Near-Future Reusable Space Logistics Vehicles[*]

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Abstract

Key elements of the needed near-future, integrated, spacefaring logistics infrastructure are space-based, fully-reusable space logistics vehicles to provide transportation for passengers and cargo. This paper addresses the conceptual design of two such reusable spaceships using near-term technologies. The first is a space tug designed to support logistics operations in Low Earth Orbit (LEO). This space tug is sized to provide materiel handling, cargo transport, and passenger transport capabilities to support the construction and operation of orbiting logistics facilities and large spaceships. The second is a space ferry designed to transport cargo and passengers throughout the Earth-Moon system. This paper defines the assumptions and mission requirements used in the conceptual design of these two spaceships, summarizes the required Delta-V and maneuvering capabilities, assesses the technology maturity of the enabling technologies, summarizes the mass property estimates, provides illustrations of the conceptual designs, and provides examples of their use and integration into a near-future, integrated, spacefaring logistics infrastructure. The intent of this paper is to describe how current technologies can enable "aircraft-like" safe and routine transport of passengers and cargo within the Earth-Moon system.

Section 1: Introduction

A near-future spacefaring nation must have ready access to space and frequent mobility within the Earth-Moon system. By implication, such a near-future capability must be one that utilizes available technologies and is within the engineering capabilities of industry to now undertake. References 1 and 2

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addressed the first challenge of making use of near-term technologies to establish fully-reusable, two-stage, rocket-powered aerospaceplanes capable of transporting passengers and cargo to and from low Earth Orbit (LEO) with "aircraft-like" safety and operability. This paper extends this work to define the conceptual design of two near-term spaceships, to be based at LEO space logistics depots, which would provide materiel handling, cargo transport, and passenger transport capabilities to support the construction and operation of in-space logistics facilities and services. The first is a space tug that, as its name implies, is used within the vicinity of LEO space logistics depots to support depot operations. The second is a larger spaceship referred to as a space ferry. As its name implies, it is used to transport passengers and cargo within the Earth-Moon system.

Section 2: Background

Defining a "near-term" system design

A near-term system design is one that can enter full-scale system development without first requiring significant additional enabling technology maturation. Within the aerospace community, one method commonly used to assess the maturity of a proposed system design is to evaluate the maturity of the enabling technologies. For this purpose, the National Aeronautics and Space Administration (NASA) has developed a Technology Readiness Level (TRL) scale on which any technology—from the initial raw observations to the final operational application—can be ranked (see Figure 1). To be considered a mature design sufficient to

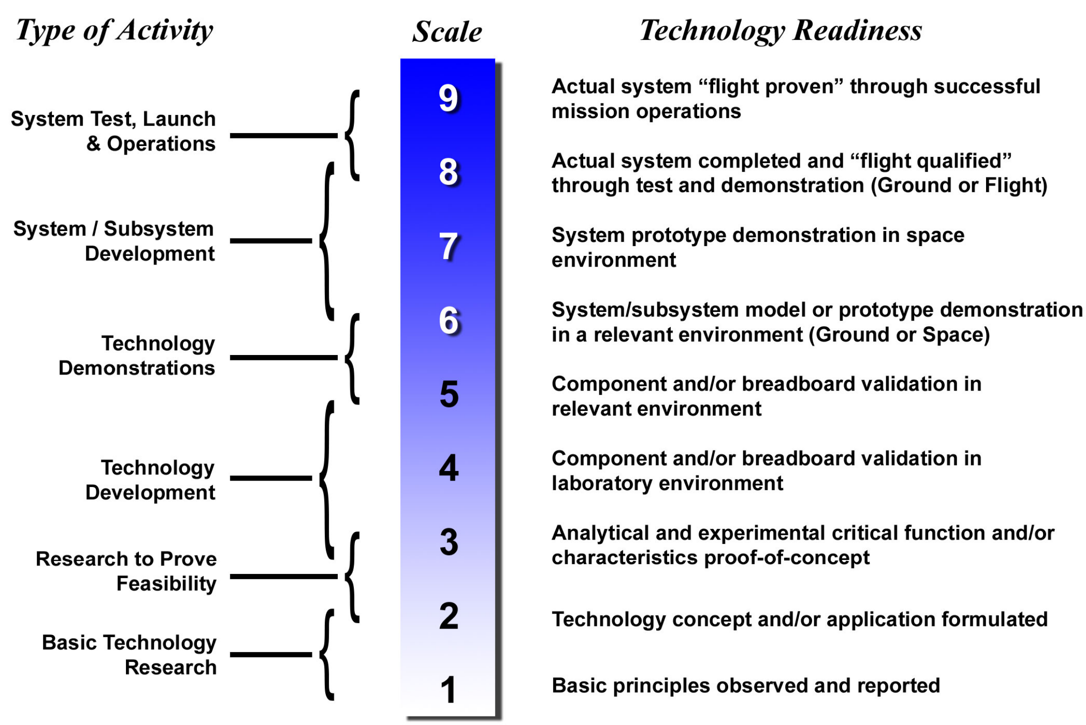

Figure 1. Technology readiness level scale.

support a decision to proceed with full-scale system development, all enabling technologies need to have achieved at least a TRL of 6—"system/subsystem model or prototype demonstration in a relevant environment (ground or space)"—prior to a formal decision to initiate system development.

With this level of maturity, a normal pace of system development will produce a production design in 3-4 years and a first production article in 5-6 years. A relevant benchmark is the Space Shuttle that started development in 1972 and was ready for first flight in 1980—about 8 years. However, many critical technologies, such as the thermal protection tiles and reusable rocket engines were only TRL 3-4 at the beginning of the system's development resulting in the development period being extended 2-3 years to complete the maturation of these technologies. Another relevant benchmark was the early 1990s Delta Clipper Experimental single-stage rocket technology demonstrator. This 20,000 kg, liquid hydrogen/liquid oxygen fueled, reusable rocket engine-propelled, low-speed demonstrator of a single-stage reusable space access system used TRL 8-9 technologies. It went from the preliminary design review to the first flight in about 18 months, demonstrating the value of using mature technologies.

Near-future Earth-to-orbit aerospaceplanes

Conceptual design studies undertaken at the Air Force Aeronautical Systems Center (Wright-Patterson Air Force Base, Ohio) have generated concepts for fully-reusable Earth-to-orbit space access systems—referred to herein as aerospaceplanes—using TRL 6-9 technologies.[1,2] Figure 2 depicts the general system configuration used in these studies. This is a two-stage, rocket-powered, vertical-takeoff, horizontal-landing system that carries cargo in an externally-mounted container or passengers in an externally-carried 6-10 passenger aerospaceplane. The standard cargo container carries payloads up to 3.7 m in diameter by 9.1 m in length. (Note: The space tug described below is shown inside the cargo container in this figure.)

The performance of this system concept, reflecting appropriate design and propulsion system margins, is also shown in Figure 2. The key performance values are the gross mass delivered to a 185 km circular orbit at 51.6° and the net cargo mass delivered to a 500 km circular orbit at the same inclination. The former value establishes the maximum mass of the passenger aerospaceplane while the latter defines the net mass of the cargo module that can be directly delivered to the LEO space logistics depot orbiting at 500 km. (Note: This aerospaceplane system concept is sized to be able to return a fully-loaded cargo container to the Earth.)

LEO space logistics depots

The aerospaceplanes will be most effective when serving as the transportation link between terrestrial spaceports and specific destinations in LEO. Two orbiting space logistics depots are assumed in this near-future spacefaring logistics infrastructure. The first is located at 28.5° at 482 km altitude while the second, as mentioned above, is located at 51.6° at 497 km altitude. The lower inclination

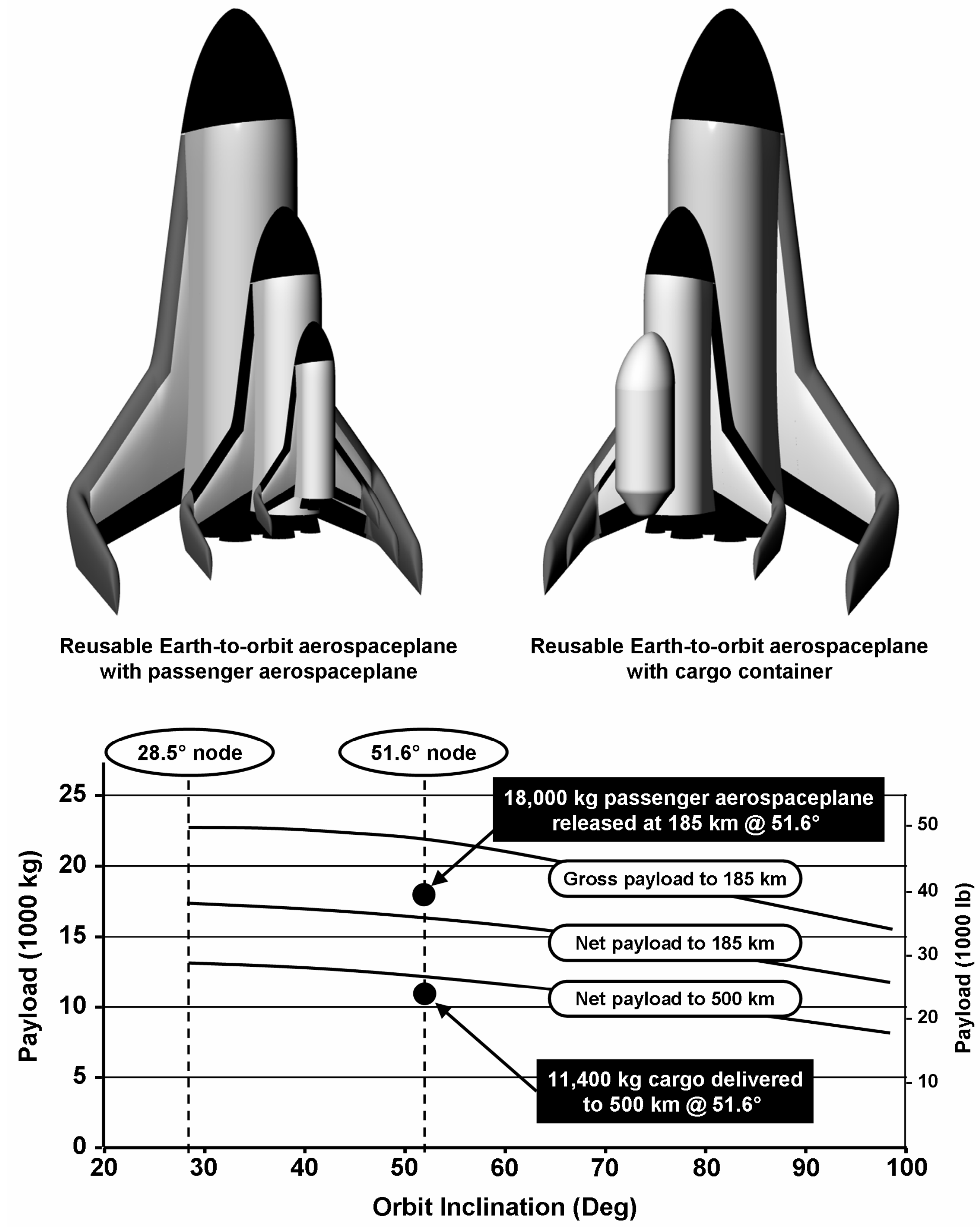

Figure 2. Generic illustrations of near-term, two-stage-to-orbit, fully-reusable aerospaceplane for passenger and cargo transport to LEO.

represents a due east launch from Kennedy Space Center (KSC) to provide the maximum payload performance into LEO. The higher inclination corresponds to that of the International Space Station (ISS) and is intended for use in providing logistics services and cargo and astronaut transport to the ISS. (Note: At these specified orbital altitudes, the ground track of each space logistics depot repeats

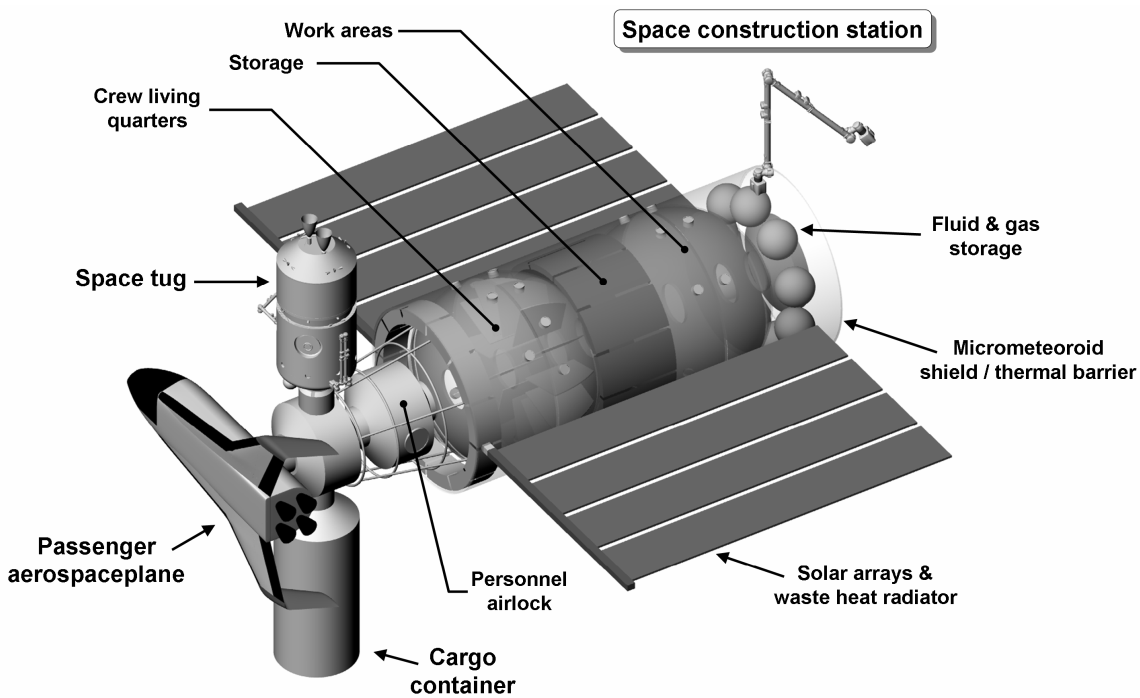

Figure 3. Spaced tug docked at space construction station.

about every 24 hours. With the correct selection of the Longitude of the Ascending Node, the ground track can be made to pass over KSC. This enables aerospaceplanes launched from KSC to access each depot daily.)

Each space logistics depot will consist of co-orbiting space logistics facilities that will receive cargo and passengers from the Earth, provide quarters for humans in LEO, provide propellants and on-site logistical services, and berth spaceships used for in-space mobility within the Earth-Moon system. The first facilities at each depot will be space construction stations used to assembly later facilities (see Figure 3). The space logistics base, shown in Figure 10, will be the first operational facility at each depot.

Mobility requirements within the Earth-Moon system

With the two LEO space logistics depots as the point of origination, it is possible to calculate the round-trip ΔVs required to move between destinations in LEO and between LEO and destinations elsewhere in the Earth-Moon system. Figure 4 depicts the Earth-Moon system with various locations of likely interest, including geostationary orbit (GEO), Earth-Moon Lagrangian Points (EML-1/2/4/5), Global Positioning Satellite (GPS) orbits, and low lunar orbit (LLO).

When operating from the LEO space logistics depots, a space ferry design ΔV of 8,550 mps (28,000 ft per sec) would enable most destinations of interest to be reached with the vehicle's design payload. The performance envelopes for space ferries with this level of performance, based at 28.5° and 51.6°, are shown in Figure 4. The overlap in the performance envelopes indicates some measure of redundancy in reaching destinations in higher Earth orbits.

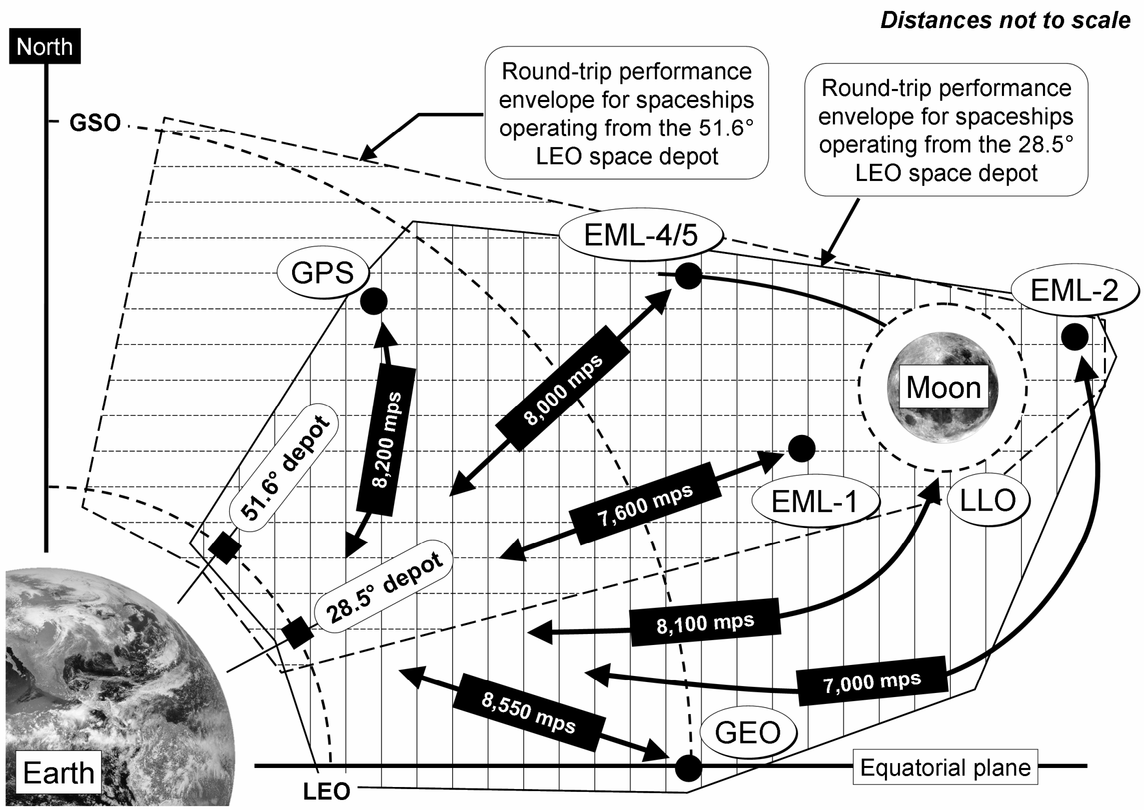

Figure 4. Performance envelopes, within the Earth-Moon system, for LEO-based space ferries.

Section 3: Space Tug

The first permanently-space-based reusable space logistics vehicle will be a space tug. It will be primarily a short-distance spaceship used to transport materiel and passengers and support on-orbit assembly operations, e.g., positioning components during assembly. The tug will also be used for space rescue and recovery operations and for retrieving cargo modules once they are delivered by the aerospaceplanes.

Space tug design objectives

The following objectives were used to develop the tug's conceptual design:

1. Fully-assembled and appropriately fueled to self-deploy to the altitude of the space logistics depot, the tug must fit within the aerospaceplane's external cargo container envelope of 3.7m (12 ft) diameter by 9.1 m (30 ft) long. When dropped off at 51.6° and 185 km circular orbit, the fueled mass must be less that the 15,400 kg (34,000 lb) net payload limit of the reusable aerospaceplane cited in References 1 and 2. When dropped off directly at the 497 km altitude of the 51.6° depot, the mass

of the system must be less than the 11,400 kg (25,000 lb) net payload limit of the aerospaceplane.

2. The tug must be able to transport a "crew module" capable of accommodating three people for four days and have the ability to operate autonomously in space.

3. With the crew module attached, the SLV tug must be able to depart the LEO space logistics depot and retrieve a 20,400 kg (45,000 lb) payload from a 185 km circular orbit and return to the LEO space logistics depot. This is a space search and rescue mission where a passenger aerospaceplane stranded at 185 km is returned to the space logistics depot.

Space tug conceptual design description

The conceptual design of a space tug, responsive to these criteria, is shown in Figure 5. This is a fully-reusable spaceship incorporating a crew module and a propulsion module. The empty mass is approximately 10,400 kg (22,900 lb). The overall length of the system is 9.1 m (30 ft) and the maximum diameter is 3.4 m (11.2 ft). The gross mass, equipped to perform the space search and rescue mission, is approximately 15,500 kg (34,100 lb), including 4,400 kg (9,700 lb) of propellants. This will provide approximately 147 mps (480 ft per sec) ΔV for maneuvering and 410 mps (1,360 ft per sec) ΔV for orbit transfer to conduct the passenger aerospaceplane retrieval mission. When delivered by the aerospaceplane at 185 km fully fueled, the space tug will arrive at the LEO space logistics depot with approximately 3,000 kg of propellant remaining.

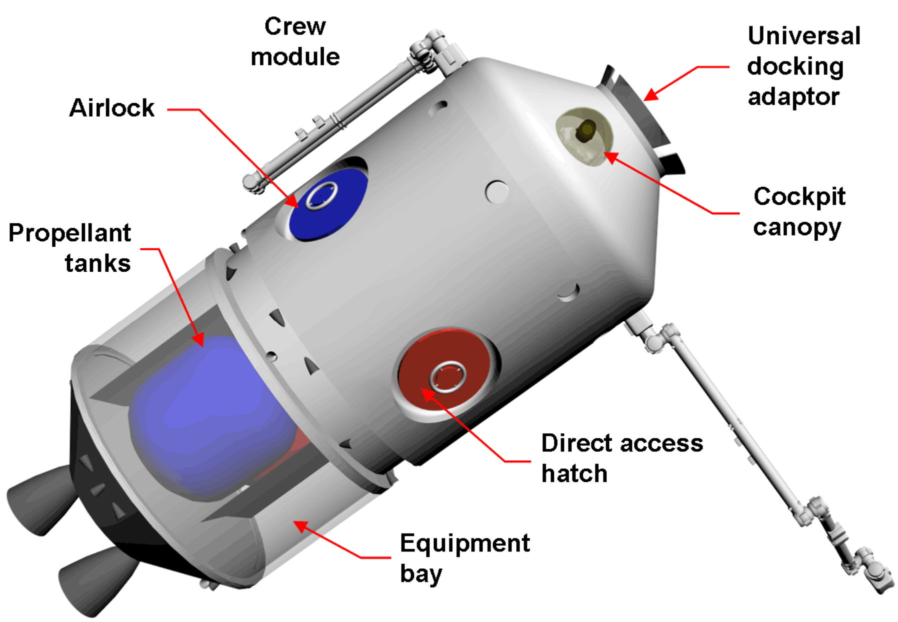

Figure 5. Conceptual sketch of the Space Tug.

- Crew module conceptual design

The conceptual design of the crew module was developed by parametrically defining the physical size and mass of the module as a function of six primary inputs: number of crew members, mission length, maximum module radius, volume allocated per crew member, and the uncertainty margin applied to mass, volume, power required, and waste heat generated. Utilizing the number of crew members and mission length as input parameter, the internal crew module system mass, volume, power, and waste heat were first estimated. (Note: The individual component sizing equations were taken from publicly available sources such as Ref. 3.) The major internal system elements were: water system, atmospheric control, crew and crew items including two space suits, galley and food, waste collection system, fire detection and suppression, personal hygiene, thermal control, and miscellaneous, which included the airlock. (Note: The crew module is not designed for operation without the propulsion module. Hence, estimates of the module's internal system mass for elements such as thermal control and electrical power only reflect those parts of the system located within the crew module.)

Using these sizing relationships, an estimate of the required internal volume of the crew module was made. With this as the input, the outer structure mass of the module was then estimated. If the required module spherical volume was larger than that permitted by the constraint of the maximum permitted module radius, the module pressure shell was modeled as a cylindrical shell with ellipsoidal end caps. The module was assumed to be fabricated from weldable aluminum. The first estimate of the pressure shell mass was made from structural calculations of the wall thickness designed by pressure loads. This was then compared with an estimate of the required average wall thickness of the pressure shell based on a radiation protection requirement of a combined 1.5 g/cm^2 for the pressure shell, multi-layer insulation, and micrometeoroid Whipple shield. (Note: A typical design criteria for human-occupied space systems is to have 1-2 g/cm^2 of aluminum shielding.[4]) The residual pressure wall thickness required to meet the total radiation protection requirement exceeded the wall thickness defined by pressure loads, even with a conservative safety factor. As a result, radiation protection rather than pressure loads established the pressure shell mass. The total pressure shell structural mass was further increased by 40 percent to account for the internal substructure (e.g., frames and stiffeners). In addition to this primary structural mass estimate, individual mass estimates of significant items, i.e., bubble canopies, were included in the total crew module mass estimate.

The final crew module was sized for three people for four days. The maximum diameter was 3.4 m (11.2 ft). The volume per crew member was 2.5 m^3. (Note: This is about 20 percent greater than the Apollo command module's per person volume.) The uncertainty factor was 15 percent. Based on these inputs, the pressure shell has a diameter of 3.2 m. The overall length, including the universal docking adapter, is 4.4 m (14.5 ft). The module contains 24 m^3 (850 ft^3) of gross internal volume. This includes a 15 percent gross volume margin and a 15 percent individual item packaging margin. The pressure shell is cylindrical with ellipsoidal end caps. The

Item	Mass (kg)	Percentage
Shell Structure & Radiation Shield	297.6	8.76%
MDPS Bumper & MLI	316.5	9.32%
2 Windows	150.0	4.42%
Structural Frames	119.0	3.51%
2 Hatches	90.7	2.67%
1 Universal Docking Interface	200.0	5.89%
Water System	138.9	4.09%
Atmosphere Control	272.3	8.02%
Crew and Crew Items	615.9	18.14%
Galley and Food	50.1	1.48%
Waste Collection System	48.4	1.42%
Fire Detection and Suppression	35.1	1.03%
Personal Hygiene	14.3	0.42%
Thermal Control System	158.8	4.68%
Miscellaneous (including airlock)	683.7	20.14%
Cabling (2.4% of above items)	76.6	2.26%
Crew module-structural interfaces	81.7	2.41%
Crew module - propulsion module interfaces	45.4	1.34%
Intermediate Sum	**3394.9**	100.0%
Growth weight	509.2	15.0%
Total	**3904.1**	-

Table 1. Mass breakdown of the space tug's crew module.

mission-ready mass of the crew module, excluding the robotic arms, is approximately 3,900 kg (8,600 lb), including a general 15 percent overall mass growth margin. A breakdown of the crew module mass is shown in Table 1.

The crew module is sized to enable a crew of two to perform a broad range of logistics support missions, including extravehicular activity (EVA), over a period of several days. The module is designed for three people so that a pilot remains inside in control of the spacecraft when two people are conducting EVA. External access to the crew module, from another pressurized habitat, is through the universal docking adapter. For EVA, an internal, single-person, airlock permits the crew to exit the crew module, after the space suits have been donned, without the need to depressurize the crew module. EVA operations will, however, primarily be intended for emergency operations such as rescue missions and "glitch" repairs where direct human interaction with faulty/balky equipment is necessary. A second hatch is provided to enable direct access to the interior of the crew module while conducting maintenance in the pressurized space hangar of the space logistics base.

In terms of power and thermal loads for the design mission, the crew module has an average power load of 2.5 kW and a total mission energy requirement of 240 kW-hr. The average sensible heat load is 3 kW. These values include the 15 percent uncertainty margin. (Note: When the robotic arms are attached to the crew module, as shown in Figure 5, these are powered by separate power lines from the propulsion module and are not interfaced with the crew module power system. These power demands are not included in the module design loads.)

- Propulsion module conceptual design

The propulsion module provides primary and maneuvering thrust and houses the primary avionics, electrical power generation, life support and thermal control. The propulsion module has a universal docking adapter and payload latching/unlocking system at its forward end. This is be used to attach the crew module or to attach a payload when the space tug is operated remotely without a crew module. This interface also includes standard interfaces for the crew module and payload that interconnect power, gases, fluids, thermal control, and data systems.

The propulsion module also has a maximum diameter of 3.4 m (11.2 ft) with a length (including engine) of 4.7 m (15.4 ft). As with the crew module, this provides a 12 cm clearance around the spacecraft for its installation into the aerospaceplane's cargo module. The propulsion module's empty mass is approximately 7,150 kg (15,800 lb). A mass breakdown of the conceptual design of the propulsion module is shown in Table 2.

The propulsion module uses methane and oxygen as propellants. Both are mild-cryogenic fluids that enable easier on-orbit propellant handling and management. Also, these propellants enable self-pressurization of the propulsion system and are also used for the reaction control systems. This reduces the number of propulsion fluids to two by eliminating the need to handle separate tank pressurization gases, such as helium or nitrogen, as well as separate propellants for

Item	Mass (kg)	Percentage
RCS thrusters (dry)	217.7	3.5%
Avionics	163.9	2.6%
Universal docking adapter	100.0	1.6%
2 robotic arms (sized for 20,000 kg)	653.2	10.5%
Active thermal control	150.1	2.4%
Wiring	388.8	6.3%
Power and distribution	3115.4	50.1%
Propellant system (tanks, pumps, etc.)	233.1	3.7%
Secondary structure	795.7	12.8%
2 engines with thrust vectoring	402.8	6.5%
Intermediate sum	**6220.6**	100.0%
Growth margin	933.1	15.0%
Total	**7153.7**	

Table 2. Mass breakdown of the space tug's propulsion module.

the reaction control system. These propellant choices also enable the entire propulsion system to be purged by opening the system to the vacuum of space. This would be used to safe the propulsion module before being brought into a LEO space logistics base's space hangar for maintenance.

The propulsion module includes provisions for mounting two medium-capacity robotic arms to support materiel handling. When the crew module is not attached, these arms will attach to the forward bulkhead of the propulsion module. The mass of the arms—each sized to handle the 20,000 kg retrieval mass of a passenger aerospaceplane or a fully-loaded cargo module—is included in the propulsion system mass.

Lithium-ion batteries are used for primary power in the space tug to eliminate on-orbit servicing of either a fuel cell system or an externally-deployed solar array system. As a consequence, battery mass (at 120 Wh/kg) becomes a significant percentage of the total system empty mass. The batteries will be charged while the space tug is docked to the space construction station, as seen in Figure 3, or the space logistics base. Because of the use of a large number of batteries and the use of robotic arms, the wiring mass has been increased. (Note: The ideal performance of the space tug can be improved significantly by replacing the heavy batteries with lighter fuel-cells. This modification could be undertaken once the depot's space logistics base becomes operational enabling maintenance to be performed on the space tug's fuel cells.)

The combination of the low self-discharge rate of the lithium-ion batteries and the low boil-off rate of the methane-oxygen propellants will enable the space tug to be parked for extended periods of time at the space logistics depot while being immediately available for operation. This is important for the space tug's use in conducting emergency space search and rescue operations.

Active thermal management, for internally-generated waste heat, is incorporated into the propulsion module to collect and discharge heat from the avionics, reaction control system, crew module, and electrical power generation and distribution. Waste heat, collected through cold plates and air-to-water heat exchangers, is eliminated through external radiators. The average heat load is approximately 5.2 kW.

The active thermal management system, reaction control system, power distribution system, wiring, fuel controls, and main engines are fully redundant. The avionics are triply redundant.

Shielding and insulation

Both modules include externally-mounted micrometeoroid and space debris shields and multi-layer insulation to control solar heating. Micrometeoroid/debris shielding is provided through a stand-off-mounted aluminum shell, referred to as a bumper, that serves to cause impacting micrometeoroids and debris to beak-up and vaporize. Multi-layer insulation, located under the bumper to control external heating from solar radiation, assists in limiting penetration of small micrometeoroids and debris.

For the crew module, the combination of the bumper, multi-layer insulation, pressure shell, and internal equipment provides radiation protection for the crew suitable for missions in LEO. Based on the equivalent spherical shell area of 40 m^2, the average radiation shielding for the crew within the crew module is approximately 10 g/cm^2. While adequate for missions in LEO, for missions above LEO additional radiation protection measures may need to be incorporated to provide a "storm cellar" against sudden solar storms.

Ideal space tug performance

Figure 6 provides an estimate of the achievable ΔV for the space tug. The zero payload mass corresponds to the mass of the propulsion module plus five percent of the total propellant mass assumed to be unavailable due to boil-off, ullage, and trapped/unusable fuels. The range of the payload mass charted—from 0 to 100 metric tons—represents the full range of likely payload mass, with the upper bound representing a space construction station or large space facilities module delivered to orbit for assemble.

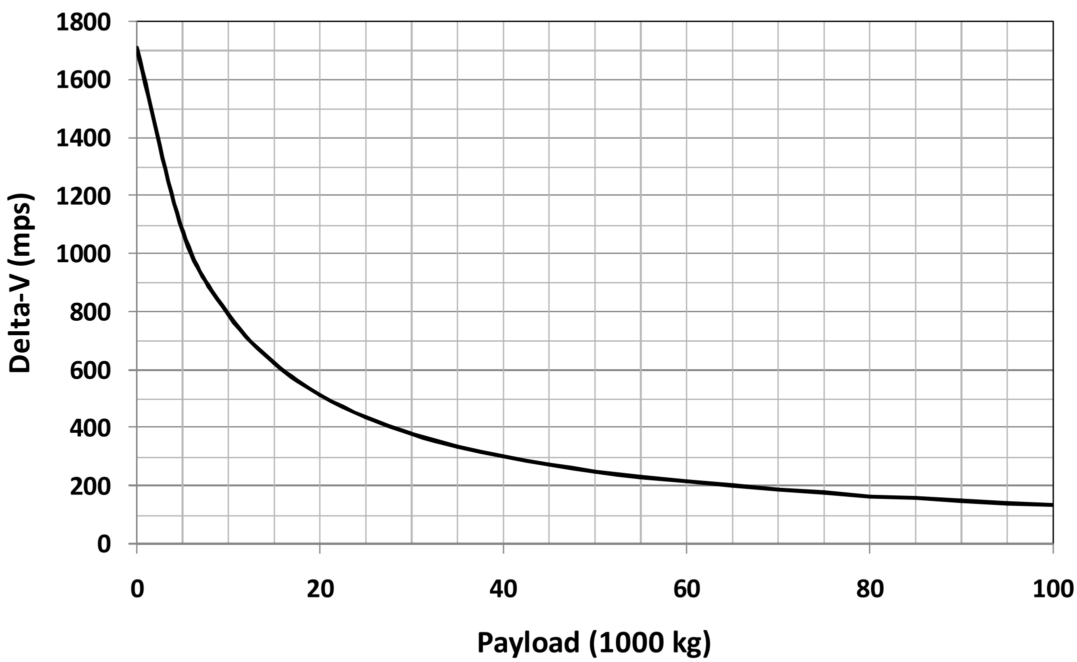

Figure 6. Space tug propulsion module (without crew module or robotic arms) ideal main propulsion ΔV (mps) based on using 95% of the available propellant.

Section 4: Space Ferry

Once improved space access to LEO and localized space operations are established at the space logistics depots, the next phase of expanding the spacefaring infrastructure will focus on extending routine, “aircraft-like” space access throughout the Earth-Moon system, as seen in Figure 4. The initial spaceship for this mission is referred to as a space ferry. It will be capable of transporting passengers, cargo, and payloads throughout the Earth-Moon system.

Propellant selection

A vehicle’s mass may be divided into three fractions— payload, hardware, and propellant—as seen in Eqn. 1.

Eqn. 1 $$M_{initial} = (Fraction_{payload} + Fraction_{hardware} + Fraction_{propellant}) \cdot M_{initial}$$

Eqn. 2 defines the propellant fraction as a function of the initial and final mass and relates this to the ΔV and rocket engine Isp or exhaust velocity.

Eqn. 2 $$Fraction_{propellant} = 1 - \frac{M_{final}}{M_{initial}} = 1 - \exp\left(\frac{-\Delta V}{g_c \cdot I_{sp}}\right) = 1 - \exp\left(\frac{-\Delta V}{v_{exhaust}}\right)$$

Figure 7 charts Eqn. 2. Four values of specific impulse are represented: Methane-oxygen at 360 secs and a range of hydrogen-oxygen at 462, 485, and 499 secs (representing different nozzle expansion ratios). The vertical line represents the needed 8,550 mps required for mobility within the Earth-Moon system, as discussed earlier.

Design studies indicate that propellant fractions much above approximately 85 percent are not readily achievable for lower density fuels like methane and

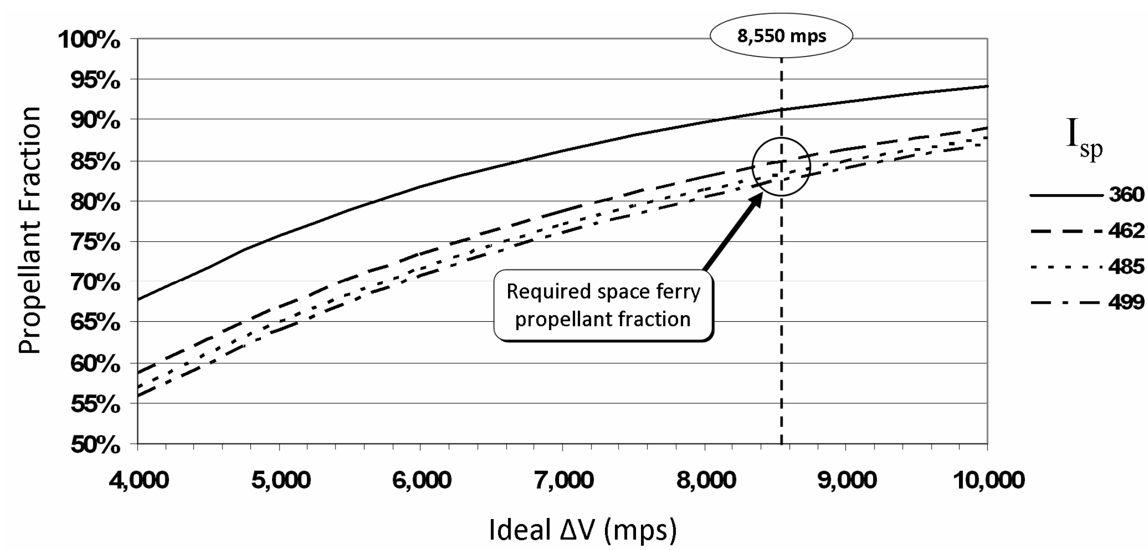

Figure 7: Propellant fraction required as function of Delta V and I_{sp}.

hydrogen. Using this as a cut-off, a methane-oxygen fueled space ferry would lack the performance needed for mobility within the Earth-Moon system. However, as seen in Figure 7, a hydrogen-oxygen engine does provide the needed performance at achievable propellant fractions.

Boeing Orbit Transfer Vehicle

In the late 1970s, NASA began investigating space-based reusable systems to provide in-space mobility. One such study was performed by Boeing, in 1982, as the second part of an orbit transfer vehicle (OTV) conceptual design study.[5] It focused on reusable OTVs responding to the following criteria:

- 1990 technology availability (indicating this, today, is a near-term design)
- OTV design life – 45 missions
- OTV mission success goal – 0.97
- Space debris probability of not impacting the OTV propellant tanks – 0.995
- Manned OTV to incorporate two engines
- OTV reference missions from 28.5 deg at 370 km (200 nm) to GEO and return carrying, one way, 7,600 kg (16,760 lb) up and 5,000 kg (11,000 lb) down

The general arrangement of the hydrogen-oxygen, space-based OTV is shown in Figure 8 (a reproduction of Figure 3.3.3-2 from the cited report). This OTV was sized to fit within the Space Shuttle orbiter's payload bay and is 14.17 m (46.5 ft) in

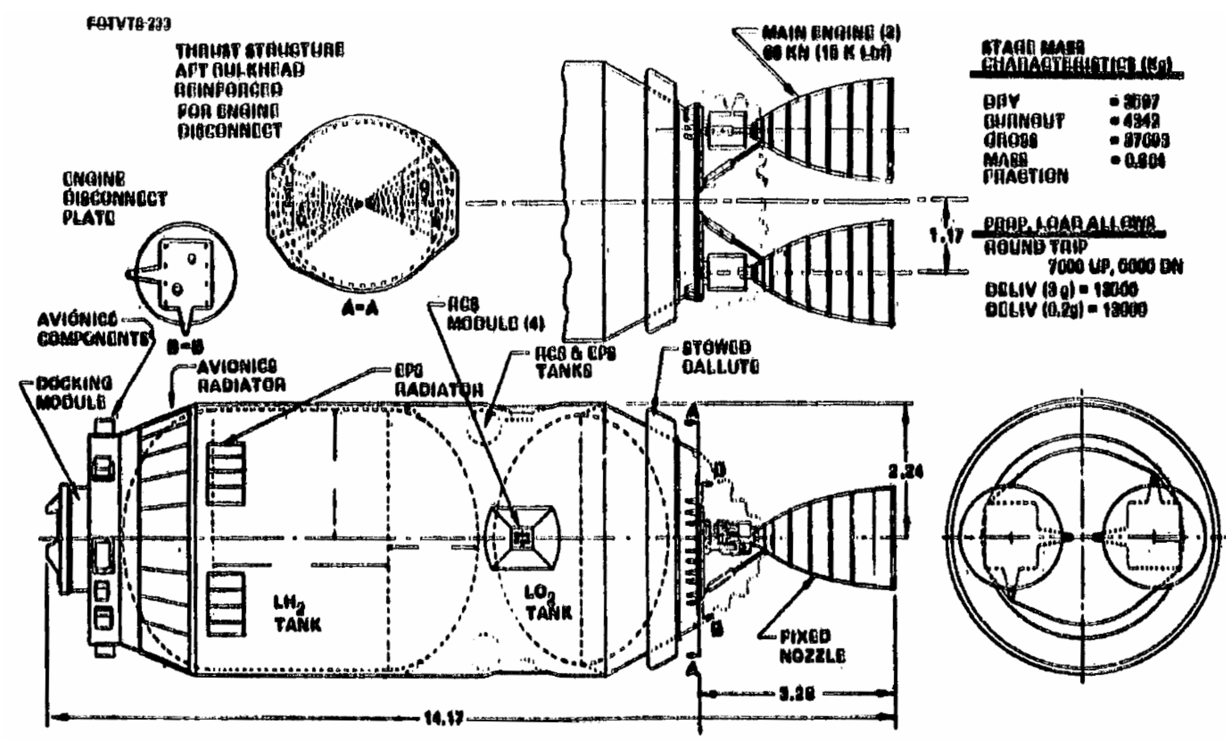

Figure 8. Boeing space-based OTV (1982); dimensions in meters.

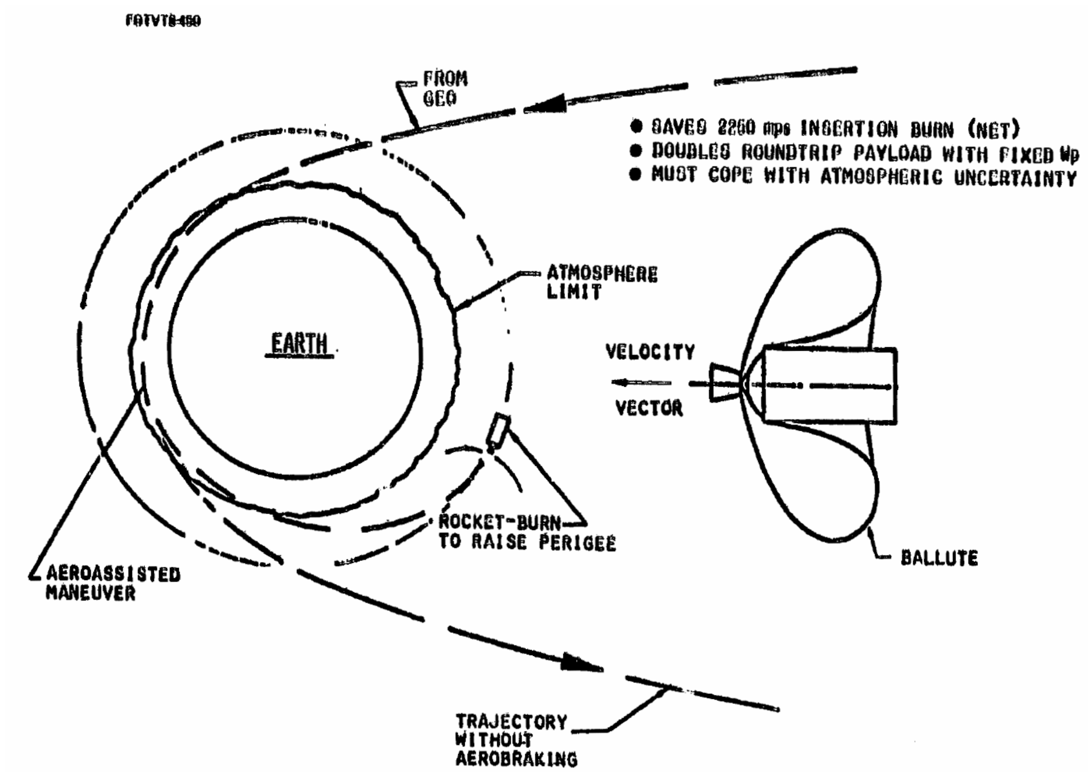

Figure 9. OTV Aerobraking Maneuver.

length and 4.48 m (14.7 ft) in diameter. It has a "dry" mass of 3,697 kg (8,150 lb) and a fueled mass, less payload, of 37,693 kg (83,100 lb) with 32,289 kg (71,200 lb) of main propellant. Using the relationship for propellant fraction discussed earlier, this OTV's propellant faction (without payload) is 32,289 / 37,693 = 0.86.

This OTV uses an "advanced" 1990's-technology reusable engine with a high expansion ratio nozzle with a specific impulse of 485 sec. When not carrying any payload, this vehicle has an ideal maximum ΔV of approximately 9,240 mps.

One important feature of this Boeing OTV was the use of a ballute to enable the spacecraft to use the Earth's atmosphere for braking on return from a higher orbit. For the return from GEO, aerobraking enables approximately 2,250 mps (7,400 fps) of velocity to be shed through atmospheric drag rather than by using the rocket engines. The roundtrip mission ΔV, as noted in Figure 9, reduces from approximately 8,550 mps to approximately 6,400 mps. This enables the roundtrip payload to significantly increase.

In the Boeing design, the inflatable ballute is attached near the end of the LOX propellant tank. On approach to the Earth's atmosphere, the ballute is inflated to provide a large drag brake. The ballute is porous, permitting the inflation gas to pass through the ballute bag and provide a cool boundary layer helping to protect the bag from excess aeroheating. In addition, the engines are run at idle and the relatively cool expanded exhaust gases exiting the nozzle also provide additional thermal protection of the engine area and bag. Once the aerobraking maneuver is complete with the Boeing design, the bag is released and the OTV completes the LEO orbital maneuvers to bring it back to the orbiting base.

Space ferry design missions

Using a modified version of the space tug conceptual design methodology described earlier, three point-designs for space ferries similar to the Boeing OTV were defined. Their basic mission capabilities, to enable logistics support operations within the Earth-Moon system, would be:

Mission 1: Conduct an 8,550 mps ΔV roundtrip mission, *without aerobraking*, to perform an unmanned maintenance support mission to refuel a satellite in GEO or replace satellite components such as a solar array.

Mission 2: Conduct a 6,400 mps ΔV roundtrip mission to a high Earth orbit to perform a manned space logistics servicing mission and then use aerobraking on the return to LEO.

Mission 3. Conduct a 6,400 mps ΔV roundtrip mission to a high Earth orbit to deliver a satellite and then use aerobraking on the return to LEO without a payload.

Design Mission ΔV's

The mission segment ΔV's used to establish the required performance for the space ferry are shown in Table 3. These values were taken from Table 3.3.5-1 of

	Without Aerobraking		***With Aerobraking***	
Mission Segment	***RCS (mps)***	***Main Engines (mps)***	***RCS (mps)***	***Main Engines (mps)***
Separation from the LEO Space Logistics Base	3	0	3	0
Phasing orbit injection	0	1370	0	1370
Transfer orbit injection	0	1098	0	1098
Mid-course correction	0	18	0	18
GEO circularization	0	1771	0	1771
GEO trim	0	9	0	9
GEO rendezvous	21	0	21	0
GEO departure transfer orbit injection	0	1846	0	1846
Mid-course correction burn	20	0	20	0
LEO entry maneuver	0	2213	67	61
LEO circularization	0	122	0	122
Rendezvous with the LEO space logistics depot	18	0	18	0
Maneuver to the LEO space logistics base	18	0	18	0
Reserves	30	107	30	107
Total	**111**	**8553**	**178**	**6402**

Table 3. Mission segment ΔV's used to size the space ferry.

Reference 5. One change to the source information was the addition of a main engine burn, at an idle setting, during part of the LEO entry maneuver when using aerobraking. This was to provide main engine exhaust flow to aid in cooling the ballute. As seen in Table 3, the use of aerobraking reduces the total mission main engine ΔV by approximately 2,150 mps. The total main engine ΔV reflects the design of the space ferry, like the OTV, for a maximum initial acceleration of 0.2g, as may be needed with g-sensitive payloads.

An important assumption in these conceptual design studies was that the ballute configuration used in the Boeing OTV study is applicable to the space ferry configurations developed in this paper. Of particular note, is the fact that the Mission 2 configuration, which includes a crew module, will see a significant center of gravity shift forward during the return to LEO segment where aerobraking is used. However, the crew module mass is comparable to the "down" payload mass for the OTV, indicating that the forward center of gravity location may be acceptable.

Space ferry conceptual design

The conceptual design of the space ferry (as seen in Figure 10), while generally similar to the Boeing OTV, has the following differences. The space ferry:

- Includes the ability to carry two robotic arms to provide mission assistance such as manipulating payloads.
- Incorporates increased electrical power and thermal management capabilities to support the crew module, robotic arms, and payloads requiring electrical power and thermal management, as appropriate for the mission.
- Includes side vehicle equipment and avionics bays, similar to those used on the space tug, to permit the crew module to be attached to the top of the space ferry using a universal docking adapter.
- Sizes the maximum propulsion module diameter at 4.57 m (15.0 ft) to reflect the fact that the space ferry is to be able to enter the space hangar at the LEO space logistics base. This reflects a tank diameter of 4.37 m (14.3 ft) and a micrometeoroid/space debris bumper height of 10.2 cm (4 in).
- Includes a fairing around the top of the engine installation for increased micrometeoroid and space debris protection.
- Includes a crew module for manned missions.

Enhanced crew module conceptual design

Mission 2 includes the use of a three-person crew module. Using the same design methodology as described previously, an enlarged crew module suitable for a longer mission duration of eight days was sized. The per person volume was increased to 4 m^3 from the 2.5 m^3 used in the smaller module. This change in volume, combined with the longer mission duration and the inclusion of approximately 1,200 kg of water for additional radiation shielding (to provide an

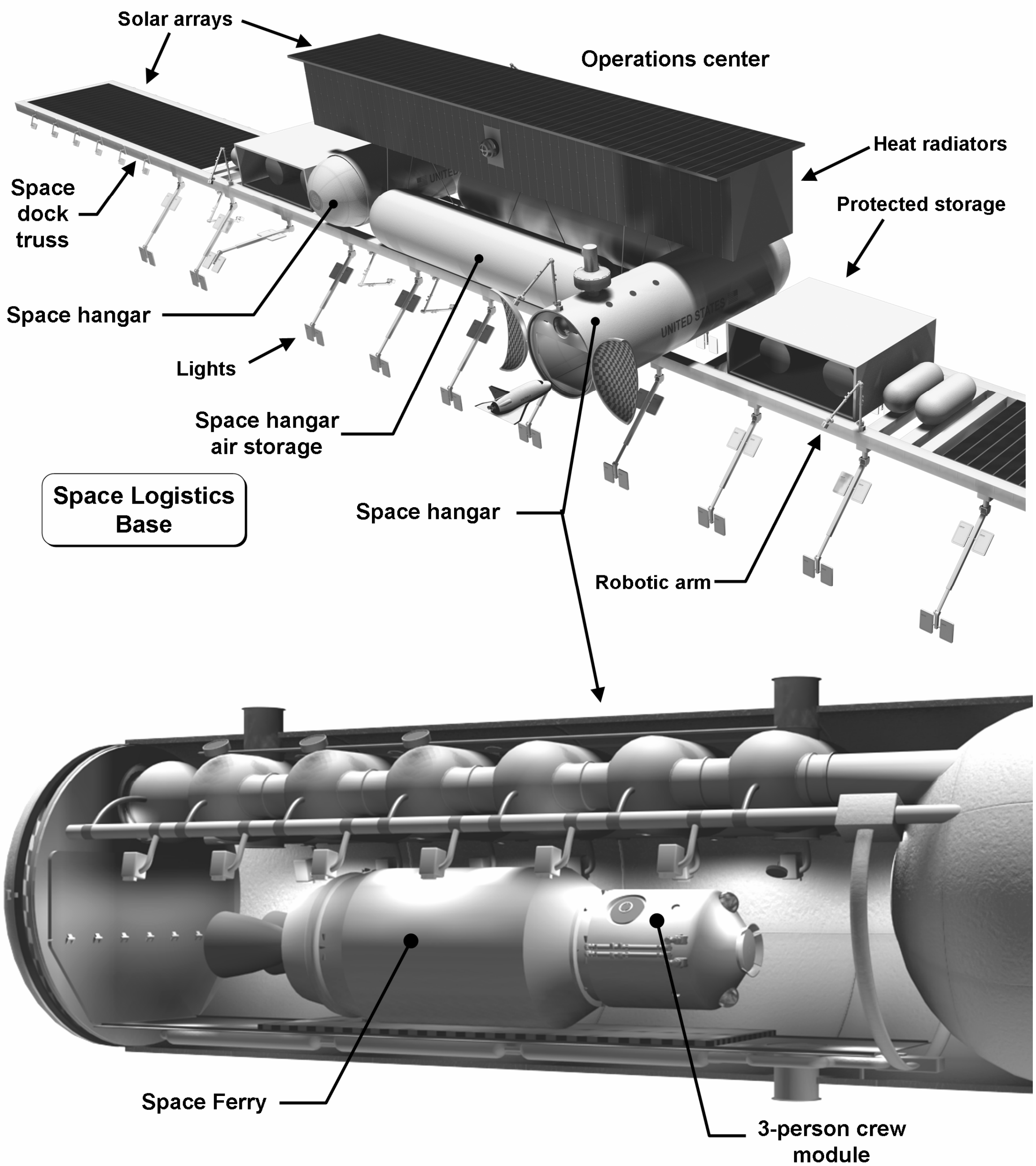

Figure 10. Space logistics base (above) and space ferry in space logistics base's space hangar (below).

effective 20 g/cm^2 radiation shelter), increased the module mass to 5,650 kg and the length to 5.4 m. These values reflect the same margins as discussed previously. (Note: The upsized crew module size and mass still enable it to be transported to LEO using the aerospaceplane's standard cargo module.)

Propulsion module conceptual design

The space tug propulsion module sizing methodology was modified to change the fuel to hydrogen from methane. The primary electrical power system was changed to fuel cells while retaining a five percent battery power reserve. The robotic arms were sized for each handling 5,000 kg. (Note: Larger robotic arms with greater handling capacity can still be used with the additional mass being charged against the allowable mission payload.) The main propulsion engines were based on the Pratt & Whitney RL60 hydrogen-oxygen rocket engine scaled to the needed thrust level with an assumed specific impulse of 485 secs. (Note: Minor main engine propulsion burns are used with an assumed idle specific impulse of 440 secs.)

Estimated mission performance and system mass estimates

Mission 2, carrying the payload of the crew module, two robotic arms, and a mission kit of 1,000 kg, was found to be the sizing mission. This yielded a space ferry with a mission gross weight of 57,900 kg (127,700 lb) and an empty mass (including crew module, robotic arms, and mission kit) of 13,350 kg (29,400 lb). The propulsion module empty mass was 6,400 kg (14,100 lb). The sized space ferry propulsion module contains 44,600 kg (98,300 lb) of total propellants. The propellant fraction was 0.77 and the payload fraction was 0.12. The propulsion module is 14.1 m (46.4 ft) in length while the combination propulsion module and crew module is 19.5 m (64 ft) in length. Each robotic arm has a mass of 150 kg (330 lb). A fifteen percent uncertainty margin was applied to the dry mass of the propulsion module.

With the size of the space ferry established, its performance for the other two missions could be established. For Mission 1—an unmanned space servicing mission with robotic arms, but without the use of aerobraking—the space ferry would be able to carry a mission mass of approximately 1,200 kg (2,650 lb). The mission gross mass was 51,900 kg (114,500 lb) and the empty mass was 6,200 kg (13,600 lb). The propellant fraction was 0.859 and the payload fraction was 0.028. For Mission 3—an unmanned payload delivery mission without robotic arms, but with aerobraking—the one-way delivered payload mass was 14,400 kg (31,750 lb). The mission gross mass was 65,400 kg (144,100 lb) and the empty mass was 6,400 kg (14,100 lb). The propellant fraction was 0.682 and the payload fraction was 0.22.

From Figure 4, the roundtrip mission from LEO@28.5° to/from GEO has the largest mission ΔV. With the space ferry sized for this mission, its performance for the other missions represented in this figure can be computed. The first mission of interest is the extension of Mission 3 to delivery cargo to the Earth-Moon L-1 point, perhaps to support a permanent human presence on the Moon. Adjusting the mission ΔV, including the use of aerobraking, yields an increase in the delivered payload to approximately 20,500 kg (45,200 lb). The second mission of interest is the extension of Mission 2 to perform a manned servicing mission to a scientific platform located at the Earth-Moon L-2 point. The servicing mission kit mass—in addition to the crew module and robotic arms—is 9,500 kg (20,950 lb).

Section 5: Conclusion

While OTV's have been discussed in the aerospace literature for many decades, the purpose of this paper has been to refresh this discussion by developing two conceptual designs of near-term spaceships that will be useful in establishing routine space operations throughout the Earth-Moon system. The results of these conceptual design analyses indicate that such spaceships can be developed using current technologies. Further, their operational deployment, especially that of space ferry, will provide substantial improvement in mobility and logistics servicing support throughout the Earth-Moon system.

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