTR mission scenario, shown earlier in Figure 3, utilized the lunar orbit rendezvous (LOR) mission mode. This all-propulsive NTR flight profile required four major impulsive burns (TLI, LOC, TEI and EOC), and cargo was returned to LEO (at 407 km) in the form of the "dry" LEV. For the FLO mission, a "1 Burn" TLI scenario is used with

Sensitivity to NTR Fuel Form

Engine Configuration	"Graphite"† (870 sec)	"Composite"†† (900 sec)
3 X 25 klbf*	202.4 t	199.6 t
2 x 50 klbf*	199.3 t	197.7 t
1 x 75 klbf*	192.9 t	190.4 t
1 x 100 klbf**	195.2 t	193.2 t
1 x 125 klbf**	199.7 t	198.0 t

† UC_2 Particles in Graphite, T_c=2550 K, expansion ratio 200:1
†† UC-ZrC-Graphite "Composite Fuel", T_c=2700 K, expansion ratio 200:1
* ZrH moderator augmentation, chamber pressure = 785 psia
** No ZrH moderator augmentation, chamber pressure = 1000 psia

Table 4. IMLEO Sensitivity to NTR Fuel Form

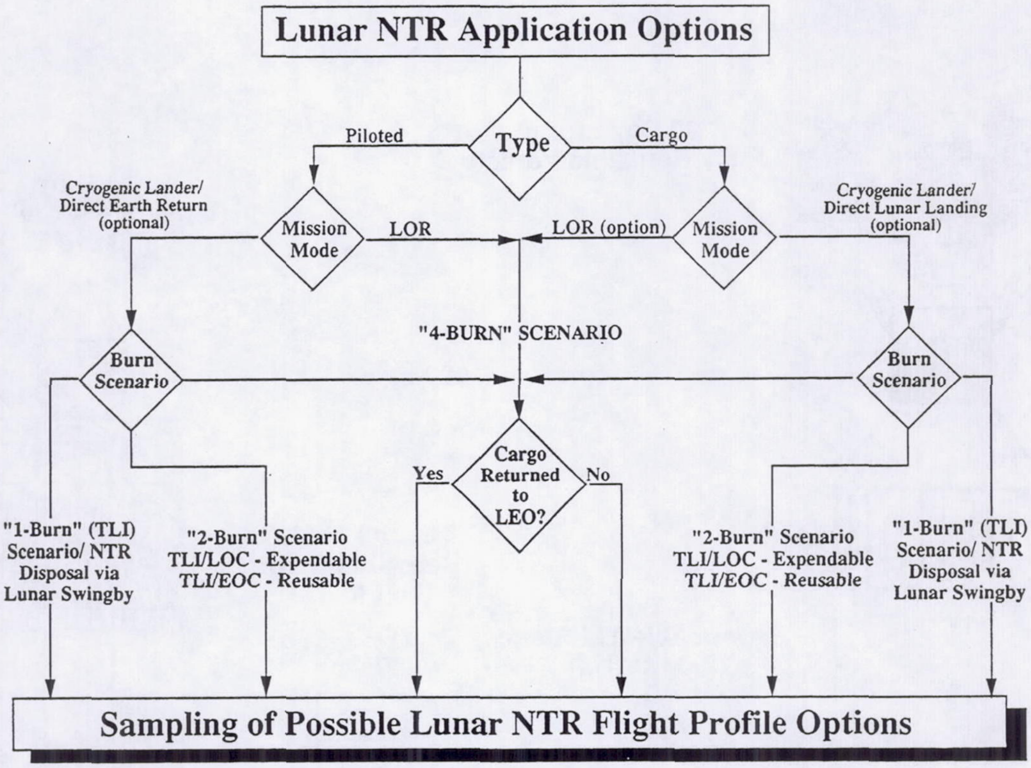


Fig. 21. Lunar NTR Mission Profile Options

NTR disposal being provided by a lunar gravity assist maneuver.

The characteristics of reusable lunar stages were also examined assuming the use of composite fuel and an "all cryogenic" piloted FLO lander (weighing ~76 t)[19]. By extending the size and LH_2 capacity of an expendable two engine FLO NTR stage from ~14.5 m and ~66 t, to ~20 m and 96 t, respectively, a single launch, reusable "2 Burn" mission scenario is possible. Following the TLI burn, this "stretched" NTR stage would target for a "leading edge" encounter with the Moon to set up a "free return" trajectory to Earth. Nearing Earth, the stage would perform a second Earth orbit capture (EOC) burn at high altitude and use its "cooldown thrust" to achieve a desired final parking orbit. The IMLEO required for the reusable "2 Burn" TLI/EOC configuration is ~202.5 t.

With a 150 t HLLV capability, a dual launch, Earth orbit rendezvous and dock scenario can be utilized to assemble a "two tank" configuration. This approach is capable of delivering into lunar polar orbit (LPO) an "all cryo" piloted lander/return stage weighing ~60 t. The lunar orbit insertion scenario[4] assumes capture into a 15 hour elliptical lunar orbit followed by a 70 degree plane change and subsequent circularization maneuver into a 60 nautical mile (~110 km) LPO. The scenario is reversed for trans-Earth injection. The first launch would carry the "core stage" consisting of a 10 m diameter by 20 m long propellant tank containing ~96 t of LH_2, and two 50 klbf composite fuel NTRs. The second launch would carry the piloted lander and a 10 m diameter by 14.5 m long supplemental "in-line" propellant tank containing ~66 t of LH_2. After lander separation and decent, the "in-line" tank would be jettisoned and the "core stage" returned to Earth orbit for reuse. The total IMLEO would be less than 275 t. Although the reuse options mentioned above have the potential to reduce "life cycle" costs, their use also necessitates the development of additional support infrastructure such as a "propellant tanker" or "fuel depot." Figure 22 compares the relative size of composite-fueled lunar NTR vehicles examined both in the 90 Day Study and in the present Fast Track Study. The first and fourth vehicles utilize a multi-launch,

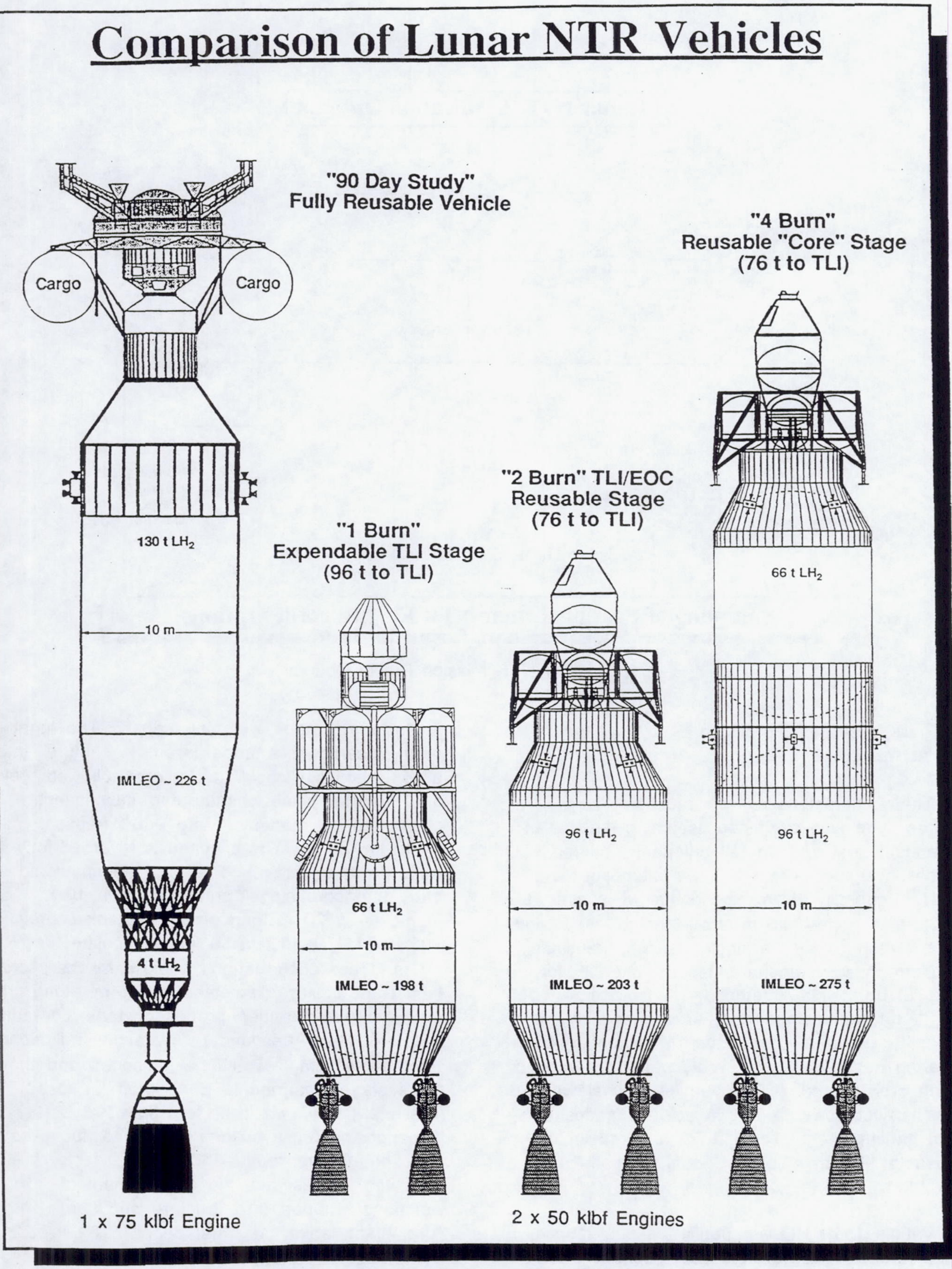


Fig. 22. Relative Size of Lunar NTR Vehicles

EOR&D scenario and 150 t-class HLLVs, while the second and third vehicles are deployed with a single launch, 250 t-class HLLV.

SUMMARY AND CONCLUSION

The results of integrated systems and mission studies are presented which quantify the rationale and benefits of using NTR propulsion for returning humans to the Moon in the early 2000's. In addition to performance benefits, the use of NTR propulsion on lunar missions can provide valuable operational experience and the technology can be "checked out" in a nearby space environment before it is used on the more demanding piloted mission to Mars.

For NASA's FLO mission, an expendable NTR stage powered by two 50 klbf engines is capable of delivering the 93 t FLO lander with its 3 t adaptor to TLI conditions for an IMLEO of ~199 t compared to 250 t for a LOX/LH_2 chemical stage. By extending the stage LH_2 tank length (from ~14.8 m to 20 m) and capacity (from ~68 t to 96 t), a single launch, reusable "2 Burn" TLI/EOC lunar stage is possible. With a 150 t-class HLLV, a dual launch, EOR&D scenario can be used to configure a two tank vehicle capable of accessing LPO and returning the "core" NTR stage to LEO for refueling and reuse. With its factor of two advantage in Isp over chemical propulsion and its high engine thrust-to-weight ratio, the NTR can form the basis for an efficient lunar space transportation system that can be appropriately modified to also satisfy subsequent Mars transportation system needs.

ACKNOWLEDGEMENTS

The authors wish to express their thanks to a number of individuals for insightful discussions and key contributions to the various topical areas addressed in this study. They include Kent Joosten, Dave Weaver and Ron Kahl (ExPO) on FLO mission and vehicle design, Lyman Petrosky (Westinghouse AES) on reactor design, Stan Gunn and Chuck Dunn (Rocketdyne) on engine design and scaling, Kyle Shepard (General Dynamics) on stage design, and Pat Rawlings (Science Applications International Corporation) for artwork depicted on the cover and in Figure 19.

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REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	October 1994	Technical Memorandum

4. TITLE AND SUBTITLE

"Fast Track" NTR Systems Assessment for NASA's First Lunar Outpost Scenario

5. FUNDING NUMBERS

WU–232–01–06

6. AUTHOR(S)

Stanley K. Borowski and Stephen W. Alexander

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135–3191

8. PERFORMING ORGANIZATION REPORT NUMBER

E–9164

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Washington, D.C. 20546–0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

NASA TM–106748
AIAA–92–3812

11. SUPPLEMENTARY NOTES

Prepared for the 28th Joint Propulsion Conference and Exhibit cosponsored by AIAA, SAE, ASME, and ASEE, Nashville, Tennessee, July 6–8, 1994. Responsible person, Stanley K. Borowski, organization code 6850, (216) 433–7091.

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Unclassified - Unlimited
Subject Categories 16 and 20

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

Integrated systems and mission study results are presented which quantify the rationale and benefits for developing and using nuclear thermal rocket (NTR) technology for returning humans to the Moon in the early 2000's. At present, the Exploration Program Office (ExPO) is considering chemical propulsion for its "First Lunar Outpost" (FLO) mission, and NTR propulsion for the more demanding Mars missions to follow. The use of an NTR-based lunar transfer stage, capable of evolving to Mars mission applications, could result in an accelerated schedule, reduced cost approach to Moon/Mars exploration. Lunar mission applications would also provide valuable operational experience and serve as a "proving ground" for NTR engine and stage technologies. In terms of performance benefits, studies indicate that an expendable NTR stage powered by two 50 klbf engines can deliver ~96 metric tons (t) to trans-lunar injection (TLI) conditions for an initial mass in low Earth orbit (IMLEO) of ~199 t compared to 250 t for a cryogenic chemical TLI stage. The NTR stage liquid hydrogen (LH_2) tank has a 10 m diameter, 14.8 m length, and 68 t LH_2 capacity. The NTR utilizes a "graphite" fuel form, consisting of coated UC_2 particles in a graphite substrate, and has a specific impulse (Isp) capability of ~870 s, and an engine thrust-to-weight ratio of ~4.8. The NTR stage and its piloted FLO lander has a total length of ~38 m and can be launched by a single Saturn V-derived heavy lift launch vehicle (HLLV) in the 200 to 250 t-class range. The paper summarizes NASA's First Lunar Outpost scenario, describes characteristics for representative engine/stage configurations, and examines the impact on engine selection and vehicle design resulting from a consideration of alternative NTR fuel forms and lunar mission profiles.

14. SUBJECT TERMS

Nuclear thermal rocket; NTR; First lunar outpost; FLO; Rover; NERVA; Modular; Space transportation

15. NUMBER OF PAGES

28

16. PRICE CODE

A03

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