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Space-to-Space Transport Systems

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Abstract

Following the start of NASA's Artemis and International Laser Ranging Service (ILRS)[1] programs, and the subsequent development of human activities in the cislunar space region, a variety of new challenges will arise. A relevant case is that of new transport systems which will be necessary to move persons and goods from Earth orbital stations to lunar orbital stations, Lagrange Point stations and other human outposts. Furthermore, advanced transport systems will also be needed to reach Near Earth Asteroids (NEAs) for mining purposes, and to move the mined ores and materials to locations where they can be processed and used for industrial purposes. Such vehicles will have requirements totally different from the rocket-machines designed to climb the Earth's gravitational well and to re-enter the Earth's atmosphere, since they never need to take-off from and return to the Earth´s surface. Just to mention two essential requirements – the robust construction to resist the launch vibrations and the insulating coating to survive the heat of the re-entry – will not be necessary for space-to-space spacecrafts. Some other requirements will take relevance, such as protection from solar and cosmic radiations, onboard simulated gravity, ergonomics and comfort. Different materials – not necessarily metals – might be used, and different propulsion systems may also be investigated. This paper describes two rotating space stations that provide a comfortable 0.8g artificial gravity environment to their occupants, with propulsion systems capable of transferring them to various cislunar or inner solar system destinations, including the Moon and Mars: a 24-person Space-to-Space Vehicle (STSV) and a 48-person Multipurpose Space Cruiser (MPSC). The production of such STSV or MPSC vehicles will be very much more convenient and profitable if assembled in space. Our trade study shows that Nuclear Thermal Propulsion (NTP) powered by liquid hydrogen propellant is the best choice, with In-Situ Resource Utilization (ISRU) for propellant production for return trips from destinations back to Earth. The SpaceX Starship and its future embodiments will facilitate the launch of STSV and MPSC components into space, with forecasted 200 metric ton payload capability and declining costs to Low Earth Orbit (LEO) from the current $1,500 - $3,000 per kilogram to as low as $150 per kilogram as tonnage increases and technologies improve.

Introduction

If we define the exploration and utilization of cislunar space and the solar system in general as a sustainable goal, we have to create a suitable environment for humans in outer space. Over past decades hundreds of astronauts have lived aboard space stations like the Russian MIR or the current ISS. Long stays in microgravity have caused profound and sometimes lasting physiological changes. The lack of gravity can lead to bone demineralization, muscle atrophy and cardiovascular deconditioning (Bukley et al 2007). To avoid these harsh effects of microgravity a key challenge will be to provide simulated gravity by rotation in space stations and space transport vehicles. Safety and technical redundancy are crucial conditions in the design of future spaceships and space stations. In Low Earth Orbit (LEO) space stations and manned vehicles are protected by the Van Allen Belt, the magnetosphere of the Earth, that is mitigating the influx of cosmic rays and solar storms. Beyond the Van Allen Belt there is a clear need for advanced radiation shielding. If we suppose the utilization of natural lunar and NEA resources and a future space industry on the Moon and in the Lagrange points we need advanced vehicles for travel of civilians in cislunar space.

1The International Laser Ranging Service (ILRS), often hosted and heavily supported by NASA, provides global satellite and lunar laser ranging data to support science, engineering, and geodetic research. Using ground stations to fire lasers at satellites, it aids in monitoring Earth's shape, rotation, and orbiting objects with millimeter precision.

“Space-to-Space” travel and transport in cislunar space

By using chemical rocket propulsion like the Apollo mission spacecrafts, it takes about three days to travel from Earth orbit to lunar orbit. The same flying distance of about 384,000 kilometers we have between the Moon and the Lagrange points L4 and L5 (**Figure 1**). For such three days flights of humans and cargo there will be the need to develop Space-to-Space Vehicles (STSV).

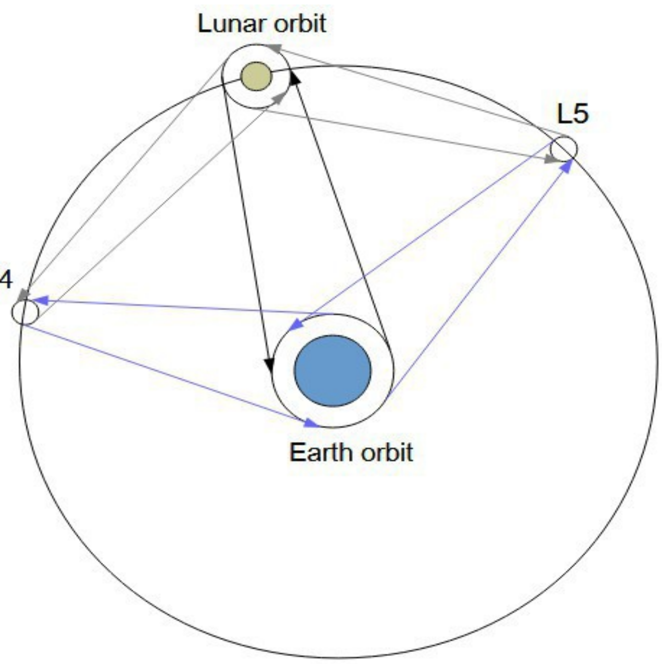
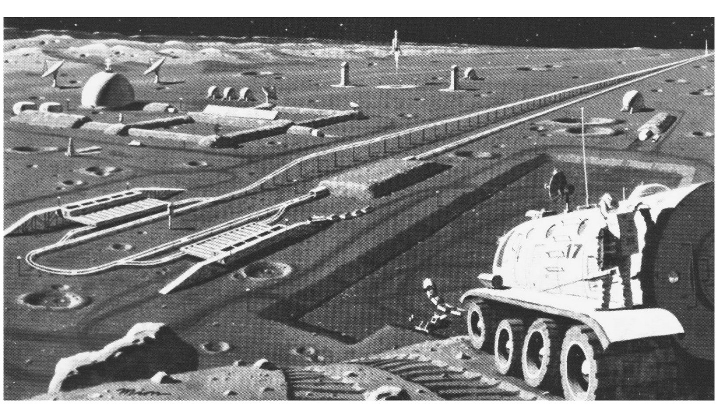


Fig 1 Future flight tracks of Space-to-Space Vehicles (STSV) in cislunar space

Transport of material and cargo in cislunar space

Our Moon has just one sixth of Earth´s gravity. In the 1970s Gerard K. O´Neill has proposed to build an electromagnetic mass driver on the Moon to “shoot” cargo boxes with lunar material into lunar orbit (O´Neill 1976). **Figure 2** shows the lunar mass driver which accelerates the cargo containers up to the lunar escape velocity of 2.38 km/sec.



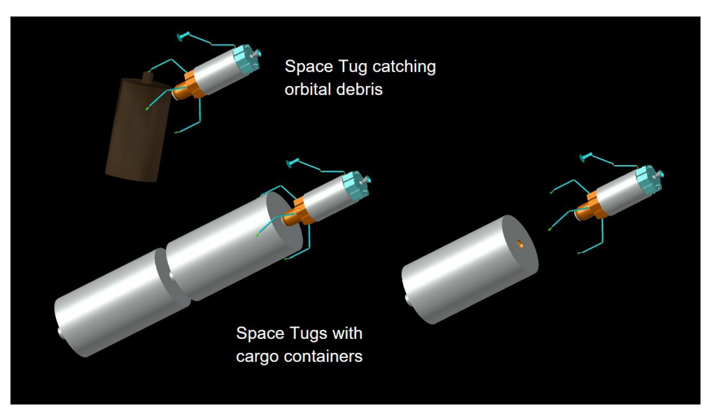


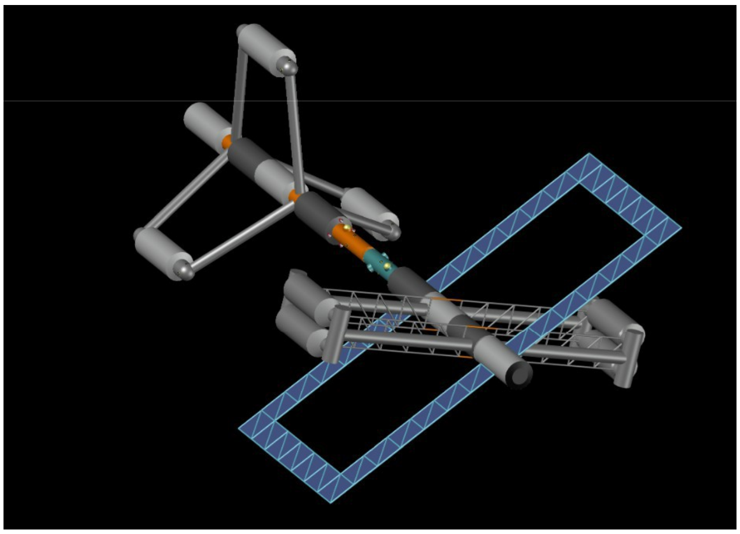
Fig. 2 electromagnetic lunar mass driver (NASA)

Fig. 3 robotic space tugs for cargo shipping (© W. Grandl)

After arrival of the cargo containers in lunar orbit, they could be “caught” by unmanned robotic space tugs and shipped to Earth orbit or the Lagrange points of the Earth-Moon system (**Figure 3**). In Earth orbit the tugs are also able to catch orbital debris of certain size and bring this waste material to space factories in L4 or L5 for recycling.

The Space Cruiser – a preliminary design of a civilian STSV

For the travel of civilians like scientists, engineers and space tourists in cislunar space we will have to provide comfort and service similar to a terrestrial hotel. That means to create simulated gravity, private rooms for each person, including toilets and shower cubicles, and a central kitchen and dining room. A necessary condition are space stations in Earth orbit and in lunar orbit, where a Space Cruiser can dock (**Figure 4**).



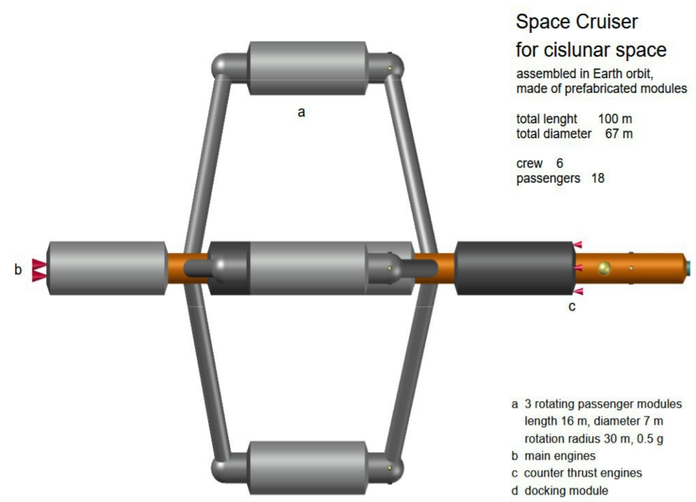
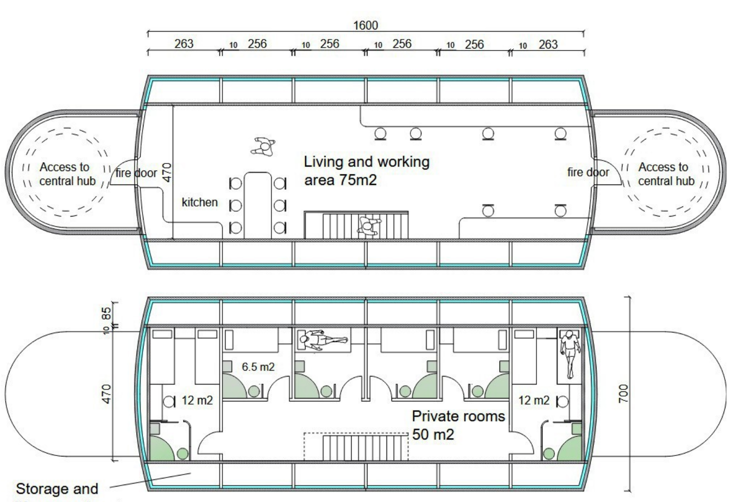


Fig.4 Space Cruiser, docked to an orbital station (© W. Grandl) **Fig. 5 Space Cruiser, side view** (© W. Grandl)

The Space Cruiser has 3 rotating passenger modules providing 0.5 g simulated gravity (**Figure 5**). It could be made of prefabricated rigid modules and assembled in Earth orbit by astronauts and robots. The vehicle has a total length of 100 m and 67 m diameter. For cislunar missions, it weighs 350 t (290 t for the STSV plus 60 t for water and nutrition. The main engines and the control thrusters are powered by liquid hydrogen and liquid oxygen. The counter- and control thrusters can turn around the spacecraft for deceleration when it approaches its destination. 18 civilian passengers and a crew of six can travel for six or seven days e. g. to lunar orbit and back to Earth. Each passenger module has two floors, the 75 m^2 upper floor for living, dining and working, and the 50 m^2 lower floor with private rooms for eight persons -six passengers and two crew members (**Figures 6 and 7**).



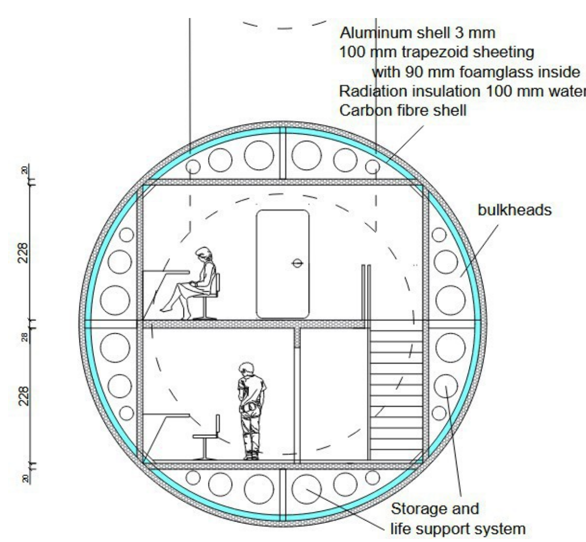


Fig. 6 passenger module floor plans (© W. Grandl) **Fig. 7 passenger module section**

Future spacecraft need maximum safety and technical redundancy to protect civilian passengers from cosmic rays, solar flares and micrometeorite impacts. For the outer hull of the modules we propose a light-weight aluminum construction with multi-layer radiation and thermal shielding, including water as an effective radiation protection (**Figure 7**). Aluminum trapezoid sheeting for the fuselage-framework and -lining provides maximum stiffness and minimum launch weight (**Figure 8**). For launching from Earth and transport of the modules and components into Low Earth Orbit (LEO) reusable SpaceX rockets or similar launchers can be used.

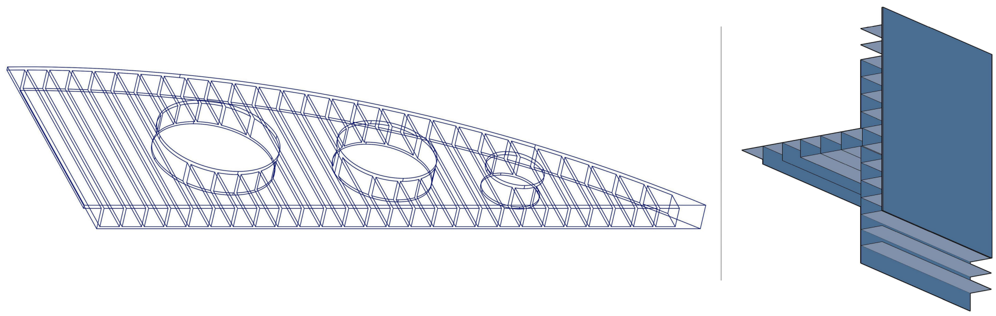
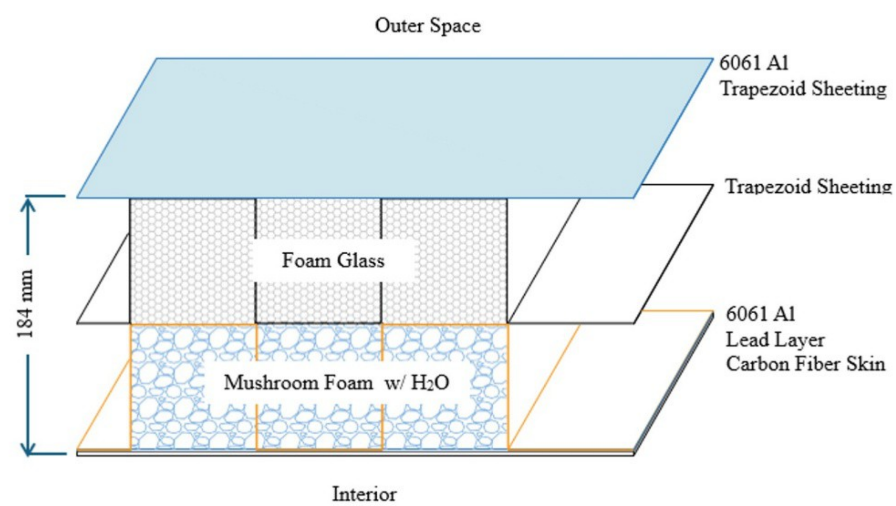


Fig. 8 Work pieces made of thin aluminum sheets and trapezoid sheeting (schematic drawings)

The spacing between the trapezoid sheets can be filled with foam glass for thermal insulation and -in an additional layer- with a "mushroom" structure to be filled with water as a favorite radiation shielding (**Figure 9**). The interior lining and furnishing are envisioned to be made of carbon fiber panels and 3D printed elements to keep the total mass limited. Each passenger module should have its own independent life support system for optimum safety and redundancy. Like in rotating space stations, the joints between rotating and non-rotating parts of the vehicle are designed as "magnetic liquid rotary seals" (Grandl 2017).



Dimensions

Outer Shell System Weight 50 kg/m^2

3 mm	aluminum skin
5 mm	trapezoid sheeting
90 mm	foam glass
5 mm	trapezoid sheeting
90 mm	"mushroom" foam with H_2O
3 mm	aluminum layer
0.5 mm	lead layer
2 mm	carbon fiber skin

Specific Weight Dry	approx.	20 kg/m^2
Specific Weight including H_2O	approx.	50 kg/m^2

Fig. 9. Outer shell system of manned module (© W. Grandl)

Reaching out for the Solar System- a Multipurpose Space Cruiser (MPSC)

To go beyond the Earth-Moon system and travel to Mars and perhaps other planets and moons of the solar system we can enlarge and improve the above Space Cruiser design. A Multipurpose Space Cruiser (MPSC) can be envisioned to be used both as a manned spacecraft and a movable space station in Earth orbit, lunar orbit or even in the orbit of Mars and other celestial bodies. We would have to add a nuclear power source and landing vehicles for descent to the surface of celestial bodies and safe return to the mothership. **Figure 10** shows a design study for an advanced Space Cruiser with two landing vehicles attached to the docking module.

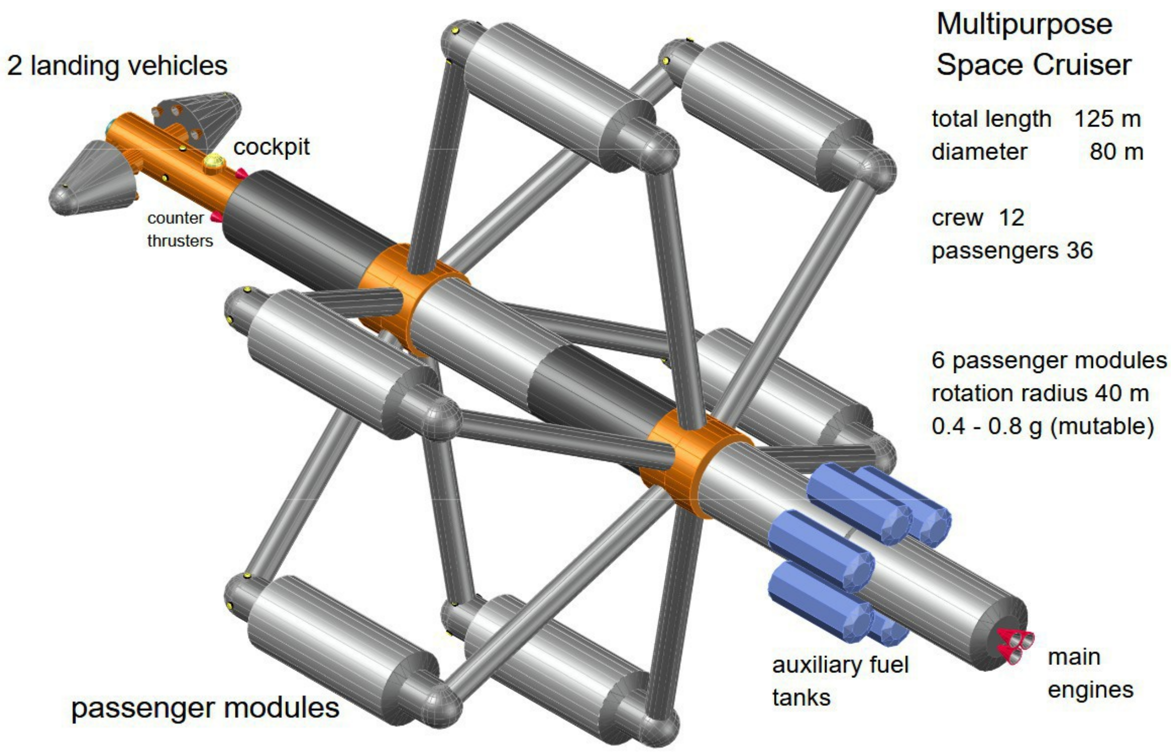


Fig. 10 Preliminary design of the MPSC; for the Space Cruiser version to Mars and beyond the main engines and the (auxiliary) fuel tanks (blue) have to be scaled and enlarged according to the calculations in the Appendix (© W. Grandl 2025)

The enlarged spacecraft design can carry 36 passengers and 12 crew members. Six rotating passenger modules can provide a mutable gravity simulation of 0.4 to 0.8 g. Each landing vehicle can land three persons on the surface of Moon, Mars or other celestial bodies smaller than Mars. When a first prototype is tested it could travel to lunar orbit, stay there for some weeks, explore different lunar sites and finally return to Earth orbit. By changing rotation rates and rim speed the effects of different g-levels on the human body could also be studied. When going to Mars the Space Cruiser could use gravity assist for acceleration by passing the Moon. The physical principles of gravity assist (or gravitational slingshot) maneuvers are simple: the flyby transfers a small amount of the Moon´s momentum to the spacecraft which in turn is accelerated, while the Moon is decelerated (Johnson 2003).

Space-To-Space Vehicle and Multipurpose Space Cruiser Weights

Table 1 shows the weight breakouts for both the STSV and the MPSC as envisioned by the authors.

STSV 18 passengers, 6 crew members			**MPSC** 36 passengers, 12 crew members		
5 zero-gravity modules	20 t	100 t	6 zero-gravity modules	20 t	120 t
2 roto-joints	10 t	20 t	2 roto-joints, fuel tanks (empty)	20 t	40 t
3 passenger modules	24 t	72 t	6 passenger modules	24 t	144 t
180 m spokes	0.5 t/m	90 t	384 m spokes	0.5 t/m	192 t
8 robots for assembly	1 t	8 t	2 more robots	1 t	2 t
		290 t			**498 t**
Eqmt, Supplies, Misc.		**50 t**	Eqmt, Supplies, Misc.		200 t
	Total	**340t**		**Total**	**698 t**

Now we must calculate the consumables: air, water, and food. Astronauts on the International Space Station consume about 2.5 kg of food and one gallon (3.78 kg) of water daily. Rather than take showers, they use wet wipes. Human waste on the ISS is handled using specialized fan-driven vacuum toilets. Liquid waste is collected, filtered, and purified into drinkable water, while solid waste is collected in airtight containers, loaded into uncrewed cargo ships, and incinerated upon re-entry into Earth's atmosphere. The Environmental Control and Life Support System (ECLSS) on the ISS manages to recycle 98% of the water used by the astronauts. As for air to breathe, air on the ISS weighs 1200 kg – 1350 kg. We'll conservatively assume 2.75 kg, 4 kg, and 1350 kg for food, water, and air respectively. Since 98% of the water is recycled, each astronauts consumes $4 \times 0.02 = 0.08$ kg water per day. We assume that 1350 kg of air on the ISS scales with the number of astronauts, and the ISS normally holds six astronauts. **Table 2** tabulates the consumable weights for the STSV and the MPSC.

Table 2. Weight of consumables for STSV and MPSC

STSV 24 occupants, 4 times ISS		**MPSC** 48 occupants, 8 times ISS
Air	5.4 t	10.8 t
Food	0.011 t times mission days	0.022 t times mission days
Water	0.00192 t times mission days	0.00384 t times mission days

There are many additional items to consider: space suits, oxygen tanks, crew hygiene items, scientific hardware and experiments, spare parts, maintenance tools, extra propellant to regularly boost the station's altitude and keep it from falling into the Earth's atmosphere, gases to replenish atmospheric pressure losses, exercise equipment, care packages from family, musical instruments, books, games, computers, and small mementos to help astronauts manage stress during their multi-month missions. Rather than tabulate each of these separately, we conservatively assume 50 tonnes for the STSV and 200 tonnes for the MPSC. If consumable weights are much less than these numbers, we will just lump everything together.

STSV Total Weight

290 t + ≈ 50 t (eqmt , supplies, misc) + ≈ t consumables + ≈ t propulsion system (depending on mission)

MPSC Total Weight

498 t + ≈ 200 t (eqmt, supplies, misc) + ≈ t consumables + ≈ t propulsion system (depending on mission)

Propulsion systems for travel in cislunar space

We will analyze both chemical and nuclear thermal propulsion for transport and travel in cislunar space. can be done with chemical rocket propulsion. For this purpose, this paper will describe a Raptor-engine powered SpaceX propulsion system. It uses sub-cooled liquid methane and fuel and liquid oxygen as oxidizer, together known as "methalox". Both the Super Heavy booster and the Starship upper stage are powered by full-flow staged combustion Raptor engines, which require this propellant combination to function. SpaceX is evolving the Raptor engine toward unprecedented thrust, extreme simplicity, and mass production to enable routine, multiplanetary travel. Through iterative engineering, the design has shifted from a complex web of parts (Raptor 1) to a clean, integrated engine (Raptor 3) that eliminates the need for external heat shields. As of 2026 SpaceX is developing the Raptor 4 with 300 to 330 tonnes force (~ 2,942 to 3,236 kN) per engine (**Figure 11**). The payload capacity for the Starship Block 4 is projected as 200 t (Berger 2024).

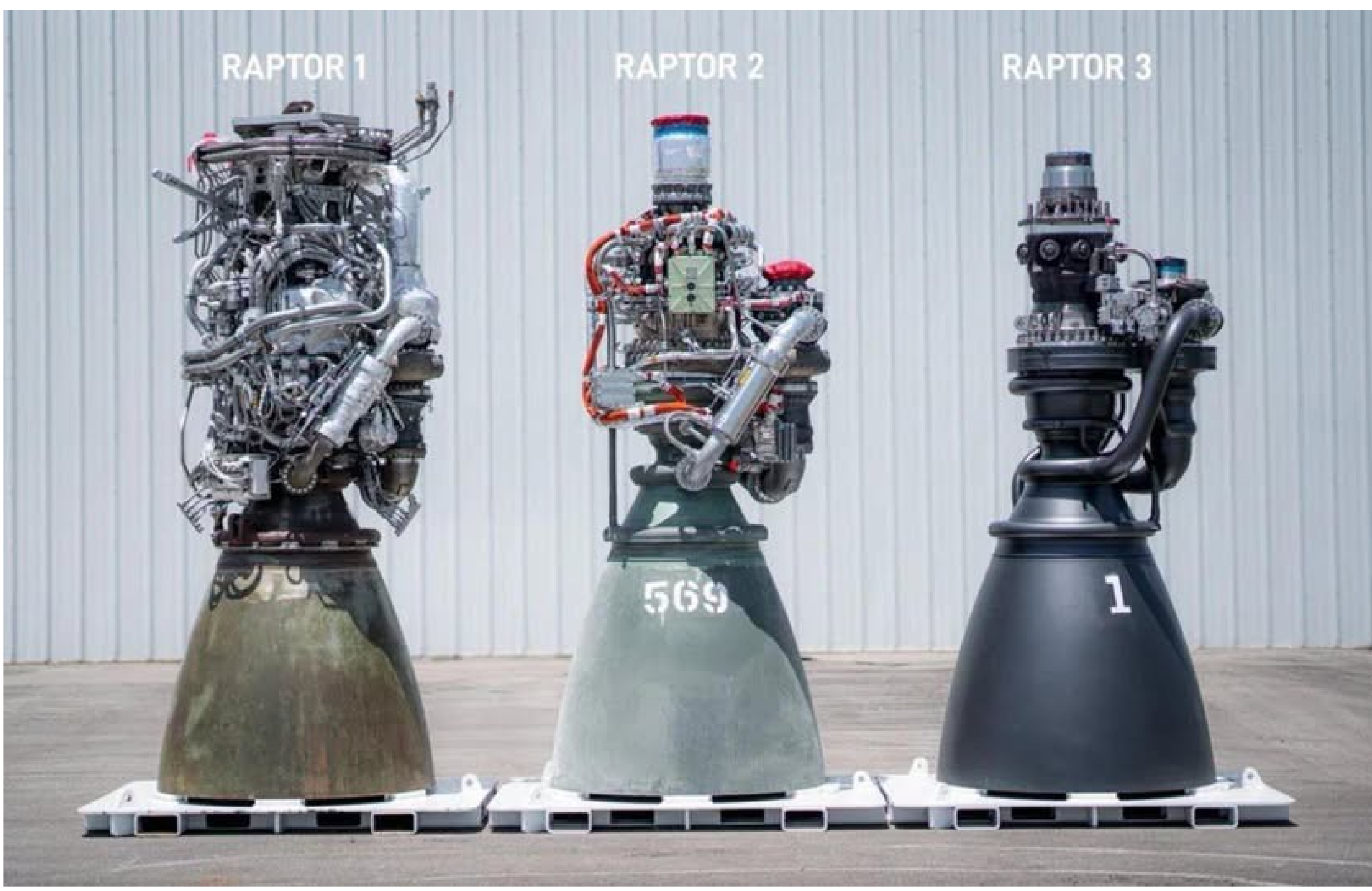


Fig.11 SpaceX Raptor Engine Evolution

Starship

SpaceX's Starship is a two-stage, fully reusable super heavy-lift launch vehicle consisting of two stages: the Super Heavy booster and the Starship spacecraft. Both stages are intended to return to the launch site and land vertically at the launch tower for potential reuse. SpaceX states that its goal is to reduce launch costs by both reusing and mass producing both stages. Once in space, the Starship upper stage is intended to function as a standalone spacecraft capable of carrying crew and cargo. At the end of its mission, Starship reenters the atmosphere using heat shield tiles similar to those of the Space Shuttle. However, Starship’s heat shield tiles differ from the Space Shuttle’s in several key ways. While both use high-purity, silica-based ceramics, Starship's tiles are standardized, hexagonal, and mechanically attached. In contrast, Shuttle tiles were square, custom-fitted, and glued. This makes Starship's system more durable, faster to repair, and cheaper to produce.

The Block 4 Super heavy-lift launch vehicle stands 142 m (466 ft) tall and will be able to carry 200 t (440,000 lb) to Low Earth Orbit (LEO) according to SpaceX’s S-1 filing with the Securities and Exchange Commission. SpaceX currently achieves launch costs to LEO ranging between \$1,500 and \$3,000 per kilogram. It is hard to predict exact numbers, but with continuing improvements to Starship, future launch costs could decline to the neighborhood of \$150 per kg. **Figure 12** shows the shorter Block 1 version of SpaceX’s Super-heavy-lift launch vehicle. We assume that the Block 4 version will be used to launch the elements of the STSV for cislunar missions and eventually the MPSC for missions to Mars and other destinations.

Fig. 12 Block 1 version of SpaceX super heavy launch vehicle on launch pad in Starbase, Texas

STSV assembly in LEO and propulsion for cislunar missions.

An in-space infrastructure must be in place to assemble the STSV. This includes robots, a space station, and refueling stations. The STSV can serve as a “flying hotel” for tourism. At least two Starship launches will be required to place the modules of the 350 t STSV into LEO and prepare it for cislunar missions. Moving large payloads from LEO to Low Lunar Orbit (LLO) requires the fully reusable SpaceX Starship acting in tandem with its Starship Tanker variant. Because taking a massive payload to the Moon requires more propellant than a single Starship can carry while launching from Earth, the architecture relies on LEO rendezvous and orbital refueling. Multiple dedicated Starship Tanker flights will launch from Earth to stockpile liquid methane and liquid oxygen in a LEO "fuel depot." A separate Cargo Starship will launch the STSV modules into LEO. The Cargo Starship will dock with the LEO fuel depot to completely top off its tanks. Once fully fueled in LEO, the Starship will fire its six Raptor 4 Vacuum engines to escape Earth's gravity and place the STSV on a trajectory to the Moon. In the first phase, the entire STSV and its propulsion system must come from Earth. Later on, in-space resources could be used.

Propulsion systems for travel in the inner solar system

SpaceX plans to travel to Mars using the fully reusable Starship system, with the ultimate goal of establishing a self-sustaining city on the planet. The initial phase involves launching several uncrewed Starships to test safe landing capabilities and deliver cargo. These flights rely on orbital docking and "tanker" ships to refuel the main spacecraft in Earth orbit before it begins the months-long journey to Mars. SpaceX plans to send uncrewed spacecraft ahead of human crews. These missions carry supplies and autonomous equipment (such as Tesla's Optimus robots) to start surveying resources, mining water ice, and building infrastructure. Pending the success of uncrewed landings, the first human crews—including private astronauts and explorers—will make the journey to begin constructing habitats and establishing a permanent surface presence. The long-term vision is to steadily increase the frequency of flights during every launch window (e.g., launching fleets of hundreds of Starships) to transport millions of tons of cargo and eventually thousands of settlers to build an entire Martian civilization.

Transporting the large MPSC beyond cislunar space, such as to Mars, the asteroid belt, or the moons of Jupiter, presents a huge challenge even for advanced propulsion systems realizable in the post-2050 timeframe. The logistics involved will also present a daunting challenge. S.D. Fraser (Fraser 2013) gives an overview of power system technologies which could be used for space mining and transport. **Table 3** shows the advantage of nuclear fission and nuclear fusion technology compared to chemical H_2/O_2 rocket propulsion. Although nuclear fusion is still not available today it will probably become available by the end of the century. The isotope helium^3 is rare on Earth but abundant on the Moon in the upper layers of regolith (Wittenberg et al 1992).

Table 3 Theoretical fuel energy density of fusion and fission fuels versus chemical fuel (Fraser 2013)

Fuel	**Energy density**
Fission	8.2 x 10^13 J/kg
Deuterium-deuterium fusion	8.8 x 10^13 J/kg
Deuterium-tritium fusion	3.4 x 10^14 J/kg
Deuterium-helium3 fusion	3.5 x 10^14 J/kg
Hydrogen-oxygen H_2/O_2	1.6 x 10^7 J/kg

Due to its huge energy density, deuterium-helium3 fusion has inspired scientists and engineers for decades. An example is the Bussard Fusion System, also known as the quiet-electric-discharge (QED) engine (Bussard 1997, 2002). This system is envisioned to use electrostatic fusion devices to generate electrical power: deuterium and helium3 are fusing to helium4 plus protons. The charged protons escape from confinement, and their kinetic energy can be converted to electricity or be used directly as a plasma beam for generating thrust.

Although the benefits of He3 for fusion propulsion are often touted by space enthusiasts, they leave out that fusion propulsion has not been proven to work, and its practicality for future propulsion is uncertain. Consequently, for missions to the inner solar system, Mars, the asteroid belt, or the moons of Jupiter we propose using a Nuclear Thermal Propulsion (NTP) system for such missions.

Obstacles to Implementing a Bimodal Nuclear Thermal Electric Propulsion for Mars Missions

A BNTEP propulsion system (Kokan et. al., 2026) is not feasible for transporting the MPSC to Mars and back. We hypothesize that BNTEP propulsion on the massive scale required for such missions will not reach technical maturity until 2050 or later. It faces serious obstacles and comes in as a distant second to using Nuclear Thermal Propulsion (NTP) for such missions.

BNTEP propulsion relies on nuclear fission reactors, liquid hydrogen, and xenon propellant. It is difficult to manufacture xenon because it is one of the rarest stable elements on Earth. It makes up only about 0.087 parts per million by volume in the Earth's atmosphere, leading to a geological mystery known as the "missing Xe problem." This extreme dilution makes finding and harvesting raw sources highly inefficient. Because it cannot be synthesized, xenon must be harvested as a by-product of large-scale air separation. Extracting it requires continuously cooling vast quantities of atmospheric air to cryogenic temperatures and liquefying it to separate nitrogen and oxygen. Even after isolating the primary gases, the remaining liquid oxygen must undergo further, highly specialized fractional distillation just to be enriched with 0.1–0.2% of a krypton/xenon mixture. Finally, the krypton and xenon mixture must undergo yet another multi-stage distillation. To get high-purity xenon for advanced applications (like in lighting, satellites, or medical resonance imaging), the gas must be separated from remaining trace impurities, requiring immense amounts of energy and highly complex distillation columns.

Finally, because commercial xenon relies entirely on atmospheric separation, the global supply is at the mercy of heavy industries like steel-making and oxygen production. Any downturn in these industries immediately constricts the supply of xenon, causing severe price spikes.

Space Nuclear Thermal Propulsion for MPSC missions to Mars

In SNTP a fission reactor fed by High-Assay Low-Enriched Uranium (HALEU) enriched with 19.75% of the fissile isotope U-235, a relatively low-radiation material, generates intense heat. A liquid propellant – usually lightweight liquid hydrogen – is pumped through the reactor core. The propellant instantly expands into a high-pressure, super-hot gas and is blasted out of a rocket nozzle to create thrust.

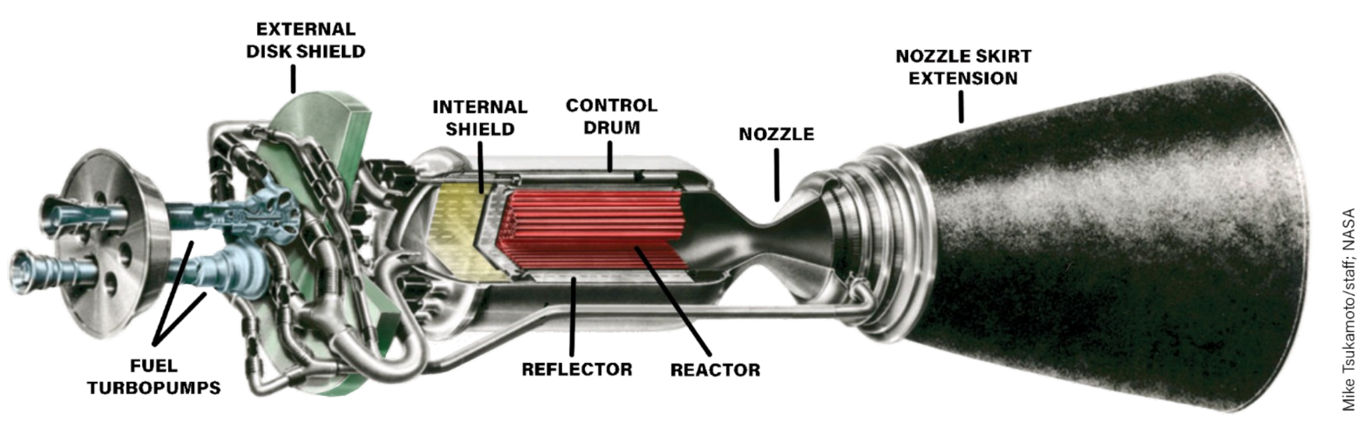


Fig. 11 SNTP system principle of operation

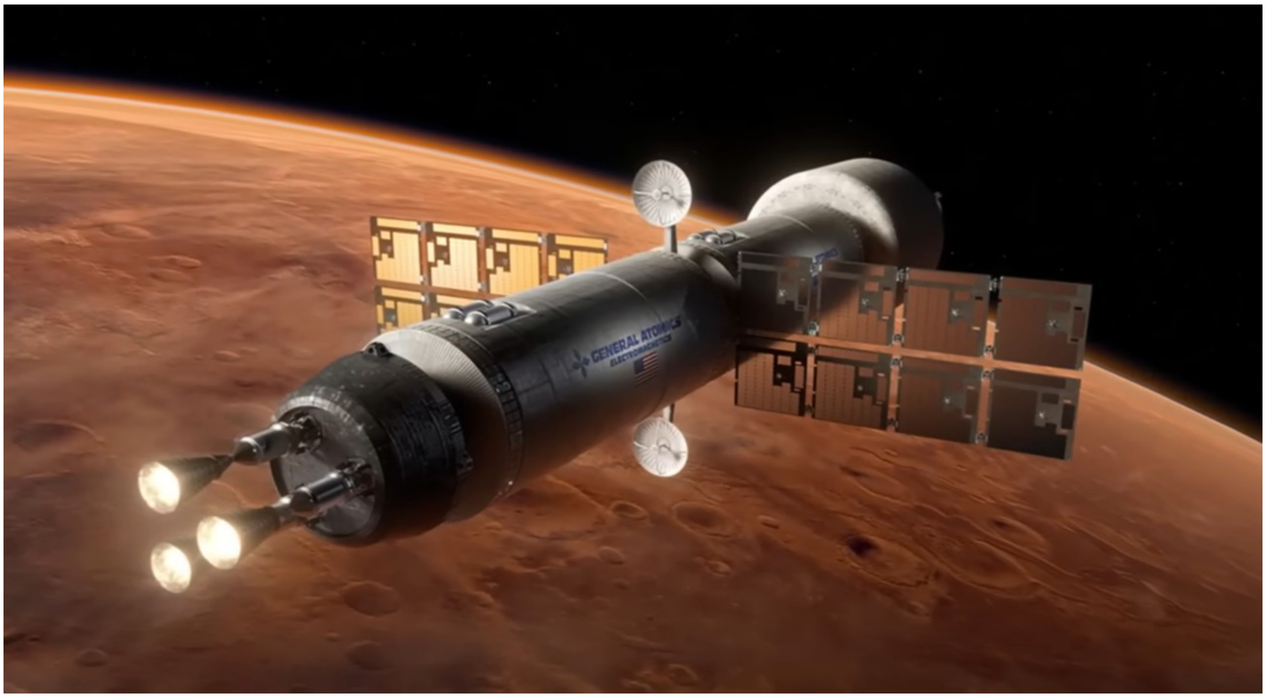


Fig. 12 Spacecraft powered by an SNTP system approaching Mars

HALEU currently faces a commercial supply bottleneck – there is not yet a guaranteed, steady, long-term customer base. This has created a "chicken-and-egg" problem, where enrichment facilities hesitate to invest in massive upfront infrastructure without assured demand. To bridge this gap, the U.S. Department of Energy (DOE) is spearheading the HALEU Availability Program to catalyze a domestic supply chain. The DOE partnered with Centrus Energy Corp to operate the first domestic commercial-scale HALEU enrichment cascade in Piketon, Ohio, and is actively issuing contracts to expedite testing and deployment of advanced reactor pilot programs.

MPSC Missions to Mars

The most energy-efficient launch periods for missions to Mars—known as Hohmann transfer windows—occur every 26 months (about 2 years and 2 months). While you can launch a spacecraft to Mars using a Hohmann transfer orbit every 26 months (the standard synodic period), the amount of energy (or) required changes drastically over time due to Mars' highly elliptical orbit. The lowest-energy launch windows occur during "perihelic oppositions," when Mars is closest to its perihelion (closest point to the Sun) and moving at its fastest. Ideal launch times occur when Mars is about 44º ahead of Earth, which is roughly 96 days before Mars Opposition.

Table 4. Best low-energy launch windows, 2040-2082

Rank	Approx. launch window center	Associated Mars opposition	Mars-Earth distance at opposition	Approx. Mars arrival
1	~May 28, 2082	Sept. 1, 2082	0.3738 AU	~Feb. 11, 2083
2	~ May 10, 2050	Aug. 14, 2050	0.3742 AU	~Jan. 24, 2051
3	~ June 28, 2067	Oct. 2, 2067	0.3997 AU	~Mar. 13, 2068
4	~April 8, 2065	July 13, 2065	0.4021 AU	~ Dec. 23, 2065
5	~July 24, 2052	Oct. 28, 2052	0.4464 AU	~Apr. 9, 2053
6	~March 12, 2080	June 16, 2080	0.4502 AU	~Nov. 26, 2080
7	~Feb. 28, 2048	June 3, 2048	0.4797 AU	~Nov. 13, 2048

AU: Astronomical Unit, 149,597,870 km

We assume a long-stay Conjunction Class mission with a surface stay of ~500 days. With a total trip A reasonable Mars surface stay time is roughly 500 days (16 to 20 months). This prolonged duration is required to wait for the planets' orbits to align favorably for a fuel-efficient return trip, making the entire journey last approximately two to three years.

We also assume a round trip mission to Mars is powered by NTP using liquid hydrogen as fuel. In Situ Resource Utilization (ISRU) can be used to extract water ice (H_2O) found abundantly beneath the planet's surface or frozen in the Martian polar regions. The process for mining Martian hydrogen involves a straightforward and established set of steps:

1. Extraction: Automated rovers and mining rigs will dig up ice-rich Martian soil (regolith) or extract solid ice from glaciers and subsurface permafrost.
2. Purification: The ice is melted and filtered. Any salts or "dirt" (like magnesium perchlorate) must be removed to prevent damaging the equipment.
3. Electrolysis: The purified water is pumped through an electrolyzer, an electrical device that splits the water molecules (H_2O) into separate oxygen and hydrogen gases.
4. Fuel Synthesis: The separated hydrogen gas can be used directly or combined with carbon dioxide from the Martian atmosphere to create methane rocket fuel (via the Sabatier reaction)

Minimizing liquid hydrogen (LH_2) losses during a Mars mission requires mitigating boil-off caused by extreme temperature differences and space radiation. Key strategies include advanced multi-layer insulation, active cryocoolers to intercept heat, and parahydrogen conversion to stabilize the molecules. Here are primary techniques to minimize losses:

1. Active Cryocooling: Because passive insulation cannot completely stop heat transfer over long durations, active mechanical cryocoolers (such as reverse-Brayton or Stirling coolers) act like deep-space refrigerators. They actively extract heat from the propellant tanks and radiate it into space NASA Cryo fluid management.
2. Advanced Multi-Layer Insulation: Tanks are wrapped in dozens of alternating layers of reflective material (like aluminized Mylar) and low-conductivity spacers to minimize radiant heat transfer.
3. Parahydrogen Conversion: Hydrogen naturally exists in two molecular states: ortho and para. Storing LH_2 as 100% parahydrogen is crucial because the ortho-to-para transition releases heat, which causes spontaneous boil-off. Catalysts are used during liquefaction to force this conversion prior to launch.
4. Vapor-Cooled Shields: Boil-off gas is channeled through tubing around the propellant tank before being vented. This creates a chilled thermal barrier that intercepts incoming heat.
5. In-Space Refueling (ISRU): Rather than carrying all the LH_2 from Earth, missions can produce liquid hydrogen on Mars via electrolysis of subsurface ice and the Sabatier process. This reduces the time LH_2 spends in transit.
6. Sunshields & Orientation: Spacecraft are designed to orient the hydrogen tanks in the mission's "shadow," utilizing sunshades and strategic trajectories to keep the tanks shaded from the sun.

Why Clustered 25k Thrust NTP Engines Win Out

It is not advantageous to cluster higher thrust NTP engines. Larger engines are studied primarily to reduce the complexity of plumbing (valves, turbopumps) or to minimize the total length of long burns (like Trans-Injection). However, the nuclear control complexities and regulatory hurdles of scaling a core too large generally outweigh the benefits. Clustering smaller engines is widely considered more practical and economical for NTP in-space transportation. NASA's Design Reference Architecture 5.0 explicitly baselines a clustered arrangement of three to six 25,000 lbf (111 kN) class engines rather than massive, single monolithic reactors.

1. Reliability and Redundancy: A 16-engine cluster as required for MPSC round trips to Mars could be broken down into smaller clusters of 4 to 6 25k lbf (111 kN), allowing for engine-out capability. If one reactor fails, the remaining engines can still execute the Mars transit.
2. Development and Testing Costs: Scaling nuclear reactors to very high thrusts (e.g., 450 kN to 800 kN) is incredibly expensive. Developing the smaller 111 kN engine allows for ground-level testing in existing, licensed facilities without building massive new test infrastructure.
3. Mass Production: Just like modern chemical rockets, serial manufacturing of a single "standard" reactor core benefits from an economy of scale and reduces the overall unit cost.
4. Spacecraft Integration: A cluster of smaller engines provides a lower profile, making it much easier to structurally mount, shield, and package inside standard heavy-lift payload fairings (such as the Space Launch System or Starship)

As shown in Appendix B.2, typical Mars missions will require ~ 259 days for travel each way, and ~ a 500+ day stay on Mars for a total of 1018+ days. Assume that consumables must conservatively last for 1200 days, allowing for delays. From Table 2 we calculate 10.8 t (air) + 26.4 t (food) + 2.7 t (water) for a total of 37.6 t, assuming that Mars ISRU is used to reload water for the return trip from Mars. We will add this to the 200 t (equipment/misc.) allowance, for an MPSC weight of 37.6 t + 498 t + 200 t = 735.6 t.

STSV and MPSC Missions to the Moon

For missions to the Moon, we will compute the Δv for travel to Earth-Moon Lagrange points L4 and L5; and to a lunar Near-Rectilinear Halo Orbit (NRHO). NRHO is a highly elongated, gravitationally balanced space trajectory where a spacecraft orbits a specific Lagrange point in the Earth-Moon system. It serves as the staging ground for NASA's Artemis lunar missions and the site for the international Gateway space station.

The orbit is "halo"" because the spacecraft effectively orbits an invisible gravitational balance point (specifically the L2 Lagrange point on the far side of the Moon). It is "near-rectilinear" because it is highly stretched out. At its closest approach (perilune), the spacecraft swoops within roughly 3,000 km of the lunar poles. At its farthest point (apolune), it swings out up to 70,000 km away. The orbit has a resonant 9:2 relationship with the Moon's orbit around Earth, meaning it takes about 6.5 to 7 days to complete a single loop.

Before settling on NRHO, NASA mission planners had to balance the benefits of standard low lunar orbits with more distant orbits. NRHO provides the best of both worlds:

- **Fuel Efficiency:** Because it operates on the edge of the Earth's and Moon's combined gravity wells, it requires very little propellant to maintain, making it ideal for the long-term (15+ year) lifespan of the [Gateway space station.
- **Continuous Communication:** Unlike low orbits which take spacecraft behind the Moon and cause temporary blackouts, NRHO's elongated geometry ensures that a direct line of sight to Earth is always maintained for mission control.
- **Eclipse Avoidance:** The orbit is tilted in such a way that the spacecraft rarely passes into Earth's shadow, allowing for constant solar power generation.
- **Lunar Surface Access:** The close approach to the lunar poles provides astronauts with easy, predictable access to the lunar surface.

Comparison (from 450 km LEO)

Destination	Departure Δv	Arrival/insertion Δv	Ideal total Δv	Practical planning Δv	Approx. transfer time
Earth–Moon L4	3.070 km/s	0.828 km/s	3.898 km/s	3.98–4.05 km/s	~4.98 days
Earth–Moon L5	3.070 km/s	0.828 km/s	3.898 km/s	3.98–4.05 km/s	~4.98 days
Lunar NRHO	3.118 km/s	0.400–0.450 km/s	3.518–3.568 km/s	3.58–3.63 km/s	~4–10 days
110 km circular LLO	3.070 km/s	0.819 km/s	3.889 km/s	3.95–4.05 km/s	~4.98 days

The direct NRHO transfer is approximately 0.32–0.38 km/s less costly than full insertion into either L4/L5 or a 110 km circular lunar orbit. *Because these Δv values are very similar, we will just define propulsion systems for travel from 450 km LEO to 110 km LLO and return*.

Deceleration when approaching the Moon could be simply done by turning around the Space Cruiser 180 degrees and firing the engines. In lunar orbit the Cruiser (as a multipurpose system) is envisioned to orbit the Moon and allow the visitors to descend to the surface by attached landing vehicles. For crewed missions or specialized landings, this transportation is often facilitated by the Starship Human Landing System (HLS) variant. The HLS variant is optimized for the lunar environment and descends from LLO (or Near-Rectilinear Halo Orbit) down to the lunar surface. In all cases

STSV/MPSC USING CHEMICAL PROPULSION

We assume that an upgraded Starship Block 4 vehicle powered by Raptor 4 vacuum engines powers STSV or MPSC missions to the Moon. Assume a five day mission each way, or 10 days for a round trip. Weight of consumables from Table 2 for a round trip is 5.72 t, assume this is included in the 50 t equipment/misc. budget.

1. **STSV one-way Moon trip assumptions used:**

- **340 t** payload already in **350 km circular LEO**
- Destination: **110 km circular Low Lunar Orbit**, and **Return to 450 km LEO**
- Propulsion: **6 × Raptor 4 Vacuum engines**
- Thrust: **330 t-force per engine**
- Total thrust: **1,980 t-force ≈ 19.4 MN**
- Vacuum Isp: **380 s assumed**
- Propellant: **LOX / liquid methane**
- Mixture ratio: **O/F = 3.6**
- Tank/feed/structure allowance: **8% of propellant mass**
- Engine dry mass: **~12 t per six-engine stage**
- One-way Δv: **~4.0 km/s**, consistent with the attached cislunar transfer reference, which gives ~4 km/s for TLI plus lunar circularization.

Item	Mass
Methalox propellant	**~801 t**
Tanks/feed/structure	**~64 t**
Engines	**~12 t**
Wet return propulsion stage	**~877 t**

2. **Round trip: 350 km LEO → 110 km LLO → 350 km LEO**

Recommended round-trip architecture: two Starship-derived stages. A single-stage fully reusable round trip is technically calculable, but at **~7,277 t propulsion-system mass**, it is not a sensible Starship-derived architecture.

1. **Outbound propulsion stage** performs LEO → LLO, then is discarded or parked in lunar orbit.

2. **Return propulsion stage** remains full until departure from LLO, then returns the 340 t payload to 350 km LEO. Parameters same as the one-way stage.

Table 3. STSV Chemical Propulsion Moon Missions Results

Case	Propellant	Tanks/feed/ structure	Engines	Wet propulsion system	Total stack with 340 t payload
One-way LEO → LLO	**~801 t**	**~64 t**	**~12 t**	**~877 t**	**~1,217 t**
Round trip, single stage	**~6,727 t**	**~538 t**	**~12 t**	**~7,277 t**	**~7,617 t**
Round trip, two-stage recommended	**~3,598 t**	**~288 t**	**~24 t**	**~3,910 t**	**~4,250 t**

<u>STSV Chemical Propulsion System Summary</u>:

4. MPSC one-way Moon trip assumptions used: Using the same assumptions as the previous calculation, but increasing the payload from 340 t to 698 t. For a 10 day round trip mission to the Moon, weight of consumables from Table 3 is 11.44 t. Count this as part of the 200 t equipment/misc. allocation.

1. One-way: 350 km LEO → 110 km LLO

Item	**Mass**
Payload	**698 t**
LOX	**~1,265 t**
Liquid methane	**~351 t**
Total methalox propellant	**~1,616 t**
Tanks/feed/structure, 8%	**~129 t**
Six Raptor Vacuum engines	**~12 t**
Wet propulsion system	**~1,757 t**
Total LEO departure stack	**~2,455 t**

5. Recommended round-trip architecture: two Starship-derived stages

This is more practical:

1. **Outbound stage** carries the payload plus a full return stage from LEO to LLO. This stage must deliver the 698 t payload + 1,757 t return stage to lunar orbit.

2. **Return stage** carries the 698 t payload from LLO back to 350 km LEO.

Return stage: same as the one-way stage:

Outbound Stage		Return Stage	
Item	**Mass**	**Item**	**Mass**
LOX	**~4,394 t**	LOX	**~1,265 t**
Liquid methane	**~1,221 t**	Liquid methane	**~351 t**
Total methalox propellant	**~5,615 t**	**Total methalox propellant**	**~1,616 t**
Tanks/feed/structure	**~449 t**	Tanks/feed/structure	**~129 t**
Engines	**~12 t**	Engines	**~12 t**
Wet outbound propulsion stage	**~6,076 t**	**Wet return propulsion stage**	**~1,757 t**

Two-stage round-trip total

Item	Mass
Payload	**698 t**
Total LOX	**~5,659 t**
Total liquid methane	**~1,572 t**
Total methalox propellant	**~7,230 t**
Total tanks/feed/structure	**~578 t**
Total engines	**~24 t**
Total wet propulsion system	**~7,833 t**
Total LEO departure stack	**~8,531 t**

Table 4. MPSC Chemical Propulsion Moon Missions Results

Case	Propellant	Tanks/feed/ structure	Engines	Wet propulsion system	Total stack with 698 t payload
One-way LEO → LLO	~1,616 t	~129 t	~12 t	~1,757 t	~2,455 t
Round trip, single-stage	~13,569 t	~1,086 t	~12 t	~14,667 t	~15,365 t
Round trip, staged architecture	~7,230 t	~578 t	~24 t	~7,833 t	~8,531 t

MPSC Chemical Propulsion Summary

The staged round-trip architecture is still enormous, but it cuts the propulsion-system mass by almost half compared with a single-stage reusable tug.

STSV/MPSC USING NUCLEAR THERMAL PROPULSION

Common assumptions

Parameter	Assumption
NTP specific impulse	**950 s**
Effective exhaust velocity	**9.316 km/s**
LH_2 density	**70.8 kg/m³**
LH_2 cooldown, chilldown, boiloff and performance reserve	**3% of burn propellant**
LH_2 tank/insulation/plumbing mass	**11% of loaded LH_2**
TLI Δv	**3.15 km/s**
Tank ullage	**5%**
Lunar orbit insertion	**0.85 km/s**
Total one-way Δv	**4.00 km/s**
Trans-Earth injection	**0.85 km/s**
Earth orbit insertion	**3.15 km/s**
Round-trip Δv	**8.00 km/s**

1. STSV NTP One-way lunar mission with 340 t payload

Engine system

Item	Value
NTR engines	**6 × 178.4 kN**
Total thrust	**1.070 MN**
Engineering-module mass	**72.62 t**
Initial departure stack mass	**682.01 t**
Initial thrust-to-weight ratio	**~0.160**

Whole propulsion-system mass

Component	Mass
Loaded LH_2	**242.69 t**
LH_2 tanks, insulation and plumbing	**26.70 t**
Six-engine NTP bus	**72.62 t**
Total wet propulsion system	**342.01 t**
Payload	**340.00 t**
Total LEO departure stack	**682.01 t**

2. STSV NTP Round trip lunar mission with a 340 t payload

Recommended engine system

Item	**Value**
NTP engines	**6 × 178.4 kN**
Total thrust	**1.070 MN**
NTP engineering-module mass	**72.62 t**
Initial thrust-to-weight ratio	**~0.097**
Approximate peak reactor thermal power	**~5.5–6 GWth**

This preserves two independent three-engine groups and provides meaningful engine-out redundancy.

Whole propulsion-system mass

Component	**Mass**
Loaded LH_2	**643.84 t**
LH_2 tanks, insulation and tank plumbing	**70.82 t**
Six-engine NTP engineering bus	**72.62 t**
Total wet propulsion system	**787.28 t**
Payload	**340.00 t**
Initial LEO departure mass	**1,127.28 t**

3. MPSC NTP one-way lunar mission with 698 t payload

Engine system

Item	**Value**
NTR engines	**12 × 178.4 kN**
Total thrust	**2.141 MN**
Engineering-module mass	**145.24 t**
Initial departure stack mass	**1,393.77 t**
Initial thrust-to-weight ratio	**~0.157**

Whole propulsion-system mass

Component	**Mass**
Loaded LH_2	**495.97 t**
LH_2 tanks, insulation and plumbing	**54.56 t**
Twelve-engine NTP bus	**145.24 t**
Total wet propulsion system	**695.77 t**
Payload	**698.00 t**
Total LEO departure stack	**1,393.77 t**

4. MPSC NTP round trip lunar mission with 698 t payload

Item	Value
NTP engines	**12 × 178.4 kN**
Total thrust	**2.141 MN**
NTP engineering-module mass	**145.24 t**
Initial thrust-to-weight ratio	**~0.095**
Approximate peak reactor thermal power	**~11–12 GWth**

The engine count doubles with payload so that acceleration and burn durations remain nearly equal to the 340 t STSV.

Whole propulsion-system mass

Component	Mass
Loaded LH_2	**1,315.76 t**
LH_2 tanks, insulation and tank plumbing	**144.73 t**
Twelve-engine NTP engineering bus	**145.24 t**
Total wet propulsion system	**1,605.73 t**
Payload	**698.00 t**
Initial LEO departure mass	**2,303.73 t**

Table 5 STSV and MPSC NTP Propulsion Moon Missions Summary

Payload	Engines	Thrust	Loaded LH_2	Tank mass	Engine-bus mass	Wet propulsion system	Total LEO stack
STSV 340 t	6	1.070 MN	**643.84 t**	70.82 t	72.62 t	**787.28 t**	**1,127.28 t**
MPSC 698 t	12	2.141 MN	**1,315.76 t**	144.73 t	145.24 t	**1,605.73 t**	**2,303.73 t**

5. MPSC NTP round trip Mars mission with 698 t payload

Assume departure from 450 km LEO and arrival into 350 km Low Mars Orbit (LMO).

Mission Δv and timing

Phase	Δv	Time
TMI: 450 km LEO → Mars transfer	**3.56 km/s**	burn ~51 min
Earth → Mars transfer	—	**~259 days**
MOI: Mars arrival → 350 km LMO	**2.09 km/s**	burn ~21 min
Mars stay / ISRU refueling	—	**500+ days**

Phase	Δv	Time
TEI: 350 km LMO → Earth transfer	**2.09 km/s**	burn ~32 min
Mars → Earth transfer	—	**~259 days**
EOI: Earth arrival → 450 km LEO	**3.56 km/s**	burn ~39 min

Total round-trip propulsive Δv:

Minimum total mission duration with a 500-day Mars stay:

Outbound leg: Earth → Mars

Nominal LH_2 use:

Burn	LH_2 used
TMI	**~587 t**
MOI	**~236 t**
Nominal outbound LH_2	**~823 t**

Recommended loaded LH_2 including boiloff, chilldown, and operating reserve:

Outbound propulsion mass:

Component	Mass
Loaded LH_2	**~835 t**
LH_2 tanks + active cryofluid management	**~125 t**
NTP engines / reactors / shield / thrust structure	**~167 t**
Wet propulsion system	**~1,127 t**
Payload	**735.6 t**
Total Earth departure stack	**~1,863 t**

Mars ISRU return leg

For the return, do **not** carry return LH_2 from Earth. The best architecture refuels in **350 km Mars orbit** using ISRU-produced hydrogen, following the same logic as the Discovery/Rhea ISRU refueling concept, where local resources are processed into LH_2/LOX and transferred to the spacecraft for the return trip.

Nominal return LH_2 use:

Burn	LH_2 used
TEI	**~372 t**
EOI	**~452 t**
Nominal return LH2	**~823 t**

Recommended Mars-orbit LH2 delivery:

Mars departure stack after ISRU refueling:

Component	Mass
Loaded ISRU LH_2	**~835 t**
Return LH_2 tanks + active CFM	**~125 t**
NTP engines / reactors / shield / structure	**~167 t**
Payload	**735.6 t**
Total Mars departure stack	**~1,863 t**

Reactor / engine system

Parameter	Estimate
Engines	**16 × 25 klbf-class HALEU NTP**
Total thrust	**1.78 MN**
Isp	**950 s**
Mass flow at full thrust	**~191 kg/s**
LH2 consumption rate	**~11.5 t/min**
Jet power	**~8.3 GW**
Reactor thermal power	**~9–10 GWth total**
Engine / reactor / shield / thrust bus	**~167 t**

The 16-engine cluster remains acceptable for the higher payload. Initial T/W is about **0.10**, which is still strong for a large NTP transfer stage; one uploaded Mars NTP trajectory study used a finite-burn NTP model with **900 s Isp** and **T/W = 0.057**, **so this architecture has comfortable thrust margin**.

Final updated mass summary

Quantity	Mass
Payload	**735.6 t**
Nominal LH_2 per leg	**~823 t**
Loaded LH_2 per leg with loss/reserve	**~835 t**
Tanks + active CFM per leg	**~125 t**
NTP engine/reactor/shield bus	**~167 t**
Maximum wet propulsion system at any one time	**~1,127 t**
Maximum stack mass at Earth or Mars departure	**~1,863 t**
Total mission LH_2 handled, Earth + Mars ISRU	**~1,670 t**
Total propulsion assets + handled LH_2	**~2,087 t**

Bottom line

For a **735.6 t payload**, the best final architecture is as follows

SUMMARY

STSV and MPSC Missions to the Moon

As stated previously, the ΔV requirements for missions from 450 km LEO to NRHO, L4, L5, and 110 km LLO are similar, so we chose ΔV ~ 4 km/sec each way as a benchmark for lunar missions.

Table 6. STSV and MPSC One-Way Lunar Missions (all values approximate)

Vehicle	Propulsion Type	Engines	Loaded Propellant Mass	Loaded Propellant Volume	Wet Propulsion Mass	Total LEO departure stack
340 t STSV	Chemical	6 x Raptor 4 Vacuum	801 t LOX/Methane	1,010 m^3 LOX/Methane	877 t	1,217 t
340 t STSV	NTP	6 x 178.4 kN	242.7 t LH_2	3,599 m^3 LH_2	342 t	682 t
698 t MPSC	Chemical	6 x Raptor 4 Vacuum	1,616 t LOX/Methane	2,038 m^3 LOX/Methane	1,757 t	2,455 t
698 t MPSC	NTP	12 x 178.4 kN	496 t LH_2	7,356 m^3 LH_2	696 t	1,394 t

Table 7. STSV and MPSC Round Trip, Two Stage Lunar Missions (all values approximate)

Vehicle	Propulsion Type	Engines	Loaded Propellant Mass	Loaded Propellant Volume	Wet Propulsion Mass	Total LEO departure stack
340 t STSV	Chemical	6 x Raptor 4 Vacuum	3,598 t LOX/Methane	4,538 m^3 LOX/Methane	3,910 t	4,250 t
340 t STSV	NTP	6 x 178.4 kN	643.84 t LH_2	9,547 m^3 LH_2	787 t	1,127 t
698 t MPSC	Chemical	6 x Raptor 4 Vacuum	7,230 t LOX/Methane	9,118 m^3 LOX/Methane	7,833 t	8,531 t
698 t MPSC	NTP	12 x 178.4 kN	1,316 t LH_2	m^3 LH_2	1,606 t	2,304 t

Table 8. MPSC NTP Propulsion One-Way Mars Mission (all values approximate)

Engines	Loaded Propellant Mass	Loaded Propellant Volume	Wet Propulsion Mass	Total LEO departure stack
16 x 178.4 kN	835 t LH_2	12,383 m^3 LH_2	1,127 t	1,863 t
16 x 178.4 kN	835 t LH_2	m^3 LH_2	1,127 t	1,863 t

Conclusion

The development and construction of reusable launchers in the last decade by SpaceX and other companies, with payloads up to 200 metric tons, will enable us to build space stations and spacecrafts based on the payload bays of these launchers. For example, 7 launches of Starship Block 4 will suffice for assembling the 698 tonnes MPSC for one-way Moon missions, where ISRU can be used to supply a return trip. Similarly, 10 launches will suffice for assembling a 736 tonne MPSC on one-way missions to Mars, assuming ISRU resources can supply a return trip. Modules and structural elements, either rigid or inflatable, can be transported into low Earth orbit and assembled by astronauts and robots to build big structures in space and on the Moon. Utilizing and processing lunar regolith will provide space-made fuels for rocketry. Space factories preferably in the Lagrange points L4 and L5 of the Earth-Moon system are envisioned to process lunar and asteroid material and produce goods for Earth and for future space settlers to build their artificial habitats. Last not least the expanding human civilization into outer space will create a crucial impetus for Earth´s economy.

Details of propulsion system analysis and assumptions made are available at:

Detailed Propulsion System Analyses and Assumptions

Appendix A
340 Metric Ton Space to Space Vehicle

Appendix A.1

Lagrange Point and Moon Missions

Appendix A.1.1 Chemical Propulsion

We assume that an upgraded Starship Block 4 vehicle powered by Raptor 4 vacuum engines powers STSV missions to the Moon. Assume a five day mission each way, or 10 days for a round trip. Weight of consumables from Table 2 for a round trip is 5.72 t, assume this is included in the 50 t equipment/misc. budget.

1. **One-way trip to the Moon assumptions used:**

- **340 t** payload already in **350 km circular LEO**
- Destination: **110 km circular Low Lunar Orbit**
- Propulsion: **6 × Raptor 4 Vacuum engines**
- Thrust: **330 t-force per engine**
- Total thrust: **1,980 t-force ≈ 19.4 MN**
- Vacuum Isp: **380 s assumed**
- Propellant: **LOX / liquid methane**
- Mixture ratio: **O/F = 3.6**
- Tank/feed/structure allowance: **8% of propellant mass**
- Engine dry mass: **~12 t per six-engine stage**
- One-way Δv: **~4.0 km/s**, consistent with the attached cislunar transfer reference, which gives ~4 km/s for TLI plus lunar circularization.

Tank volumes for one-way trip to the Moon

Using approximate densities:

- LOX: **1,141 kg/m³**
- Liquid methane: **422 kg/m³**

Tank	Propellant mass	Fluid volume	With 5% ullage
LOX tank	~627 t	~549 m³	**~577 m³**
Methane tank	~174 t	~413 m³	**~433 m³**
Total	**~801 t**	**~962 m³**	**~1,010 m³**

So the one-way tug needs roughly:

Burn time

At 19.4 MN thrust and 380 s Isp:

Total burn time:

Most of the ~3-day trip is coast time.

2. Round trip: 350 km LEO → 110 km LLO → 350 km LEO

Recommended round-trip architecture: two Starship-derived stages

3. **Outbound propulsion stage** performs LEO → LLO, then is discarded or parked in lunar orbit.
4. **Return propulsion stage** remains full until departure from LLO, then returns the 340 t payload to 350 km LEO.

Return stage

Same as the one-way stage:

Item	Mass
Methalox propellant	**~801 t**
Tanks/feed/structure	**~64 t**
Engines	**~12 t**
Wet return propulsion stage	**~877 t**

Outbound stage

The outbound stage must deliver both the **340 t payload** and the **full 877 t return stage** to LLO.

Item	Mass
LOX	**~2,189 t**
Liquid methane	**~608 t**
Total methalox propellant	**~2,797 t**
Tanks/feed/structure, 8%	**~224 t**
Six Raptor 4 Vacuum engines	**~12 t**
Wet outbound propulsion stage	**~3,033 t**

Total two-stage round-trip system

Item	Mass
Outbound propellant	**~2,797 t**
Return propellant	**~801 t**
Total methalox propellant	**~3,598 t**
Total tanks/feed/structure	**~288 t**
Total Raptor Vacuum engines	**~24 t**
Total wet propulsion system	**~3,910 t**
Payload	**340 t**
Total LEO departure stack	**~4,250 t**

Two-stage tank volumes

Stage	LOX volume + ullage	Methane volume + ullage	Total tank volume
Outbound stage	~2,015 m³	~1,513 m³	**~3,528 m³**
Return stage	~577 m³	~433 m³	**~1,010 m³**
Total	**~2,592 m³**	**~1,946 m³**	**~4,538 m³**

Final summary

Case	Propellant	Tanks/feed/structure	Engines	Wet propulsion system	Total stack with 340 t payload
One-way LEO → LLO	**~801 t**	**~64 t**	**~12 t**	**~877 t**	**~1,217 t**
Round trip, single stage	**~6,727 t**	**~538 t**	**~12 t**	**~7,277 t**	**~7,617 t**
Round trip, two-stage recommended	**~3,598 t**	**~288 t**	**~24 t**	**~3,910 t**	**~4,250 t**

About 21 to 22 Starship Block 4 launches will be required to place the 4,250 t STSV into LEO.

A single-stage fully reusable round trip is technically calculable, but at **~7,277 t propulsion-system mass**, it is not a sensible Starship-derived architecture.

Appendix A.1.2 Nuclear Thermal Propulsion

Common assumptions

Parameter	Assumption
NTP specific impulse	**950 s**
Effective exhaust velocity	**9.316 km/s**
TLI Δv	**3.15 km/s**
Lunar orbit insertion	**0.85 km/s**
Trans-Earth injection	**0.85 km/s**
Earth orbit insertion	**3.15 km/s**
Round-trip Δv	**8.00 km/s**
LH_2 tank dry mass	**11% of loaded LH2**
Cooldown, chilldown, boiloff and performance reserve	**3% of burn propellant**
LH_2 density	**70.8 kg/m³**
Tank ullage	**5%**

A roughly 4 km/s one-way requirement for TLI plus lunar circularization is consistent with the cislunar NTP reference. That paper also identifies a conventional lunar transfer time of about three days, while the Discovery lunar DRM uses approximately **3.4 days each way**.

I retain the same engine counts as in the preceding round-trip cases:

- **340 t payload:** two engineering modules, six engines.
- **698 t payload:** four engineering modules, twelve engines.

1. STSV NTP One-way mission with 340 t payload

Engine system

Item	Value
NTR engines	**6 × 178.4 kN**
Total thrust	**1.070 MN**
Engineering-module mass	**72.62 t**
Initial departure stack mass	**682.01 t**
Initial thrust-to-weight ratio	**~0.160**

LH2 requirement by burn

Burn	Δv	LH_2 consumed	LH_2 loaded, including 3% reserve
TLI	3.15 km/s	**195.66 t**	**201.53 t**
Lunar-orbit insertion	0.85 km/s	**39.96 t**	**41.16 t**
Total	**4.00 km/s**	**235.62 t**	**242.69 t**

The approximately **7.07 t difference** between loaded and consumed hydrogen covers cooldown, chilldown, residuals and operating margin. The Discovery reference explicitly uses LH_2 after major burns to remove reactor decay heat before the Brayton cooling loops can handle the remaining thermal load.

Tank configuration

A practical modular arrangement is:

Tank group	Number	LH_2 per tank	Dry mass per tank	Volume per tank with ullage	Spherical-equivalent diameter
TLI drop tanks	**2**	**100.77 t**	**11.08 t**	**~1,494 m³**	**~14.2 m**
LOI tank	**1**	**41.16 t**	**4.53 t**	**~610 m³**	**~10.5 m**
Total	**3 tanks**	**242.69 t**	**26.70 t**	**~3,599 m³**	—

The two TLI tanks are discarded after Earth departure. The LOI tank remains attached through lunar capture and may then be detached or retained for later use.

Whole propulsion-system mass

Component	Mass
Loaded LH_2	**242.69 t**
LH_2 tanks, insulation and plumbing	**26.70 t**
Six-engine NTP bus	**72.62 t**
Total wet propulsion system	**342.01 t**
Payload	**340.00 t**
Total LEO departure stack	**682.01 t**

Burn duration

At 1.070 MN total thrust:

Burn	Approximate duration
TLI	**28.4 min**
LOI	**5.8 min**
Total full-power firing time	**34.2 min**

After lunar-orbit insertion:

- Mass before disposing of the LOI tank: **~418.35 t**
- Payload plus NTP bus after tank disposal: **~412.62 t**

2. STSV NTP Round trip with a 340 t payload

Recommended engine system

Use two Discovery-derived engineering modules:

Item	Value
NTP engines	**6 × 178.4 kN**

Item	Value
Total thrust	**1.070 MN**
NTP engineering-module mass	**72.62 t**
Initial thrust-to-weight ratio	**~0.097**
Approximate peak reactor thermal power	**~5.5–6 GWth**

- This preserves two independent three-engine groups and provides meaningful engine-out redundancy.
- **Propellant by maneuver**

Maneuver	Loaded LH_2	LH_2 consumed	Reserve remaining before tank disposal
TLI	**333.10 t**	323.40 t	9.70 t
LOI	**68.04 t**	66.06 t	1.98 t
TEI	**61.25 t**	59.47 t	1.78 t
EOI	**181.44 t**	176.15 t	5.28 t
Total	**643.84 t**	**625.08 t**	**18.76 t**

- The reserve includes NTR reactor cooldown hydrogen. The Discovery reference explicitly uses LH2 after each burn for reactor decay-heat removal and reports cooldown requirements of roughly 1.8–3.6 t per burn for its three-engine module.
- **Tank configuration**
- The minimum-mass configuration uses five tank modules:

Tank set	Number	LH_2 per tank	Dry mass per tank	Spherical-equivalent diameter
TLI drop tanks	**2**	166.55 t	18.32 t	**16.8 m**
LOI tank	**1**	68.04 t	7.48 t	**12.4 m**
TEI tank	**1**	61.25 t	6.74 t	**12.0 m**
EOI tank	**1**	181.44 t	19.96 t	**17.3 m**
Total	**5 tanks**	**643.84 t**	**70.82 t**	—

- The TLI, LOI, and TEI tanks are discarded after their respective burns. The EOI tank remains attached until Earth-orbit capture and can then be removed or refurbished.
- **Whole propulsion-system mass**

Component	Mass
Loaded LH_2	**643.84 t**
LH_2 tanks, insulation and tank plumbing	**70.82 t**
Six-engine NTP engineering bus	**72.62 t**
Total wet propulsion system	**787.28 t**
Payload	**340.00 t**
Initial LEO departure mass	**1,127.28 t**

- **Approximate burn durations**

Burn	Duration
TLI	**46.9 min**
LOI	**9.6 min**
TEI	**8.6 min**
EOI	**25.6 min**
Total full-power firing time	**~90.7 min**

Appendix B
698 Metric Ton Multipurpose Space Cruiser

Appendix B.1
Lagrange Point and Moon Missions

Appendix B.1.1 MPSC Missions to the Moon Using Chemical Propulsion

Using the same assumptions as the previous calculation, but increasing the payload from 340 t to 698 t. For a 10 day round trip mission to the Moon, weight of consumables from Table 3 is 11.44 t. Count this as part of the 200 t equipment/misc. allocation.

1. One-way: 350 km LEO → 110 km LLO

Item	Mass
Payload	**698 t**
LOX	**~1,265 t**
Liquid methane	**~351 t**
Total methalox propellant	**~1,616 t**
Tanks/feed/structure, 8%	**~129 t**
Six Raptor Vacuum engines	**~12 t**
Wet propulsion system	**~1,757 t**
Total LEO departure stack	**~2,455 t**

One-way tank volume

Tank	Propellant mass	Tank volume with 5% ullage
LOX	~1,265 t	**~1,164 m³**
Liquid methane	~351 t	**~874 m³**
Total tank volume	—	**~2,038 m³**

At full thrust, mass flow is about **5.21 t/s**, so total engine burn time is:

2. Recommended round-trip architecture: two Starship-derived stages

This is more practical:

3. **Outbound stage** carries the payload plus a full return stage from LEO to LLO.
4. **Return stage** carries the 698 t payload from LLO back to 350 km LEO.

Return stage

Same as the one-way stage:

Item	Mass
LOX	**~1,265 t**
Liquid methane	**~351 t**
Total methalox propellant	**~1,616 t**
Tanks/feed/structure	**~129 t**
Engines	**~12 t**
Wet return propulsion stage	**~1,757 t**

Outbound stage

This stage must deliver:

to lunar orbit.

Item	Mass
LOX	**~4,394 t**
Liquid methane	**~1,221 t**
Total methalox propellant	**~5,615 t**
Tanks/feed/structure	**~449 t**
Engines	**~12 t**
Wet outbound propulsion stage	**~6,076 t**

Two-stage round-trip total

Item	Mass
Payload	**698 t**
Total LOX	**~5,659 t**
Total liquid methane	**~1,572 t**
Total methalox propellant	**~7,230 t**
Total tanks/feed/structure	**~578 t**
Total engines	**~24 t**
Total wet propulsion system	**~7,833 t**
Total LEO departure stack	**~8,531 t**

Two-stage tank volume

Stage	LOX volume with ullage	Methane volume with ullage	Total
Outbound stage	**~4,044 m³**	**~3,037 m³**	**~7,080 m³**
Return stage	**~1,164 m³**	**~874 m³**	**~2,038 m³**
Total	**~5,207 m³**	**~3,911 m³**	**~9,118 m³**

Final summary

Case	Propellant	Tanks/feed/structure	Engines	Wet propulsion system	Total stack with 698 t payload
One-way LEO → LLO	**~1,616 t**	**~129 t**	**~12 t**	**~1,757 t**	**~2,455 t**
Round trip, single-stage	**~13,569 t**	**~1,086 t**	**~12 t**	**~14,667 t**	**~15,365 t**
Round trip, staged architecture	**~7,230 t**	**~578 t**	**~24 t**	**~7,833 t**	**~8,531 t**

About 43 Starship Block 4 launches will be required to place the 8,531 t MPSC into LEO.

The staged round-trip architecture is still enormous, but it cuts the propulsion-system mass by almost half compared with a single-stage reusable tug.

Appendix B.1.2 MPSC Missions to the Moon Using Nuclear Thermal Propulsion

One-way mission with 698 t payload

Engine system

Item	Value
NTR engines	**12 × 178.4 kN**
Total thrust	**2.141 MN**
Engineering-module mass	**145.24 t**
Initial departure stack mass	**1,393.77 t**
Initial thrust-to-weight ratio	**~0.157**

LH2 requirement by burn

Burn	Δv	LH_2 consumed	LH_2 loaded, including 3% reserve
TLI	3.15 km/s	**399.86 t**	**411.85 t**
Lunar-orbit insertion	0.85 km/s	**81.67 t**	**84.12 t**
Total	**4.00 km/s**	**481.53 t**	**495.97 t**

Tank configuration

A convenient arrangement uses the same approximate tank sizes as the 340 t vehicle, but doubles the number:

Tank group	Number	LH_2 per tank	Dry mass per tank	Volume per tank with ullage	Spherical-equivalent diameter
TLI drop tanks	**4**	**102.96 t**	**11.33 t**	**~1,527 m³**	**~14.3 m**
LOI tanks	**2**	**42.06 t**	**4.63 t**	**~624 m³**	**~10.6 m**
Total	**6 tanks**	**495.97 t**	**54.56 t**	**~7,356 m³**	—

The four large tanks are discarded after TLI. The two smaller tanks provide lunar capture propellant and can be discarded after insertion into the 110 km lunar orbit.

Whole propulsion-system mass

Component	Mass
Loaded LH_2	**495.97 t**
LH_2 tanks, insulation and plumbing	**54.56 t**
Twelve-engine NTP bus	**145.24 t**
Total wet propulsion system	**695.77 t**
Payload	**698.00 t**
Total LEO departure stack	**1,393.77 t**

Burn duration

At 2.141 MN total thrust:

Burn	Approximate duration
TLI	**29.0 min**
LOI	**5.9 min**
Total full-power firing time	**34.9 min**

After lunar-orbit insertion:

- Mass before disposing of the LOI tanks: **~854.94 t**
- Payload plus NTP buses after tank disposal: **~843.24 t**

Final one-way comparison

Payload	NTR engines	Total thrust	Loaded LH_2	Tank mass	Engine-bus mass	Wet propulsion system	Total departure stack
340 t	6	1.070 MN	**242.69 t**	**26.70 t**	**72.62 t**	**342.01 t**	**682.01 t**
698 t	12	2.141 MN	**495.97 t**	**54.56 t**	**145.24 t**	**695.77 t**	**1,393.77 t**

STSV and MPSC Missions to the Moon Using Nuclear Thermal Propulsion

2. Round trip with a 698 t payload

Recommended engine system

Use four Discovery-derived engineering modules:

Item	Value
NTP engines	**12 × 178.4 kN**
Total thrust	**2.141 MN**
NTP engineering-module mass	**145.24 t**
Initial thrust-to-weight ratio	**~0.095**
Approximate peak reactor thermal power	**~11–12 GWth**

The engine count doubles with payload so that acceleration and burn durations remain nearly equal to the 340 t vehicle.

Propellant by maneuver

Maneuver	Loaded LH_2	LH_2 consumed	Reserve remaining before tank disposal
TLI	**680.74 t**	660.91 t	19.83 t
LOI	**139.04 t**	134.99 t	4.05 t
TEI	**125.18 t**	121.54 t	3.65 t
EOI	**370.79 t**	359.99 t	10.80 t
Total	**1,315.76 t**	**1,277.43 t**	**38.33 t**

Tank configuration

The minimum-mass configuration uses eight tank modules:

Tank set	Number	LH_2 per tank	Dry mass per tank	Spherical-equivalent diameter
TLI drop tanks	**4**	170.19 t	18.72 t	**16.9 m**
LOI tank	**1**	139.04 t	15.29 t	**15.8 m**
TEI tank	**1**	125.18 t	13.77 t	**15.2 m**
EOI tanks	**2**	185.40 t	20.40 t	**17.4 m**
Total	**8 tanks**	**1,315.76 t**	**144.73 t**	—

The two EOI tanks are retained through the return transfer. All other tank sets can be discarded immediately after use.

Whole propulsion-system mass

Component	Mass
Loaded LH_2	**1,315.76 t**
LH_2 tanks, insulation and tank plumbing	**144.73 t**
Twelve-engine NTP engineering bus	**145.24 t**

Component	Mass
Total wet propulsion system	**1,605.73 t**
Payload	**698.00 t**
Initial LEO departure mass	**2,303.73 t**

Approximate burn durations

Burn	Duration
TLI	**47.9 min**
LOI	**9.8 min**
TEI	**8.8 min**
EOI	**26.1 min**
Total full-power firing time	**~92.7 min**

Comparison with the previous methalox systems

Payload	Staged methalox propulsion system	Staged LH_2 NTP propulsion system	Approximate reduction
340 t	~3,910 t	**~787 t**	**~80%**
698 t	~7,833 t	**~1,606 t**	**~79%**

The reduction results primarily from increasing specific impulse from approximately 380 s for Raptor Vacuum to approximately 950 s for the advanced NTP concept.

Mission duration and hydrogen retention

Phase	Approximate duration
LEO → LLO	**~3.4 days**
LLO → LEO	**~3.4 days**
Total transfer time	**~6.8 days plus lunar stay**

The Discovery reference assumes active cryogenic refrigeration, helium cooling loops, multilayer insulation, reflective coatings and deployable shades. Its vehicle model assumes only **0.05% LH_2 leakage or boiloff per month**, making coast-phase loss negligible for short lunar missions.

Final results

Payload	Engines	Thrust	Loaded LH_2	Tank mass	Engine-bus mass	Wet propulsion system	Total LEO stack
340 t	6	1.070 MN	**643.84 t**	70.82 t	72.62 t	**787.28 t**	**1,127.28 t**
698 t	12	2.141 MN	**1,315.76 t**	144.73 t	145.24 t	**1,605.73 t**	**2,303.73 t**

Using the same assumptions as the preceding round-trip calculation, but deleting the **TEI and Earth-orbit-insertion burns**, the one-way mission is:

The attached cislunar NTP study uses approximately **4 km/s** for translunar injection plus lunar circularization and describes a conventional lunar transfer as roughly three days.

Common design assumptions

Parameter	**Value**
TLI Δv	**3.15 km/s**
Lunar-orbit insertion Δv	**0.85 km/s**
Total one-way Δv	**4.00 km/s**
NTP specific impulse	**950 s**
Effective exhaust velocity	**9.316 km/s**
Loaded-LH2 reserve	**3% of burn propellant**
Tank/insulation/plumbing mass	**11% of loaded LH_2**
LH2 density	**70.8 kg/m³**
Tank ullage	**5%**
Transfer duration	**about 3–3.5 days**

The reference *Spaceship Discovery* engineering module uses three gimbaled **178.4 kN** NTR engines, producing **535 kN total thrust**, at an assumed **950 s Isp**. Its total module mass is about **36.31 t**. The design includes Brayton power conversion, reactor shielding, radiators and structures.

Compared with the prior round-trip cases:

Payload	**Round-trip NTP propulsion**	**One-way NTP propulsion**	**Reduction**
340 t	~787 t	**~342 t**	**~445 t / 57%**
698 t	~1,606 t	**~696 t**	**~910 t / 57%**

These are preliminary system-level estimates. A realistic finite-burn trajectory, detailed tank stress analysis, reactor cooldown model and payload structural-integration study could change the results by approximately **±15%**.

Appendix B.2

Mars Missions

MPSC Round Trip to Mars Specifics

Mission Δv and timing

Phase	**Δv**	**Time**
TMI: 450 km LEO → Mars transfer	**3.56 km/s**	burn ~51 min
Earth → Mars transfer	—	**~259 days**
MOI: Mars arrival → 350 km LMO	**2.09 km/s**	burn ~21 min
Mars stay / ISRU refueling	—	**500+ days**
TEI: 350 km LMO → Earth transfer	**2.09 km/s**	burn ~32 min
Mars → Earth transfer	—	**~259 days**
EOI: Earth arrival → 450 km LEO	**3.56 km/s**	burn ~39 min

Minimum total mission duration with a 500-day Mars stay:

Outbound leg: Earth → Mars

Nominal LH_2 use:

Burn	LH_2 used
TMI	**~587 t**
MOI	**~236 t**
Nominal outbound LH2	**~823 t**

Recommended loaded LH_2 including boiloff, chilldown, and operating reserve:

Outbound propulsion mass:

Component	Mass
Loaded LH_2	**~835 t**
LH_2 tanks + active cryofluid management	**~125 t**
NTP engines / reactors / shield / thrust structure	**~167 t**
Wet propulsion system	**~1,127 t**
Payload	**735.6 t**
Total Earth departure stack	**~1,863 t**

Mars ISRU return leg

Do **not** carry Earth-return LH2 from Earth. Carry only the outbound LH2 from Earth, arrive in Mars orbit nearly empty, wait through the long conjunction-class stay, then refill the return tanks in **350 km Mars orbit** before TEI. This follows the logic in the NTP sources: NTP provides high thrust and >900 s-class Isp, hydrogen is the preferred propellant for high Isp, long-duration cryogenic storage needs active cryofluid management (CFM), and conjunction-class Mars missions naturally support long Mars stays of ~500+ days.

Nominal return LH2 use:

Burn	LH_2 used
TEI	**~372 t**
EOI	**~452 t**
Nominal return LH2	**~823 t**

Recommended Mars-orbit LH_2 delivery:

Mars departure stack after ISRU refueling:

Component	Mass
Loaded ISRU LH_2	**~835 t**
Return LH_2 tanks + active CFM	**~125 t**
NTP engines / reactors / shield / structure	**~167 t**

Component	Mass
Payload	**735.6 t**
Total Mars departure stack	**~1,863 t**

Tank layout

Outbound tank set

Tank group	Number	LH_2 per tank	Approx. tank volume each, incl. ullage	Spherical-equivalent diameter
TMI drop tanks	**4**	**~147 t**	**~2,180 m³**	**~16.1 m**
MOI tanks	**2**	**~118 t**	**~1,750 m³**	**~15.0 m**

Return tank set

Tank group	Number	LH_2 per tank	Approx. tank volume each, incl. ullage	Spherical-equivalent diameter
TEI tanks	**2**	**~186 t**	**~2,760 m³**	**~17.4 m**
EOI tanks	**2**	**~226 t**	**~3,350 m³**	**~18.6 m**

The return tanks are larger because the vehicle must retain the EOI propellant all the way from Mars departure to Earth arrival.

LH_2 loss allowance

For this architecture, the flight vehicle only stores one leg's worth of LH_2 at a time. With active cryofluid management (CFM), sunshields, multilayer insulation, vapor-cooled shields, low-conductivity struts, and zero-boiloff/very-low-boiloff cryocoolers, we have:

Segment	Predicted LH_2 loss
LEO checkout before TMI	**<0.5 t**
Outbound cruise after TMI	**~2–3 t**
Mars-orbit refueling / chilldown	**~1–2 t**
Return cruise after TEI	**~2–3 t**
Total mission non-propulsive LH_2 loss	**~6–8 t**

The NTP sources specifically identify long-duration cryogenic storage, cryocoolers, zero boiloff, and zero leakage as key NTP technologies, and the NTP vehicle trade source notes that LH_2's high performance comes with the need for active cryofluid management on long-duration missions.

Reactor / engine system

Parameter	Estimate
Engines	**16 × 25 klbf-class HALEU NTP**
Total thrust	**1.78 MN**
Isp	**950 s**
Mass flow at full thrust	**~191 kg/s**

Parameter	Estimate
LH2 consumption rate	**~11.5 t/min**
Jet power	**~8.3 GW**
Reactor thermal power	**~9–10 GWth total**
Engine / reactor / shield / thrust bus	**~167 t**

The 16-engine cluster remains acceptable for the higher payload. Initial T/W is about **0.10**, which is still strong for a large NTP transfer stage; one uploaded Mars NTP trajectory study used a finite-burn NTP model with **900 s Isp** and **T/W = 0.057**, ***so this architecture has comfortable thrust margin.***

Final updated mass summary

Quantity	Mass
Payload	**735.6 t**
Nominal LH_2 per leg	**~823 t**
Loaded LH_2 per leg with loss/reserve	**~835 t**
Tanks + active CFM per leg	**~125 t**
NTP engine/reactor/shield bus	**~167 t**
Maximum wet propulsion system at any one time	**~1,127 t**
Maximum stack mass at Earth or Mars departure	**~1,863 t**
Total mission LH_2 handled, Earth + Mars ISRU	**~1,670 t**
Total propulsion assets + handled LH_2	**~2,087 t**

References

Benton, Mark G. Sr.“Human Exploration of the Saturn System Using the Spaceship Discovery NTR Vehicle Architecture,” AIAA 2025-2562, 31 Jan 2025.

Benton, Mark G. Sr.“Crew Exploration Lander for Ganymede, Callisto, and Earth’s Moon – Vehicle Systems Design,” AIAA-2009-5179, August 2009.

Benton, Mark G. Sr.“Crew and Cargo Landers for Human Exploration of Mars – Vehicle System Design,” AIAA 2008-5156, July 2008.

Berger, Eric. “Elon Musk just gave another Mars speech–this time the vision seems tangible,” *Ars Technica* April 8, 2024.

Bukley A., Lawrence D., Clement G.: Generating artificial gravity onboard the Space Shuttle. In: Acta Astronautica 60 pages 472-478 (2007)

Bussard R.W. System Technical and Economic Features of QED-Engine Drive Space Transportation. In: 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference, AIAA 97-3071 (1997)

Bussard R.W.: An Advanced Fusion Energy System for Outer-Planet Space Propulsion. Space Technology and Applications International Forum 608 (2002)

Fraser S.D.: Electric Power System Options for Robotic Miners. In: Asteroids-Prospective Energy and Material Resources ed. V. Badescu, pages 247-270 (2013)

Grandl W., Autino A.V., Böck C.: Artificial Gravity Orbital Station (AGOS) -the simulation of gravity in a rotating spaces station, IAC-23, B3,3,1, x75535, 74th IAC, Baku, Azerbaijan (2023)

Guzek, Brian J., Horton, James F., Joyner, C. Russell II, "Mission Analysis for Mars Opposition Missions 2033-2048," AAS 20-065, February 2022.

Johnson R.C.: The slingshot effect, Department of Mathematical Sciences, University of Durham (2003)

Kokan, Timothy: Joyner, C. Russell II; Levack, Daniel J.H.; Muzek, Brian J.; Noble, Rodney W.; Reynolds, Christopher B. "Bimodal and Hybrid Nuclear/Thermal Nuclear Electric Propulsion Concepts for Human Mars Missions," AIAA 2026-1804, 8 January 2026.

Muzek, Brian J., Kokan, Timothy, Reynolds, C. Britton, Horton, James F., Joyner, C. Russell II, "High-Fidelity End-to-End Mars Opposition Class Trajectories for High and Low Thrust Nuclear Powered Spacecraft," AAS 23-176.

O´Neill G.K.: The High Frontier -Human Colonies in Space (1976).

Payton, Erik D., Merrill, Colby, Peck, Mason A., Savransky, Dmitry, "Optimal Cislunar Trajectories with Continuous, High-Thrust Nuclear Thermal Propulsion," AIAA 2025-0087, 3 January 2025.

Wittenberg L.J.: Cameron E.N., Kulcinski G.L., Ott S.H., Santarius J.F., Sviatoslavsky G.I., Sviatoslavsky I.N., Thompson H.E.: A Review of He-3 Resources and Acquisition for Use as Fusion Fuel, Fusion Technology 21(4), (1992)

Generative AI statement:

The authors declare that no generative Artificial Intelligence was used for the design of the Multipurpose Space Cruiser.