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Conceptual Design of a Lunar Shuttle Transport Vehicle

Matthew David Finney
University of Tennessee - Knoxville

To the Graduate Council:

I am submitting herewith a thesis written by Matthew David Finney entitled "Conceptual Design of a Lunar Shuttle Transport Vehicle." I have examined the final electronic copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Master of Science, with a major in Aviation Systems.

Dr. Ralph Kimberlin, Major Professor

We have read this thesis and recommend its acceptance:

Dr. U. Peter Solies, Dr. Alfonso Pujol

Accepted for the Council:
Carolyn R. Hodges

Vice Provost and Dean of the Graduate School

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Anne Mayhew
Vice Provost and
Dean of Graduate Studies

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CONCEPTUAL DESIGN OF A LUNAR SHUTTLE TRANSPORT VEHICLE

A Thesis
Presented for the
Master of Science
Degree
The University of Tennessee, Knoxville

Matthew David Finney
May 2004

DEDICATION

This thesis is dedicated to my wife, Katherine C. Finney, whose support and friendship has been instrumental in my happiness over the last ten years.

ABSTRACT

In 1961 the former Soviet Union successfully launched the first human into space marking the beginning of the "Space Race" with the United States. Forty years later, the United States and Russia are working together in support of the International Space Station (ISS). The US Space Shuttle fleet and Russian Soyuz capsule and rockets are being used to replenish the ISS. In light of the latest shuttle accident and aging systems, NASA has been pursing alternatives to replace the shuttle fleet.

This study is a conceptual design of a spacecraft designed to meet the following requirements: 1. Transport a crew of eight from Kennedy Space Center in Florida to and from the International Space Station recovering at the Edwards Air Force Base complex in southern California, 2. Transport a crew of eight from the Kennedy Space Center to a future lunar base, and 3. Refuel at the future lunar base using propellant sources mined from moon, launch and return to earth.

The spacecraft system, Lunar Shuttle Transport (LST), was designed by tailoring the aircraft design methods presented in Raymer's, "Aircraft Design: A Conceptual Approach" (1999) to spacecraft design. A design method outline was developed to establish a roadmap for the vehicle design.

This study found that the desired configuration for the vehicle would be very similar in shape to the proposed lifting body designs of NASA's Assured Crew Return Vehicle and Orbital Space Plane. Unlike NASA's cancelled X-33 demonstration program, the LST system would not be a single stage to orbit design but rather would launch using a rocket system with multiple stages. The Lunar Shuttle Transport (LST)

would use aerodynamic braking to decelerate during reentry into earth's atmosphere and would rely on a parachute system and rocket engines for the final landing on skids.

For the lunar mission, the LST would use an additional stage for the translunar orbit insertion. The LST would rely on its main engines both for insertion into the low lunar orbit and the eventual landing on the moon. The launch from the moon would require that the LST be refueled by a source on the moon. The lunar launch and return trip to earth would be accomplished using the LST main engines.

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ABBREVIATIONS AND NOMENCLATURE

a	semi-major axis (orbital parameter)
a_t	semi-major axis of transfer orbit
A	aspect ratio
ACRV	Assured Crew Return Vehicle
AMS	attitude maneuvering system
C_D	coefficient of drag
C_L	coefficient of lift
e	eccentricity (orbital parameter)
E_t	energy of orbits
EVA	extra vehicular activity
g	gravity
g_o	gravity at earth's surface (9.81 m/s^2 or 32.174 ft/s^2)
GPS	Global Positioning System
H_2	hydrogen
hr	hour
i	inclination (orbital parameter)
I_{sp}	specific impulse
ISS	International Space Station
kg	kilogram
km	kilometer
kN	kilonewton
KSC	Kennedy Space Center
KTAS	knots true airspeed
lb(s)	pound(s)
m	meter
LEO	low earth orbit
LLO	low lunar orbit
LST	Lunar Shuttle Transport
LSTME	LST Main Engine
$Mass_{prop}$	propellant mass
mN	meganewton
MSL	mean sea level
NASA	National Aeronautics and Space Administration
N_2O_4 and MMH	Nitrogen tetroxide and mono-methyl hydrazine
NTO/MMH	Nitrogen tetroxide and mono-methyl hydrazine
O_2	oxygen
OSP	Orbital Space Plane
q	dynamic pressure
RP-1	hydrocarbon fuel (refined kerosene)
R	orbital radius
r	radius of parachute
s	second

SL	sea level
SSME	space shuttle main engine
SSTO	single stage to orbit
STD	standard day
TOF	time of flight
UTSI	University of Tennessee Space Institute
V_e	exhaust velocity
vac	vacuum
W	weight
W_o	ignition weight
W_f	burnout weight
W_{PL}	payload weight
ΔV	delta V (change in velocity)
α	angle of attack
Λ	wing sweep angle
ρ	density (sea level STD = 1.29 kg/m^3)
μ_E	gravitational parameter – earth
μ_M	gravitational parameter – moon

1. MISSION REQUIREMENTS

1.1 HISTORY

Since 1981 the United States has been relying on the reusable Space Shuttle Transport system for manned space flight. The Space Shuttle has been used to transport crews to and from space stations, deliver satellites into orbit, and as a space laboratory for experiments. The Shuttle was also designed and used to execute secret military missions.

In the mid 1980s it became evident that the shuttle system was very expensive as a satellite transport system. The US Space Shuttle fleet was plagued with delays and the integration effort and costs became excessive. Both the military and other organizations relied primarily on unmanned rockets for launching satellites. Since the establishment of the International Space Station (ISS), the Space Shuttle has been used primarily as a crew delivery system. The Russian rockets have been used primarily for re-supply of the station.

As technology progresses, it is likely that nearly all satellite and re-supply missions will be executed using unmanned rocket systems. Since the rocket systems do not have to be rated for manned flight, the integration time and effort is greatly reduced.

As the aging shuttle fleet nears retirement, the United States will have to move to a new platform for manned space flight. It is likely that the United States will require a new vehicle to transport crews to and from space stations. In fact, NASA has been pursuing an alternative to the space shuttle named the Orbital Space Plane (OSP). NASA has considered a vehicle that resembles a space shuttle as well as a capsule. However, the future of the OSP is uncertain.

It is conceivable that follow-on missions to the moon will occur in the first half of this century. Shortly after their historic first flight on October 15, 2003, the Chinese expressed a goal to establish a lunar base.

This study is a conceptual design of a Lunar Shuttle Transport (LST). The study was performed to develop a design that would fulfill both the space station crew transport mission and lunar shuttle mission. This study assumed that a future lunar base will be established prior to launching the vehicle to the moon.

1.2 GENERAL MISSION PROFILES

The vehicle was designed to fly the following profiles:

1. Space Station Profile
2. Lunar Profile
3. Lunar Return Profile

Space Station Profile
Launch from Kennedy Space Center (KSC), FL into a low earth orbit with an inclination of the International Space Station. Rendezvous and dock with ISS and deliver crew.

Return from the ISS and land in the Edwards Air Force Base complex in southern California.

Lunar Profile
Launch from Kennedy Space Center, FL into a low earth orbit. Transfer to a lunar transfer orbit. Enter a low lunar orbit and land on the moon. As a contingency, the spacecraft should also be able to bypass the lunar landing and return to earth.

Lunar Return Profile
Launch from the moon with the crew and return to the Edwards Air Force Base complex in southern California. It is assumed that a fuel source will be available on the moon in the future. The spacecraft should be able to refuel at the lunar base and return to earth.

1.3 SPECIFIC REQUIREMENTS

The specific requirements were:

- Eight crew
- Space station profile payload of 4,500 kg (not including crew)
- Lunar profile payload of 2,500 kg (not including crew)
- Lunar return profile payload – none
- Transport payload and crew to and from the International Space Station (ISS)
- Docking capability with the ISS
- Capability to recover in the Edwards Air Force Base complex
- Capability to fly and land on the moon
- Capability to fly lunar transfer orbit and return to Earth without lunar landing
- Return from the moon after refueling
- Solar radiation protection
- Reentry heat protection (thermal protection)
- Air lock system for space station docking/lunar base docking

The following were not requirements:

- EVA capability
- Satellite delivery/retrieval
- Capability to land and return from the moon without refueling

Illustrations of the three profiles are presented in figures 1, 2, and 3.

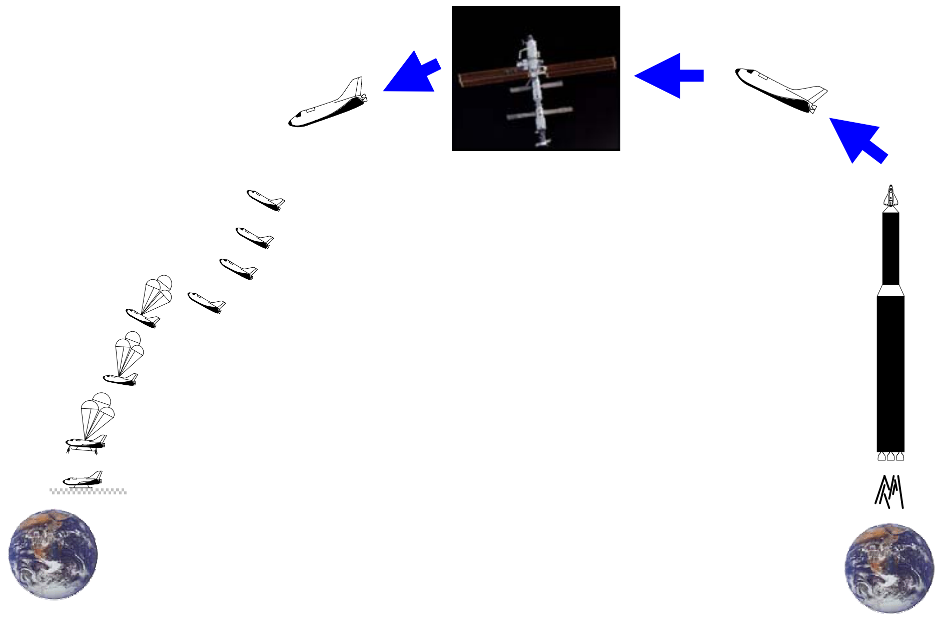

Figure 1: Space Station Profile Overview

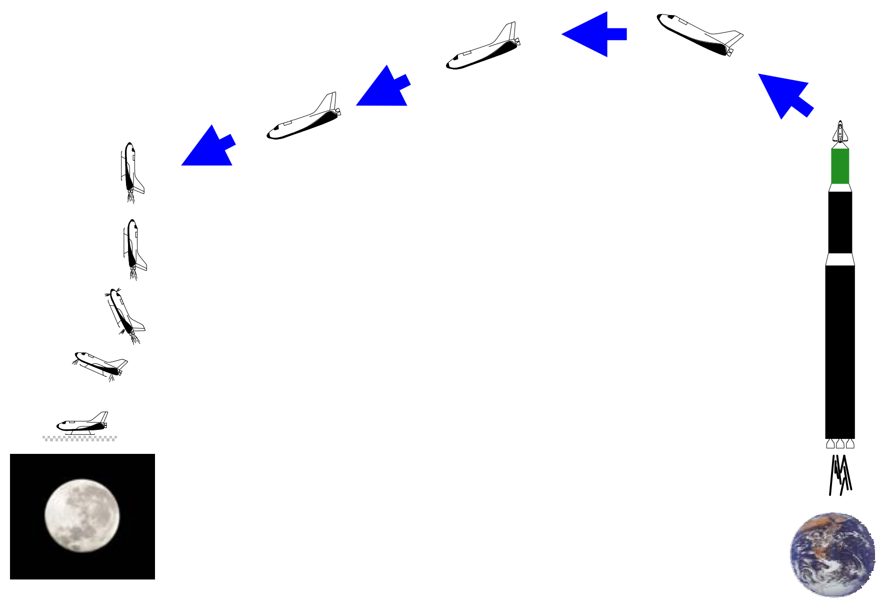

Figure 2: Lunar Profile Overview

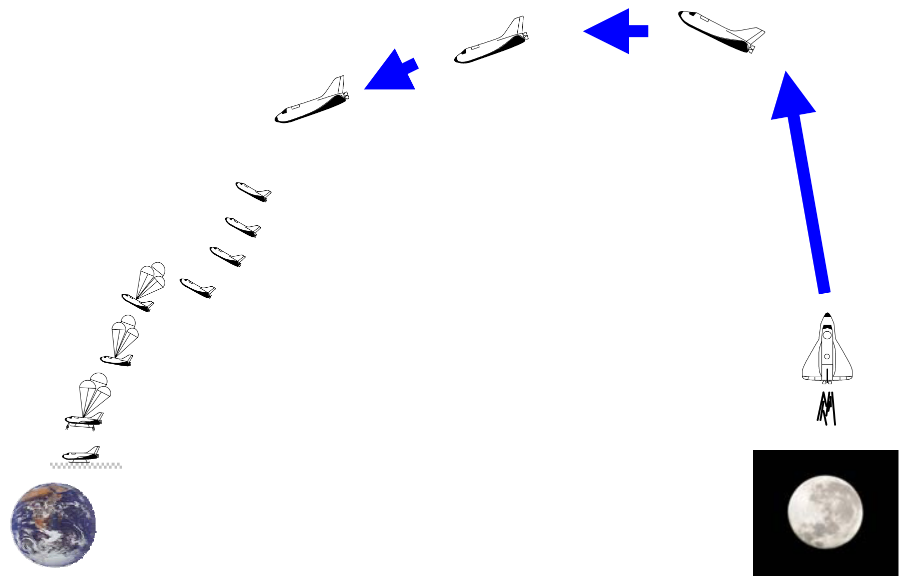

Figure 3: Lunar Return Profile Overview

2. DESIGN METHOD

A design method outline was developed using the following references:

- Space Mission Analysis and Design. (1992)
- Aircraft Design: A Conceptual Approach Third Edition. (1999)
- UTSI class AS506 Lecture Notes. Solies (2002)

The design method outline is presented in appendix C. The primary approach was very similar to the methods described in Aircraft Design: A Conceptual Approach. Existing technology and historical examples were used to estimate vehicle mass, fuel/propellant consumption, and other vehicle parameters. When possible, calculations were performed to more precisely determine specific vehicle parameters. The design sequence (method, results, and decisions) is presented in appendix D. Calculations are presented in appendix E.

3. DESIGN ASSUMPTIONS

The primary design assumed that a source of liquid hydrogen and oxygen would be available at a future lunar base. The LST would carry only enough fuel for one-way missions between the earth and moon. The LST would require the establishment of a lunar base prior to its arrival on the moon.

In December 1996 the US government announced that the spacecraft Clementine had made measurements indicating the presence of frozen water in a lunar crater located on the moon's south pole.[1] In 1998 the Lunar Prospector's neutron spectrometer detected hydrogen concentrations in the north and south poles. Preliminary analysis of the data found that approximately 260 metric tons of frozen water was present at the lunar poles.[2]

Missions to the moon would be required to establish a lunar base and means of mining the frozen water. Once the frozen water was mined, it could be split into hydrogen and oxygen using a nuclear generator or solar powered electric stations.[3] This would provide a source of fuel for the LST.

Orbital insertion velocity requirements were estimated using various sources. Detailed drag and gravity loss calculations were not performed. It was assumed that drag losses for the earth launch would be 3 percent of the total orbital insertion velocity. Gravity losses during the earth launch were estimated to be 1,300 meters per second.[4] The calculated propellant requirements for the earth launch were increased by 10 percent to account for the inefficiencies in the propulsion process and to provide an adequate margin to ensure orbital insertion. Additionally, it was assumed that the vehicle would launch to the east from Kennedy Space Center to take advantage of the earth's rotation.

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[1] Source: http://www.cmf.nrl.navy.mil/clementine/clementine.html
[2] Source: Moon-Based Advanced Reusable Transportation Architecture.
[3] Source: http://www.windows.ucar.edu/tour/link=/earth/moon/lunar_water.html&edu=high
[4] Source for drag and gravity loss assumptions: Space Mission Analysis and Design (p. 668).

4. DESCRIPTION OF DESIGN

The Lunar Shuttle Transport (LST) space vehicle is designed for crew delivery to the International Space Station and as a transport vehicle for missions to the moon. The LST carries a total of eight crew consisting of two pilots and six passengers. For lunar missions, the LST can carry internal payload of 2,500 kilograms. For space station missions, internal fuel is reduced and the internal payload capacity is increased to 4,500 kilograms.

The LST features a lifting body shape with a delta wing and vertical tail. Flaps and ailerons located on the trailing edge of the wing are used for attitude control during the reentry and landing evolutions. A rudder surface, used for yaw control during the earth-landing phase, is located on the vertical tail. Digital control flight computers control the control surfaces. Normal landings are accomplished in an automatic mode without pilot inputs. Flight controls are provided at the pilot stations for manual control should the autopilot system malfunction.

Four main engines, located on the aft end of the aircraft, provide thrust for orbital maneuvers, lunar landings, and lunar launches. The main engines use liquid hydrogen and oxygen as propellant and are capable of producing a total of 87.5 kN of thrust.

The LST is also equipped with an attitude maneuvering system (AMS) for attitude control in space and during landings. The AMS is a system of rocket thrusters using NTO/MMH for propellant. The AMS provides pitch, roll and yaw control while on orbit. The AMS jets also provide thrust to cushion the landing on the earth and moon.

General vehicle parameters are presented in table 1.

For landing on the earth, the LST is equipped with two landing skids and a parachute system consisting of a three parachutes. The LST relies on aerodynamic braking during reentry for the initial deceleration. As the LST slows to approximately 250 knots, the parachute system deploys to arrest the rate of descent. Prior to landing, the attitude maneuvering system jets fire to cushion the landing.

Table 1: General LST Parameters

Parameter	Value
Length	17.5 m
Wing span	11.0 m
Height	7.8 m
Space station profile on-orbit mass	29,800 kg
Space station profile payload	4,500 kg
Lunar profile on-orbit mass	30,240 kg
Lunar profile payload	2,500 kg
Lunar return profile launch mass	27,740 kg
Lunar return profile payload	0 kg

For lunar landings, the LST primary engines fires to control rate of descent as the vehicle approaches the surface in a vertical attitude. Prior to landing a coordinated rotation is performed using the AMS jets and the main engines. This allows the LST to land on the skids.

An airlock is located on the upper surface of the vehicle just aft of the pilot stations. The airlock is used for docking with the space station or a future lunar base.

The LST can be launched in three primary configurations. The first configuration consists of a two-stage rocket system used to insert the LST into a low earth orbit to rendezvous with the space station. The second configuration consists of a larger three-stage system that inserts the LST into a translunar orbit for the trip to the moon. The third launch configuration is the LST launching using only its main engines from the moon.

The vehicle structure is composed primarily of aluminum and advanced composite materials. Additionally, the surface is covered with tiles for heat protection during reentry. Vehicle and launch system configurations illustrations are presented in appendix A. Detailed vehicle parameters are listed in table B-1 in appendix B. A two-view illustration of the LST is presented in figure 4.

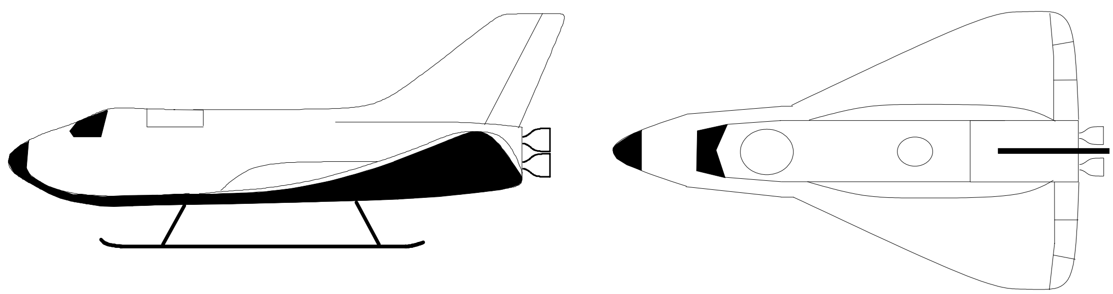

Figure 4: LST Line Drawing

5. PROPULSION SYSTEMS

5.1 LST MAIN ENGINES

The LST is equipped with four main engines located on the aft end of the vehicle. The engines use a liquid hydrogen and oxygen mixture for fuel rated at a specific impulse of 450 seconds. Each engine provides variable thrust up to 21.9 kilonewtons. The four engine nozzle angles can be varied to provide stability during orbital burns and lunar launches and landings. For the lunar mission, the LST carries 13,940 kg of propellant. For the space station mission, the internal fuel is reduced to 11,500 kg to provide additional space for payload.

For space station missions, the main engines are used to provide thrust for rendezvous and the initial phases of docking. After separation from the space station, the main engines provide thrust for the deorbit burn to reenter the atmosphere. The main engines are not used during the landing evolution.

For lunar missions the main engines provide the acceleration required for transfer from the translunar orbit to the low lunar orbit and the thrust required for the lunar landing. The final stage of the landing is accomplished using both the main engines and the AMS jets. During the descent to the lunar surface, the main engines provide a thrust-to-weight ratio of greater than 1.8.

For the lunar return mission, the main engines provide the thrust for launch from the moon, transfer to a return orbit, and the deorbit burn for reentry.

5.2 LST ATTITUDE MANUEVERING SYSTEM

The LST is equipped with an attitude maneuvering system or AMS. The AMS is a system of jets similar to the system used on the US Space Shuttle. The AMS jets use nitrogen tetroxide and mono-methyl hydrazine (NTO/MMH) for propellant. The LST carries 1,100 kg of NTO/MMH for both the lunar and station missions. The NTO/MMH has a specific impulse of 289 seconds. The AMS jets are located on the vehicle to provide attitude maintenance in all three axes. For lunar missions, the NTO/MMH is not refueled.

The AMS jets also provide attitude control and cushioning during lunar and earth landings. For earth landings, the AMS jets located on the lower surface and forward section of the LST provide thrust for cushioning and slowing forward velocity prior to impact. For lunar landings, the AMS performs coordinated burns with the main engines to rotate the LST to land on the skids.

The AMS system has two modes of control. The first mode is a fully automatic mode used for landing evolutions. The second mode is a manual mode. The pilot can manually control the AMS through use of hand controllers for maneuvering during evolutions such as docking with the ISS.

5.3 SPACE STATION PROFILE PROPULSION CONFIGURATION

For missions to the space station, the LST will launch using a two-stage boosting system. The two-stage system will insert the LST into a LEO with an inclination of 51.8 degrees in plane with the space station. The two-stage system can provide up to 11.2 km/s change in velocity (ΔV). Since insertion into the ISS plane orbit from KSC requires only 10.2 km/s, a 10 percent margin is designed into the two-stage system. An illustration of the configuration, figure A-2, is presented in appendix A.

Stage 1 will use RP-1, a liquid propellant, to provide the initial thrust on takeoff and during the ascent. Stage 1 is designed to provide a ΔV for stage 2 and the LST of 4.1 km/s. At burnout, the stage will separate and fall to the ocean.

Stage 2 will use liquid hydrogen as a propellant and will provide the final acceleration to LEO. Stage 2 is designed to provide a ΔV of 7.0 km/s for the LST. At burnout, stage 2 will separate and burn up during atmosphere reentry.

For station missions the LST is carries 11,500 kg of propellant (liquid oxygen and hydrogen). Fuel tanks are removed to allow space for additional internal payload. This process will take place during vehicle assembly prior to launch. Removal of the fuel tanks provides 7 m^3 of additional volume for the internal payload.

5.4 LUNAR PROFILE PROPULSION CONFIGURATION

The LST will be used for lunar missions after the establishment of a future lunar base. The future lunar base will provide a fueling station for the LST. Liquid hydrogen and oxygen mined on the moon will be used to refuel the LST after the lunar landing.

For the lunar mission, the LST will launch using a three-stage boosting system. Stages 1 and 2 will accelerate the LST and stage 3 into a LEO. Stage 3 will insert the LST into a translunar orbit. An illustration of the lunar configuration, figure A-3, is presented in appendix A.

The LST and stage 3 will separate once the LST is established in the translunar orbit. The LST will then use its main engines to enter a low lunar orbit and the eventual landing on the moon.

Stage 1 will use RP-1, a liquid propellant, to provide the initial thrust on takeoff and during the ascent. Stage 1 is designed to provide a ΔV for stage 2, 3 and the LST of 3.9 km/s. At burnout, stage 1 will separate and fall to the ocean.

Stage 2 will use liquid hydrogen as a propellant and will provide the final acceleration to LEO. Stage 2 is designed to provide a ΔV of 6.8 km/s for the stage 3 and the LST. At burnout, stage 2 will separate and burn up during atmosphere reentry.

Stage 3 will use liquid hydrogen as a propellant and will provide the acceleration into the translunar orbit. Stage 3 is designed to provide a ΔV of 3.1 km/s for the LST. At burnout, stage 3 will separate.

5.5 LUNAR RETURN PROFILE PROPULSION CONFIGURATION

The lunar return profile will launch under the power of only the main engines. No allowance for internal payload is provided. Launch mass is approximately 27,740 kilograms. The main engines will provide a velocity change of 1.6 km/s for insertion into the low lunar orbit as well as the required 0.7 km/s for insertion into an earth return orbit.

The LST will use aerodynamic braking and the main engines to enter the earth's atmosphere.

6. EXTERNAL LAYOUT

Illustrations of the LST are presented in appendix A. A detailed list of vehicle parameters is presented appendix B in table B-1.

6.1 FLIGHT CONTROL SYSTEMS

The primary flight controls consist of trailing edge flaps and ailerons on the wing and a rudder surface on the trailing edge of the vertical tail. An illustration of the flight control surfaces is presented in figure 5.

Redundant digital flight control computers provide inputs for the flight control surfaces. Normal landings on earth are controlled by an autopilot system that coordinates deceleration of the LST, deployment of the parachutes, and deployment of the landing skids as well as firing of the AMS jets prior to touchdown. All systems can be manually controlled at the pilot stations as well.

Two side-stick controllers are located on the pilot consoles. The side-stick controllers sense pitch, roll and yaw inputs. The flight control computers measure force and displacement of the side-stick controllers to position the control surfaces using electric actuators. The flight control system is strictly electric and does not contain a hydraulic system.

The LST was designed to operate without pilot inputs for several reasons. Following sustained missions in a micro-gravity environment, a pilot may be physically unable to manipulate the controls during the earth landing. After extended periods in space, the pilot may also be “out of practice” for landings. In an emergency situation such as abandoning a space station, a capable pilot may not be aboard the LST.

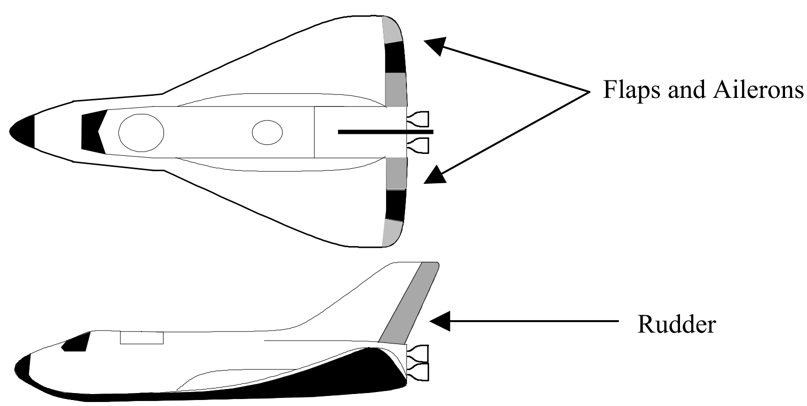

Figure 5: LST Flight Control Surfaces

6.2 PARACHUTE SYSTEM

The LST is designed to use a three-parachute system for deceleration during the earth landing. At approximately 15,000 feet MSL, the parachutes will deploy once the LST has decelerated to approximately 250 KTAS. The three parachutes have a radius of 24.7 meters. The parachutes will slow the rate of descent to 8 meter per second. The parachute system is located on the upper surface of the vehicle aft of the airlock just forward of the landing center of gravity. The LST will approach the earth's surface with a slight nose up attitude. Prior to impact the AMS system rockets will fire to cushion the landing. At ground impact the parachute system will be jettisoned. The parachute system mass is 612 kilograms.

During normal landings, the flight control system commands the parachute system to deploy. However, manual control is provided to the pilots if the autopilot system is not used to land the LST.

6.3 DOCKING SYSTEM

The LST is designed to dock with the International Space Station in a manner similar the US Space Shuttle. The pilots control the vehicle attitude and closure using the main engines for the initial rendezvous and the AMS system for the smaller adjustments and the final docking stage. A camera system located on the airlock assist the pilots during the rendezvous and final docking stage.

The airlock is located on the upper surface just aft of the pilot seats. This provides unobstructed egress and ingress. Unlike the space shuttle, the LST airlock is simply a passageway and not a chamber to be used for EVAs. The airlock consists of a small chamber and two hatches. There is insufficient volume for a crewmember to be closed in the airlock. After the LST is docked to either the space station or the lunar base, the airlock is pressurized so that a transfer can occur. Once pressures are equalized, the hatches are opened. The pressurization time is estimated to take approximately 2 hours.

Controls for opening, closing and pressurizing the airlock are located at the pilot station.

6.4 THERMAL PROTECTION

The LST surface is covered with fibrous refractory composite insulation tiles. The tiles vary in thickness depending on the location on the vehicle. The average thickness is 3 inches. The lower surface and vertical tail are covered with high temperature tiles with a density of 12 lb/ft^3. The upper surface is covered with low temperature tiles with a density of 9 lb/ft^3. The leading edges are covered with reinforced carbon-carbon to provide additional heat protection. The low temperature tiles can withstand temperatures up to 1200º F. The high temperature tiles can withstand temperatures up to 2300º F. The reinforced carbon-carbon provides protection for temperatures up to 3200º F.

The tiles protect the LST against the extreme heating conditions during reentry. The LST will enter the atmosphere with nose-high attitude. The lower surface and leading edges of the wings and tail will be exposed to the highest temperatures.

6.5 LANDING SKIDS

The LST is equipped with two skids, which are used during earth and lunar landings. The landing skids were chosen to reduce mass and system complexity. The skids deploy from the lower surface of the LST just prior to landing. Illustrations of the parachute system and landing phases are presented in figures 6, 7 and 8. Detailed landing phase illustrations, figures A-6 and A-7, are presented in appendix A.

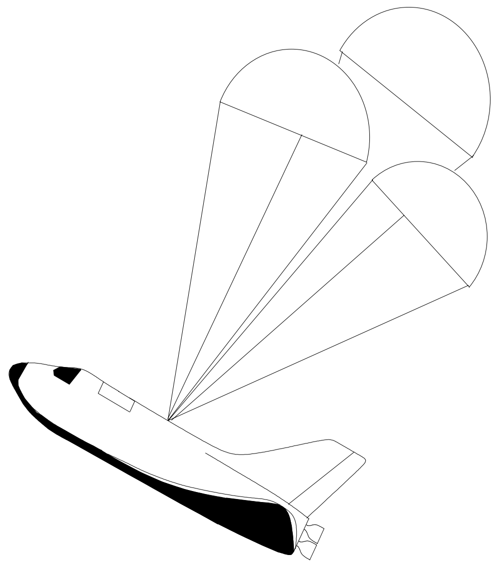

Figure 6: LST Parachute System

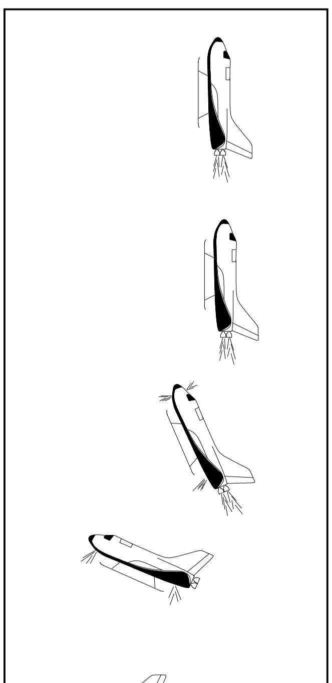

Figure 7: Lunar Landing Phases

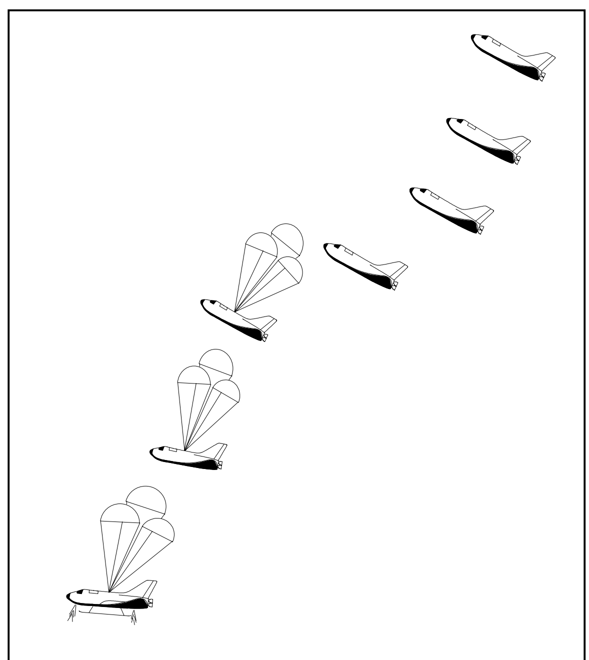

Figure 8: Earth Landing Phases

7. INTERNAL LAYOUT

7.1 CREW POSITIONING

The LST is designed to carry a crew of eight. Two crewmembers are designated as pilots. The two pilots seats are located in the forward portion of the crew cabin behind the windows. The remaining 6 crew or passengers sit behind the pilots in three rows. While in a micro-gravity environment, the six passenger seats can be folded into the floor to provide additional living space. The crew seats are extremely light weight and have stroking capability similar to a helicopter which provide protection from injury should the AMS jets fail to cushion the landing.

Food and supplies for the crew are located in lightweight lockers along the sides of the crew cabin. Each crewmember is allocated 5 kg of food per day for an 11-day mission.

Four lightweight sleeping bags are carried on the LST that can be attached the cabin walls. The habitable volume of the LST is 45.3 cubic meters.

7.2 CREW CONTROLS/INTERFACE

Controls and instruments to monitor the LST systems are located in the forward portion of the crew cabin around the pilots. The pilots can control the AMS and main engines during docking and rendezvous. Additionally, the pilots can manually fly the LST using redundant controls during the landing phases.

The toilet and waist system is located on the right of the crew cabin behind the passengers. The controls for the waist system are located at the station. The airlock controls are located at the pilot station.

7.3 PAYLOAD ARRANGEMENT

The payload of the LST is carried internally and is transferred through the airlock hatches. For lunar missions an internal payload of 2,500 kg is carried. For space station missions, propellant tanks are removed to allow an additional 2,000 kg to be carried. Removal of the propellant tanks and reconfiguration of the crew cabin is accomplished during the vehicle preparation phase prior to launch. For the space station mission, the LST carries 11,500 kg of propellant (2,440 kg less than the lunar mission). The payload is located in the aft section of the crew cabin and below the cabin floor. An illustration of the payload arrangement, figure A-5, is presented in appendix A.

7.4 PROPELLANT ARRANGEMENT

The main engine propellant is carried internally in the wings and in the aft section of the LST fuselage. The LST carries 13,940 kg of liquid hydrogen and oxygen for the lunar profile and lunar return profiles. For the space station mission, the LST carries 11,500 kg of liquid propellant. Propellant tanks are removed for the space station mission to provide additional space for the increased internal payload.

An illustration of the cabin layout is presented in figure 9. Illustrations of the two different propellant configurations are presented in figure 10.

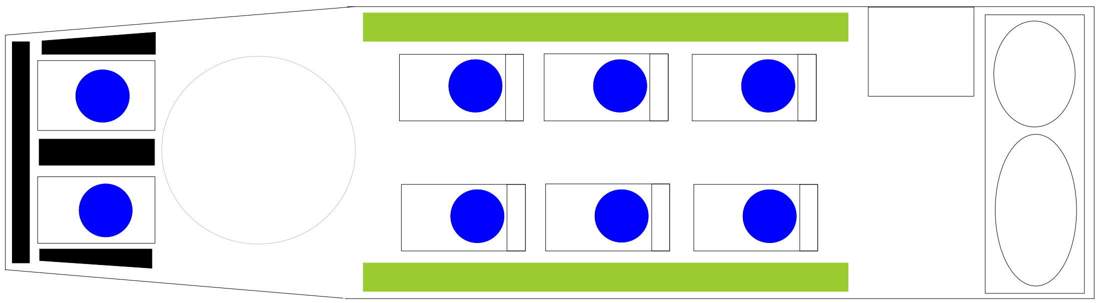

Figure 9: LST Crew Cabin Illustration

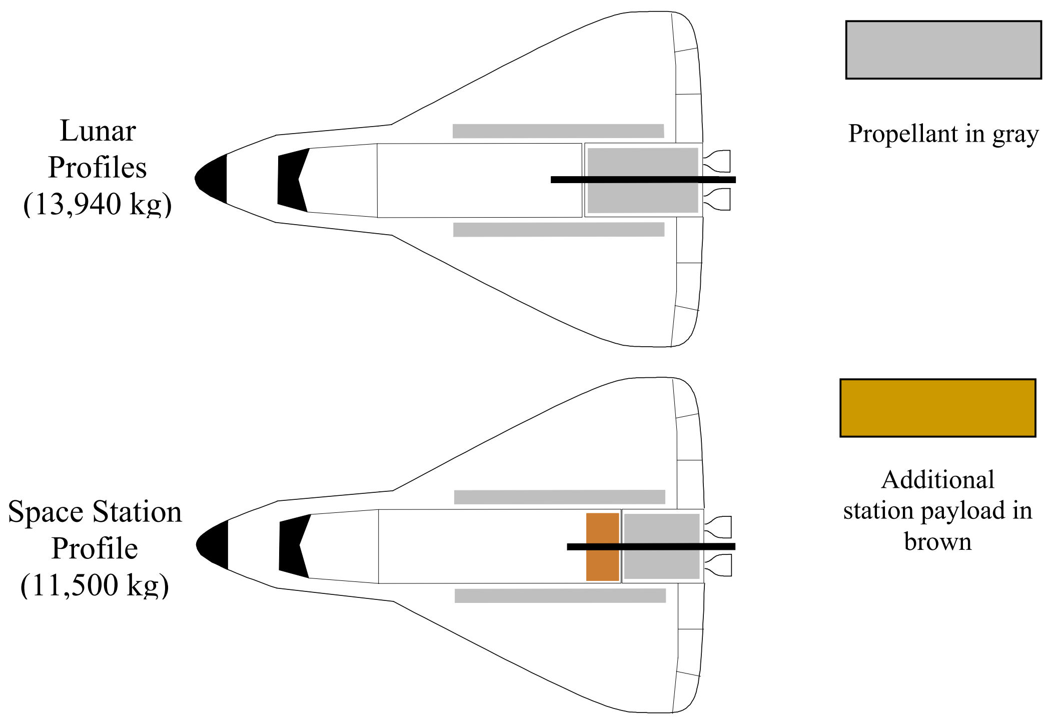

Figure 10: LST Internal Propellant Configurations

8. SUMMARY OF APPENDICES

The following list is a description of the material contained in the appendices.

A: Vehicle Drawings

Appendix A contains vehicle and configuration drawings of the LST and the proposed launch systems.

B: Vehicle Parameters

Appendix B contains a table listing many of the vehicle parameters and dimensions.

C: Design Method Outline

Appendix C contains the Design Method Outline used to during the design process.

D: Design Sequence and Decisions

Appendix D presents the reasoning and design decisions that were made during the design process.

E: Performance Calculations

Appendix E contains both examples of sample calculations and actual calculations made during the design process.

F: Reference Data

Appendix F contains reference material used during the design process.

G: Definitions and Vehicle Descriptions

Appendix G contains descriptions of other space vehicles and definitions to be used as a reference for the reader.

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BIBLIOGRAPHY

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Note: Some websites have since changed content available or no longer exist.

APPENDICES

APPENDIX A: VEHICLE DRAWINGS

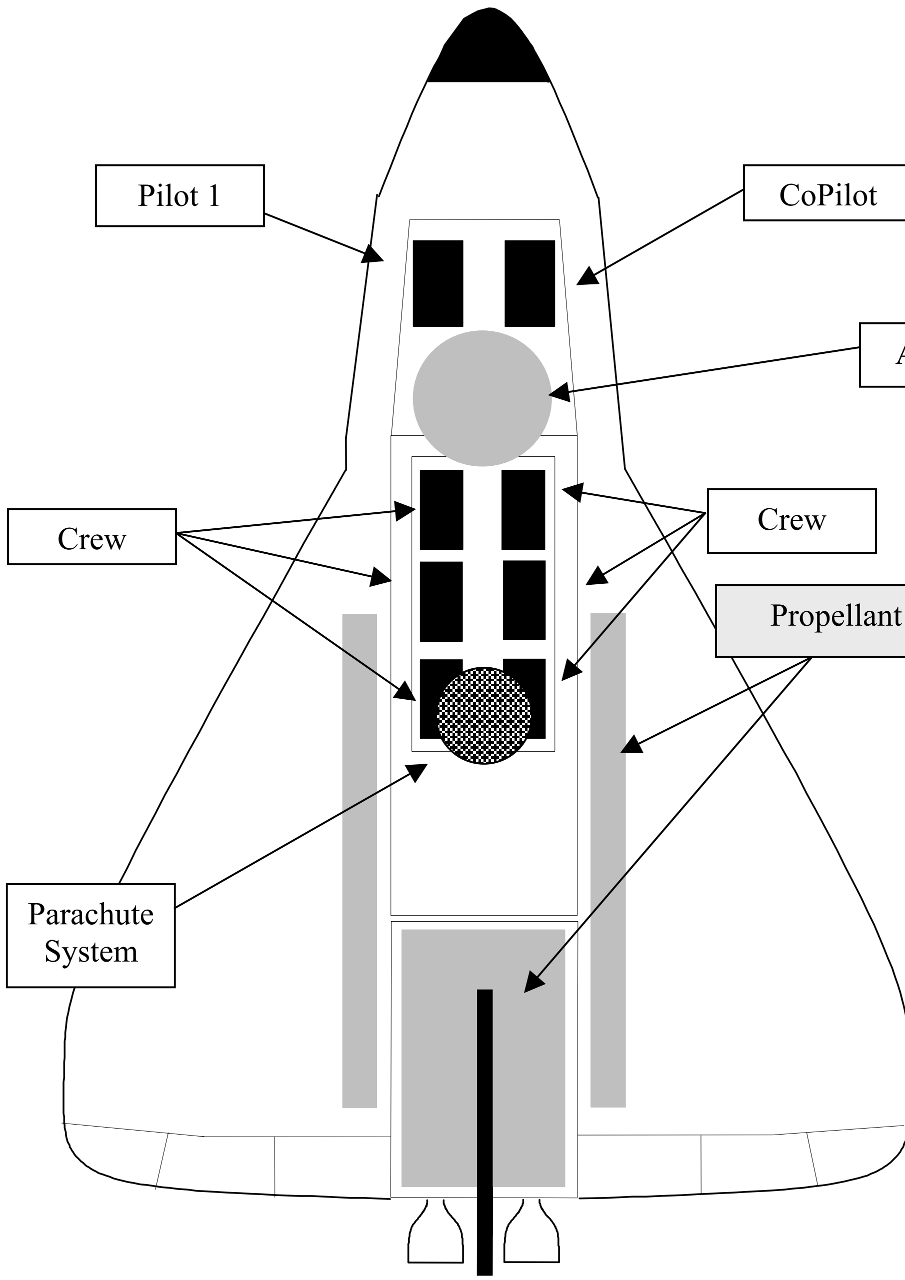


Figure A-1: LST Top View Illustration

Figure A-2: Space Station Profile Launch Configuration Illustration

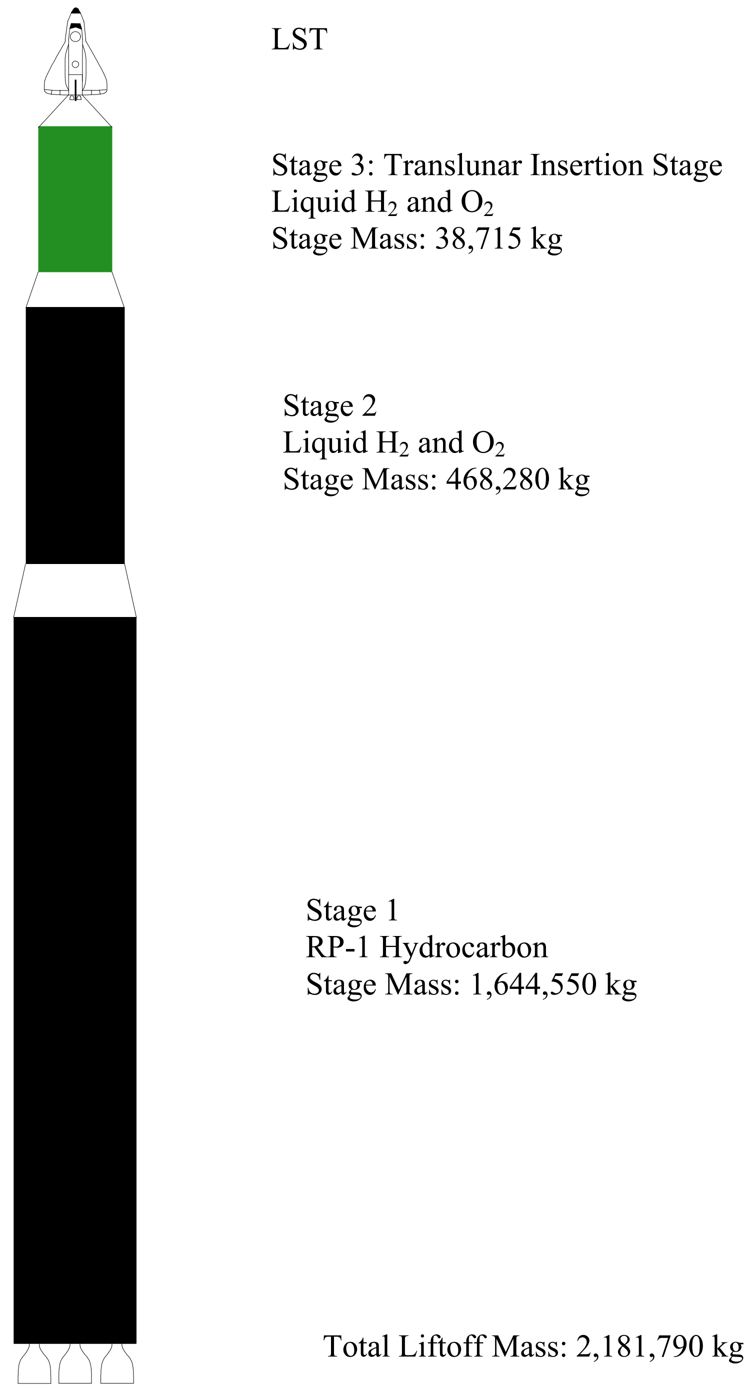
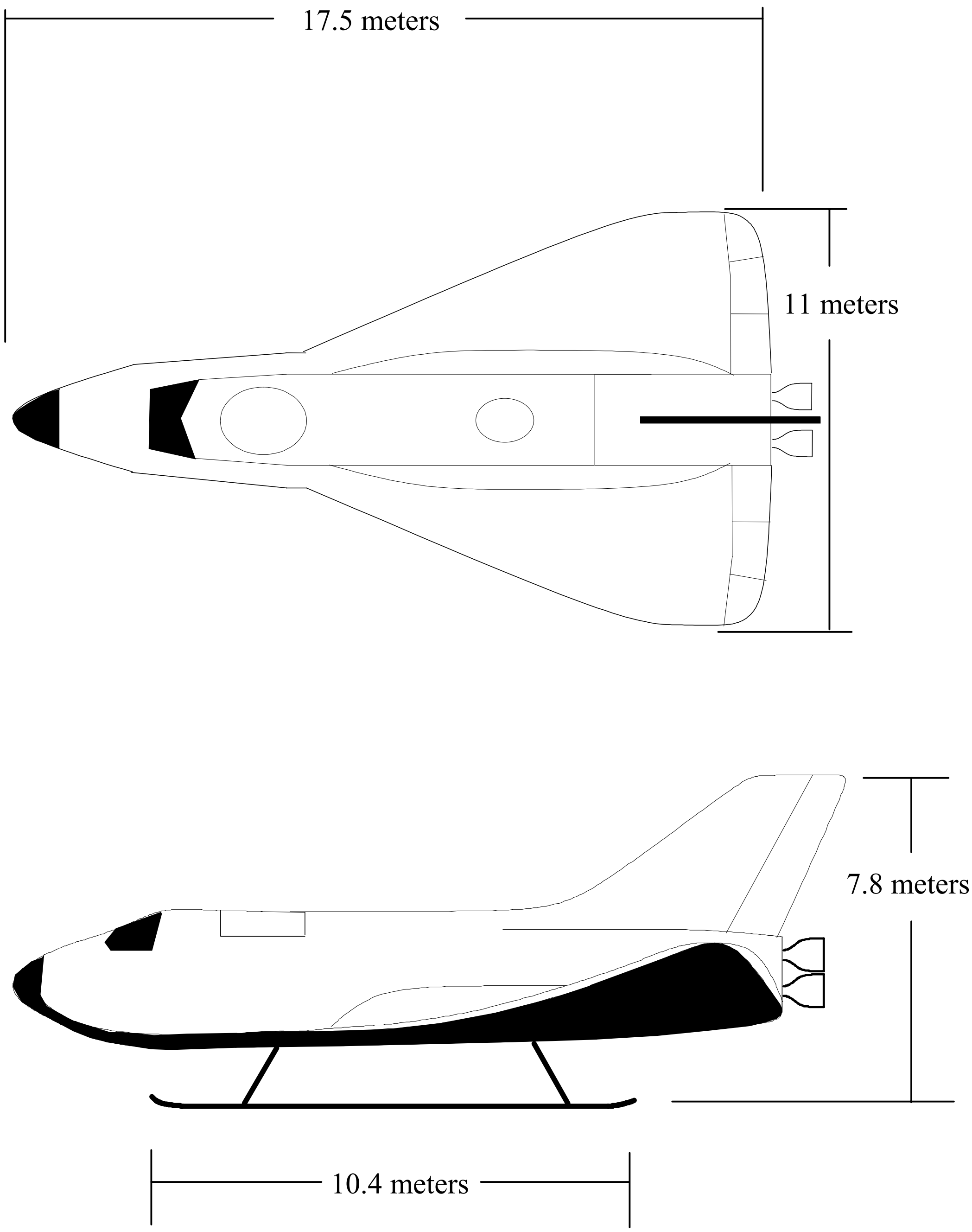


Figure A-3: Lunar Profile Launch Configuration Illustration



Space Station Mission Launch Mass:	29,800 kg
Lunar Mission Launch Mass:	30,240 kg
Lunar Return Mission Launch Mass:	27,740 kg

Figure A-4: LST Detailed Two View Illustration

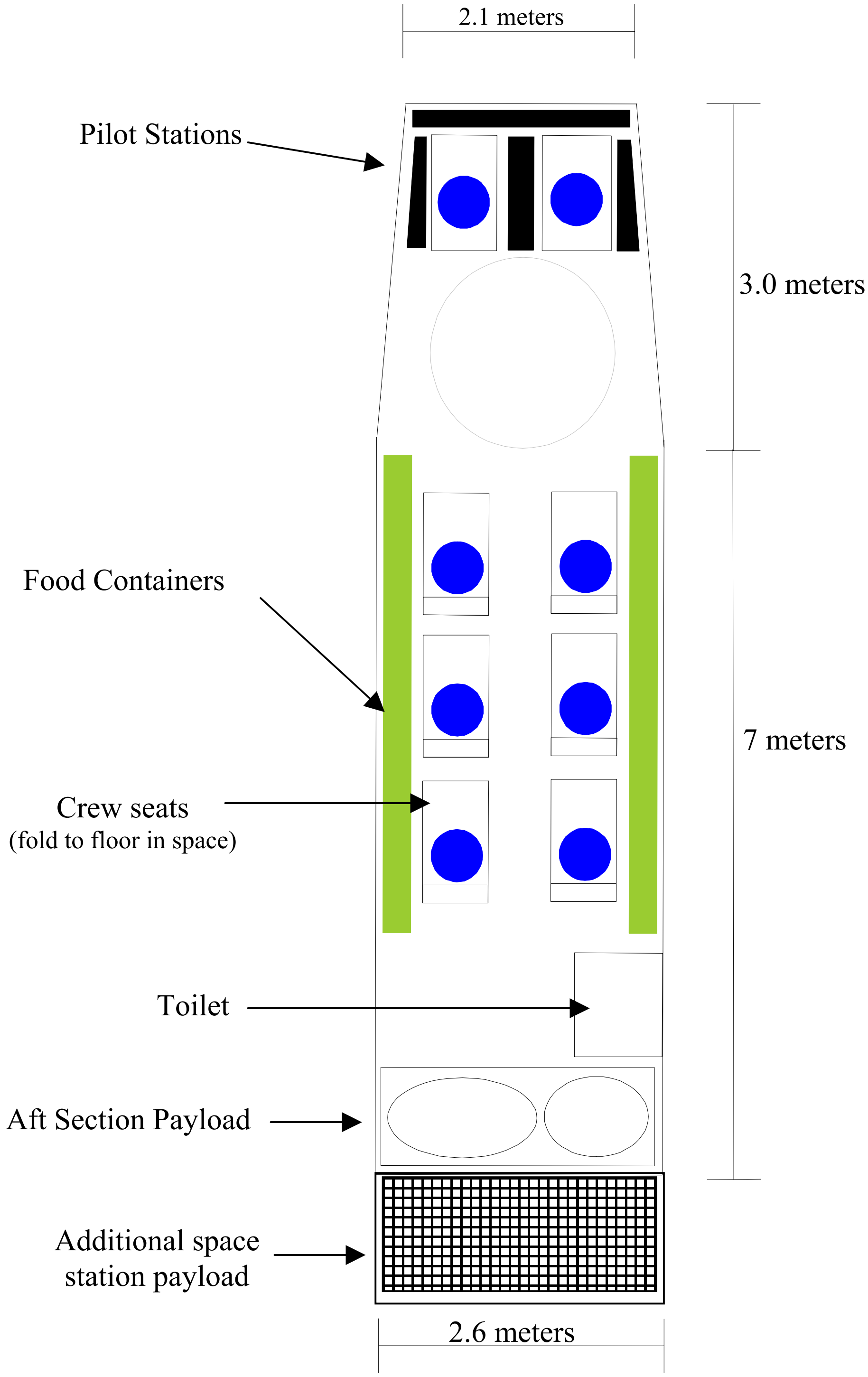


Figure A-5: Detailed LST Internal Layout Illustration

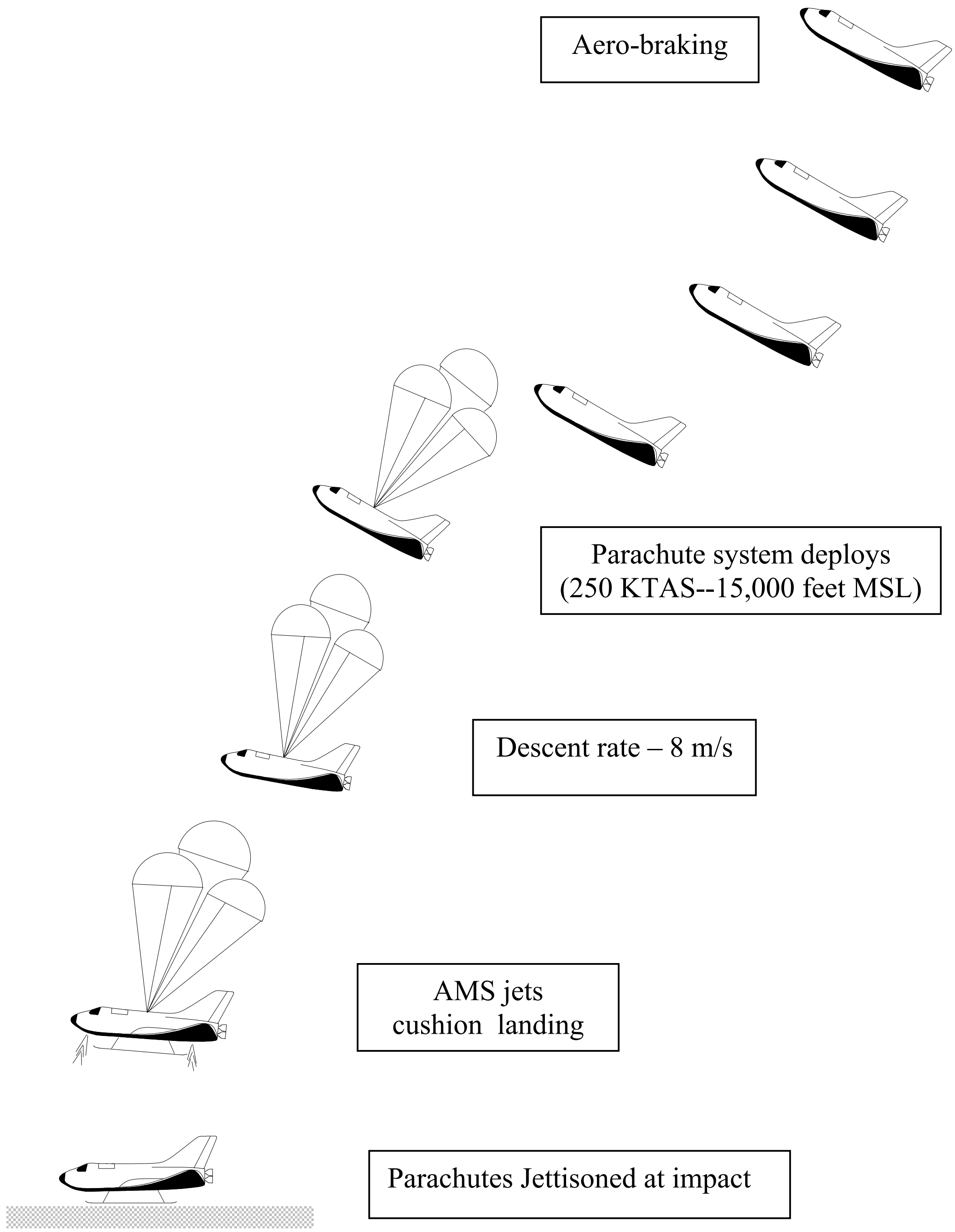


Figure A-6: Detailed Earth Landing Phases

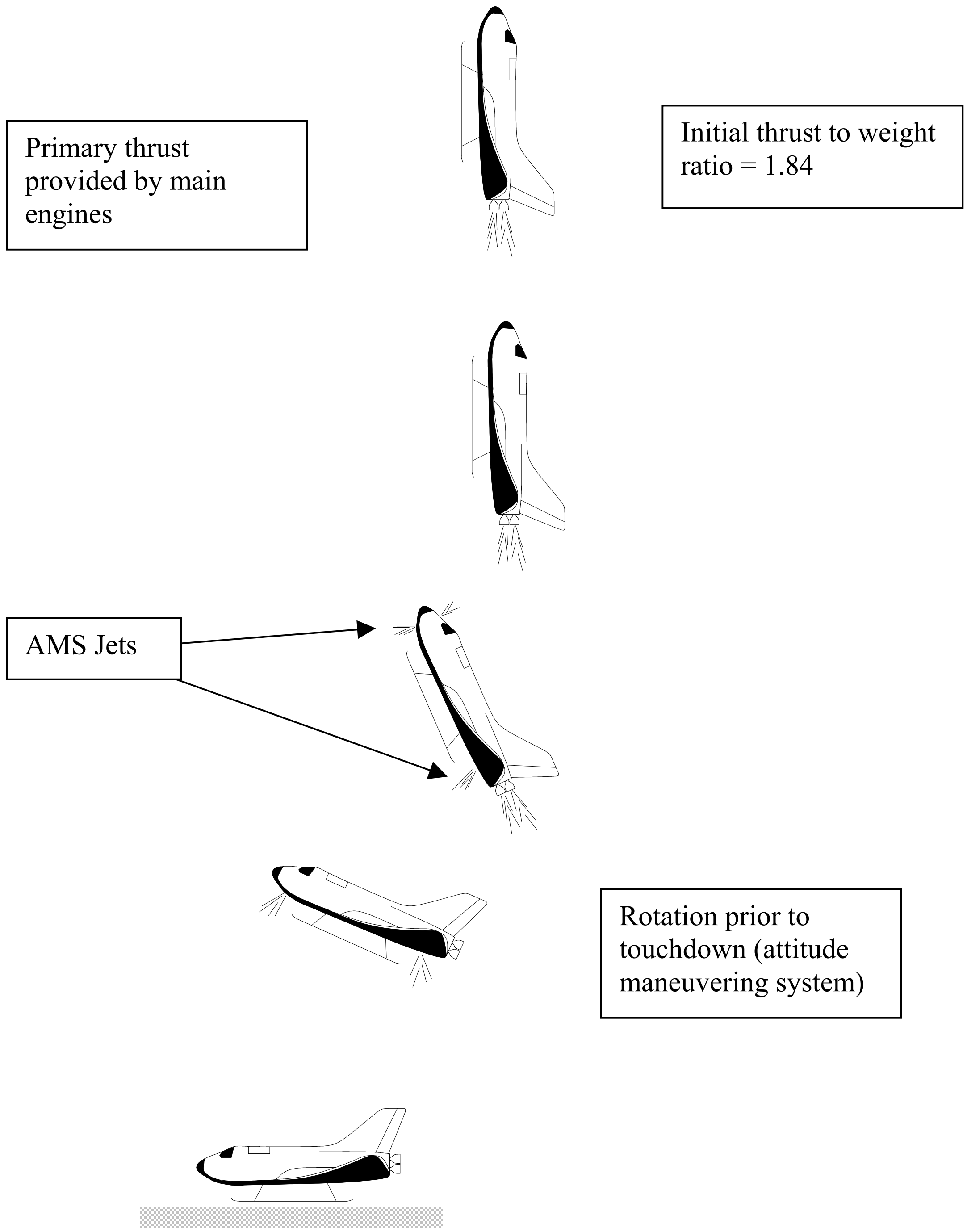


Figure A-7: Detailed Lunar Landing Phases

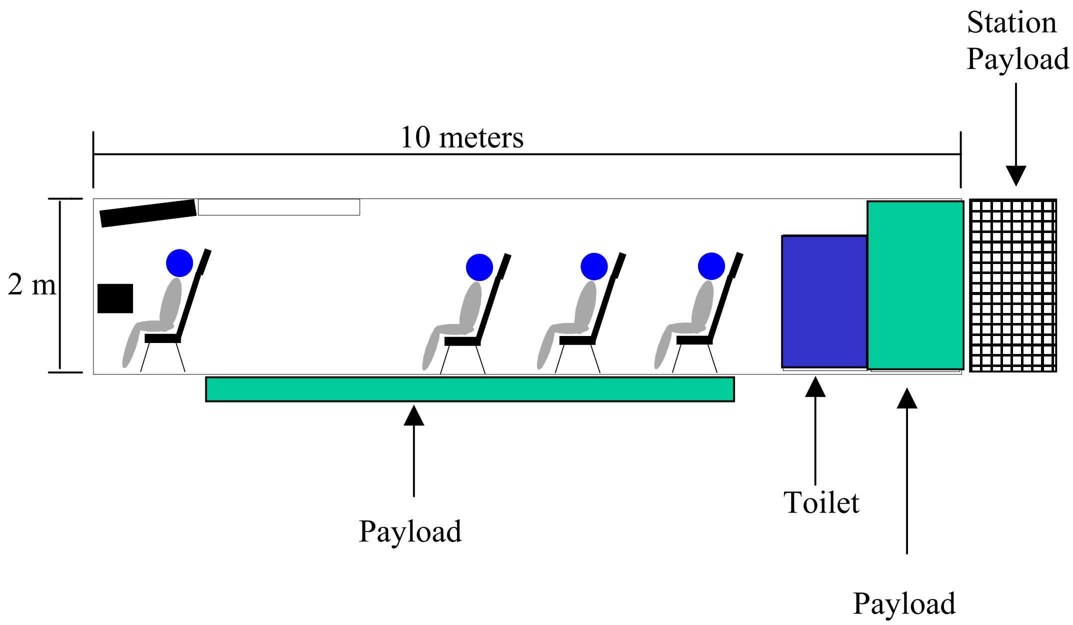


Figure A-8: LST Cabin Crew Side View

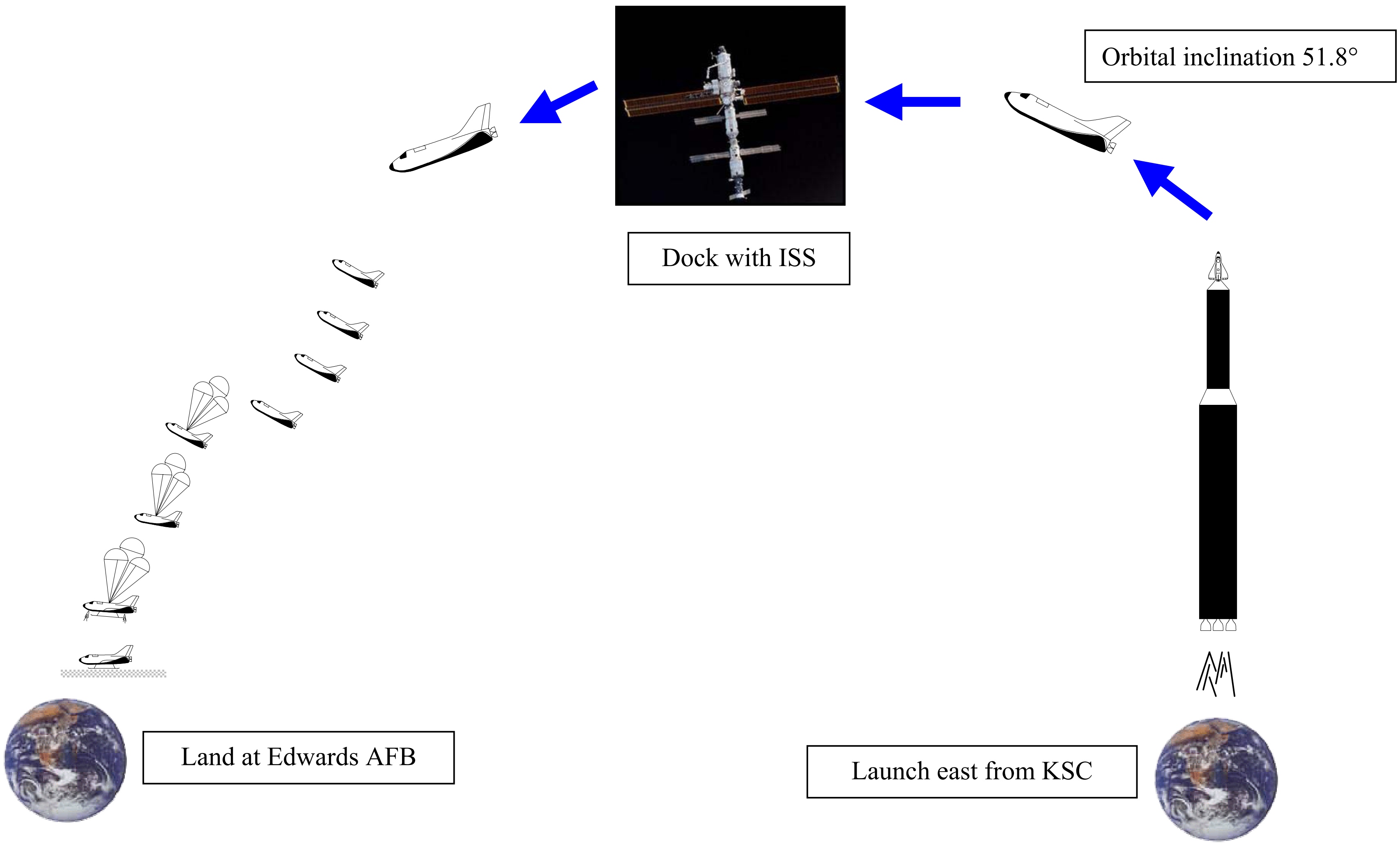


Figure A-9: Detailed Space Station Profile Overview

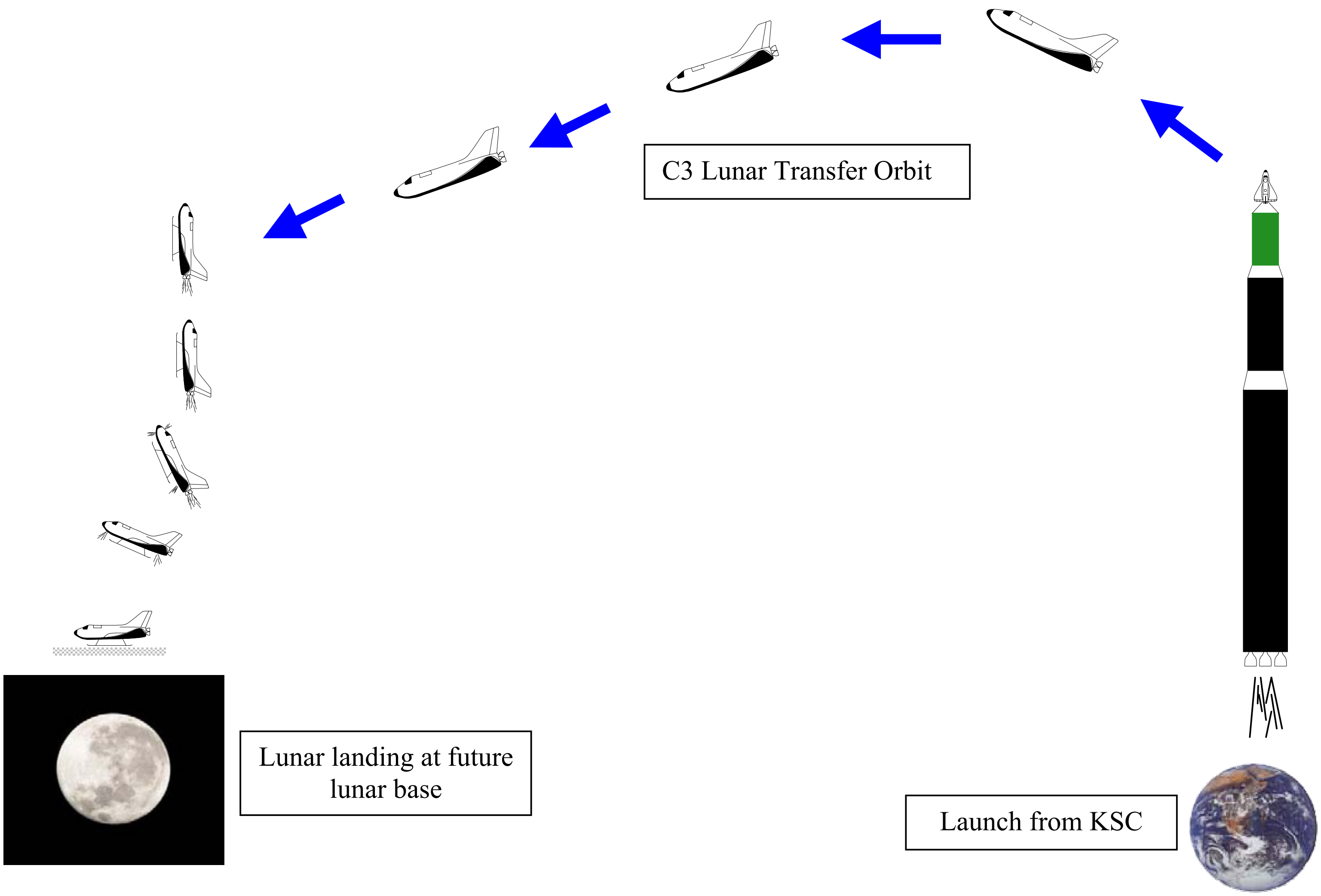


Figure A-10: Detailed Lunar Profile Overview

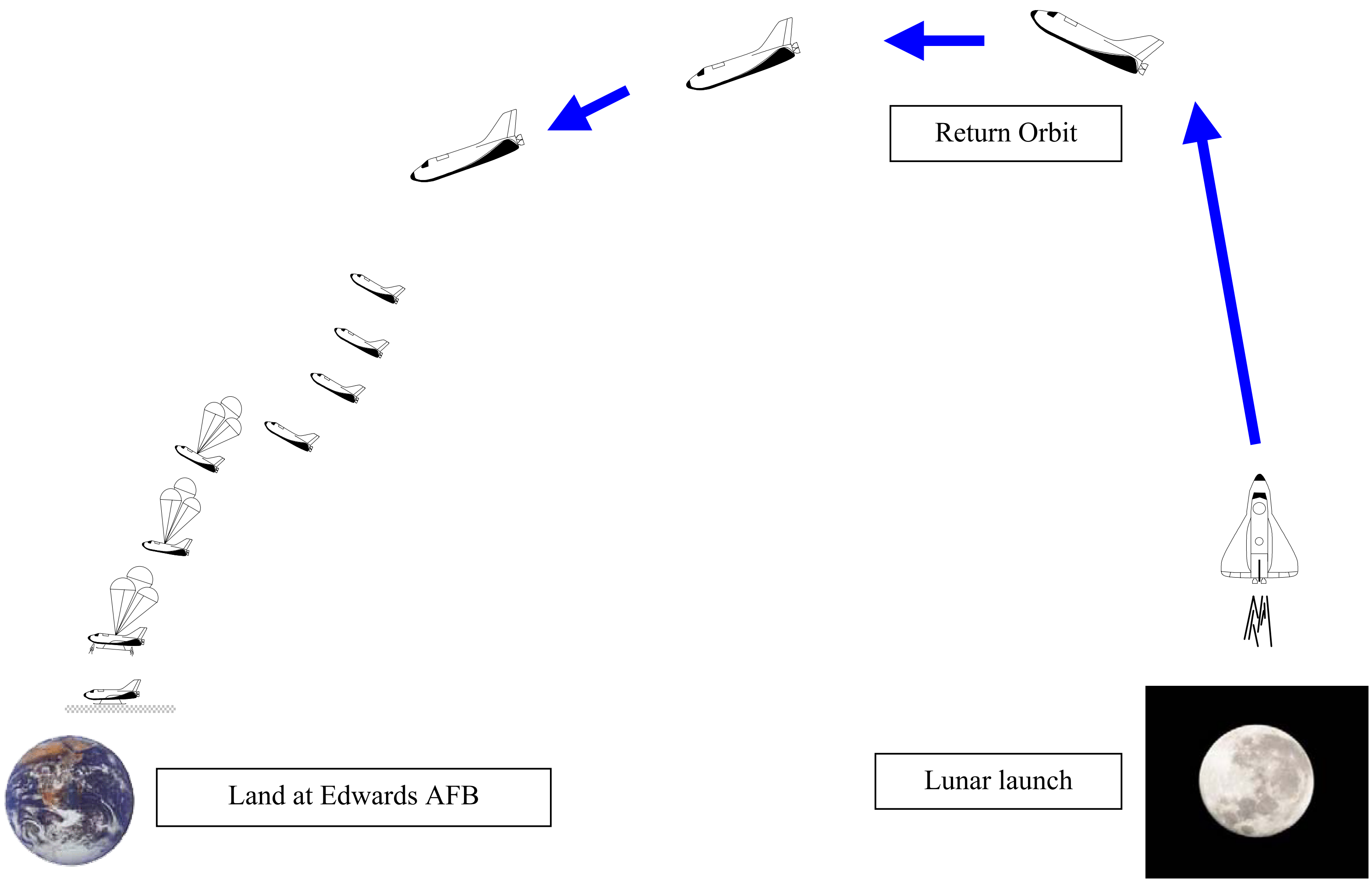


Figure A-11: Detailed Lunar Return Profile Overview

APPENDIX B: VEHICLE PARAMETERS

Table B-1: LST Vehicle Parameters

Vehicle Parameter	Value
Crew	8
Length	17.5 m
Wing Span (b)	11 m
Height (skids extended)	7.8 m
Height (skids retracted)	6.5 m
Landing skid length	10.4 m
Landing skid height	1.3 m
Wing Area (S)	60 m^2
Aspect Ratio (A)	2.0
Sweep Angle (Λ)	59 degrees
Vertical tail area	12.5 m^2
W/S during earth landing (wing loading)	3.27 kN/m^2
LST lunar configuration on-orbit mass	30,240 kg
LST space station configuration on-orbit mass	29,800 kg
LST lunar launch mass (launch from moon)	27,740 kg
Internal propellant mass (main engines)	13,940 kg
Internal payload (lunar mission)	2,500 kg
Internal payload (space station mission)	4,500 kg
Vehicle attitude control system propellant mass	1,100 kg
Consumables (fuel, food, life support)	16,540 kg
Crew mass (with flight gear)	720 kg
Dry mass	13,000 kg
Maximum earth landing mass	17,000 kg
Main engine maximum thrust (vacuum)	21.88 kN (each)
Main engine propellant and oxidizer	H_2 and O_2
Attitude maintenance system propellant and oxidizer	NTO/MMH
Stall Speed (V_S)	180 KTAS
Habitable volume	45.3 m^3
Lunar profile stage masses	1,644,550 kg (stage 1) 468,280 kg (stage 2) 38,715 kg (stage 3)
Lunar liftoff total mass	2,181,790 kg
Space station profile stage masses	952,500 kg (stage 1) 230,940 kg (stage 2)
Space station profile liftoff total mass	1,213,670 kg

APPENDIX C: DESIGN METHOD OUTLINE

A design method outline was developed using the following references:

- Space Mission Analysis and Design. (1992)
- Aircraft Design: A Conceptual Approach Third Edition. (1999)
- UTSI class AS506 Lecture Notes. Solies (2002)

Design Method Outline

Mission Requirements Definition
- Profiles
- Duration
- Crew and Payloads

Basic Conceptual Decisions
- Layout
- Profiles

Initial Sizing Estimates (Historical)
- Mass
- Dimensions

Initial Propulsion Requirements
- Space Station Profile
  - Concept of Operations
  - Orbital Calculations and Delta V Budget/ Propellant Requirements and Budget
  - Earth Reentry Thrust Requirements
  - Propulsion Choices
- Lunar Profile
  - Concept of Operations
  - Orbital Calculations and Delta V Budget/ Propellant Requirements and Budget
  - Lunar Landing Thrust Requirements
  - Propulsion Choices
- Vehicle Attitude Maintenance and Control

External Vehicle Layout
- Aerodynamic Braking Drag
- Lifting Body Airfoils
- Parachutes
- Docking requirements
- Heat/Radiation requirements
- Wing/Tail Sizing
- Drag Determination

Internal Vehicle Layout
- Crew Accommodation/Crew Interface

- Payload Arrangement
- Drawings

Stability and Control Systems

- Launch Environment
- Space Environment
  - Orbital Transfers
  - Vehicle Attitude Maintenance
  - Docking
- Recovery
  - De-burn / Reentry
- Landing

Refined Sizing Estimates

Refined Propulsion Estimates

APPENDIX D: DESIGN SEQUENCE AND DECISIONS

DESIGN METHOD OUTLINE

The design method outline presented in appendix C was used as a starting template to organize the design process. The outline was not strictly adhered to in all cases. Certain steps were deleted, added, or performed out of sequence to facilitate the process.

MISSION REQUIREMENTS DEFINITION

PROFILES

The Lunar Shuttle Transport (LST) is designed to fly three different profiles. They are described below.

Space Station Profile
Launch from Kennedy Space Center (KSC), FL into a low earth orbit with an inclination of the International Space Station. Rendezvous and dock with ISS and deliver crew. Return from the ISS and land in the Edwards Air Force Base complex in southern California.

Lunar Profile
Launch from Kennedy Space Center, FL into a low earth orbit. Transfer to a lunar transfer orbit. Enter a low lunar orbit and land on the moon. As a contingency, the spacecraft should also be able to bypass the lunar landing and return to earth.

Lunar Return Profile
Launch from the moon with the crew and return to the Edwards Air Force Base complex in southern California. It is assumed that a fuel source will be available on the moon in the future. The spacecraft should be able to refuel at the lunar base and return to earth.

DURATION

The mission duration was an important element in determining the required sizing of the vehicle for crew accommodation and living space. The estimates of mission duration are presented in tables D-1, D-2 and D-3.

The analysis of the missions showed that the lunar mission would likely last 3 ½ days, but may extended to 11 days if the lunar landing had to be bypassed.

The space station profile would likely last between 3 to 7 days not including delays at the space station. While docked the astronauts would rely on the station's systems for life support.

Table D-1: Space Station Profile Mission Duration Estimates

Event	Time
Launch and insertion into low earth orbit	10 minutes[3]
Rendezvous with International Space Station (ISS)	1 day
Docking evolution	
Approach/docking	3 hours[4]
Pressurization time for air lock	2 hours[2]
Crew transfer (not including time at ISS)	1 hour
Undocking evolution	
Depressurization	2 hours[2]
Undocking/Departure	2 hours
Return to earth (allow for day landing)	1 day
Deorbit burn to touchdown	1 hour[3]
Margin (weather, additional rendezvous time)	3 days
Total Mission Time Estimate[1]	5 days 10 hours

Notes: 1. Delays at ISS not included in total.
2. Source: http://spacelink.nasa.gov
3: Source: Columbia Accident Investigation Report: Report Volume I
4. Source: http://www.shuttlepresskit.com/STS-98/rendezvous13.htm

Table D-2: Lunar Mission Duration Estimates

Event	Time
Launch and insertion into low earth orbit	10 minutes[2]
C3 Transfer Orbit (LEO to Lunar Orbit)	75 hours[3]
Insertion / orbit time in low lunar orbit	10 minutes
Delay in lunar orbit	12 hours
Lunar landing evolution	1 hour 10 minutes[1]
Pressurization time for airlock	2 hours
Crew transfer to lunar station	1 hour
Contingency (Lunar Landing Aborted)	
Return C3 Transfer Orbit	75 hours
Return to earth (allow for day landing)	1 day
Deorbit burn to touchdown	1 hour[2]
Margin (weather)	3 days
Total Mission Time (no lunar landing)	11 days

Notes: 1. Source: http://nssdc.gsfc.nasa.gov/database/MasterCatalog?sc=1969-059C
2. Source: Columbia Accident Investigation Report: Report Volume I
3. Source: http://visitor.broaddaylight.com/spacekids/FAQ_44_359.shtm

Table D-3: Lunar Return Mission Duration Estimates

Event	Time
Launch and insertion into low lunar orbit	10 minutes
C3 transfer orbit (Lunar to LEO)	75 hours[1]
Return to earth (allow for day landing)	1 day
Deorbit burn to touchdown	1 hour[2]
Margin (weather)	3 days
Total Mission Time	7 days

Notes: 1. Source: http://nssdc.gsfc.nasa.gov/database/MasterCatalog?sc=1969-059C
2. Source: Columbia Accident Investigation Report: Report Volume I

CREW AND PAYLOADS

The LST is designed to carry eight crew for both the station and lunar profiles. The primary purpose of the vehicle is a crew transport system.

The design requirements called for two different internal payload requirements. The LST launch system is designed to carry an internal payload of 4,500 kg for the station profile and 2,500 kg for the lunar profile. The LST is not required to carry an internal payload for the launch from the moon.

The primary purpose of the LST is carrying crew to and from the locations, not necessarily heavy payloads. Therefore, the internal space for experiments and payloads is very limited.

The LST is designed so that transfer of the payload can be accomplished through the air lock. Therefore payload dimensions are limited.

BASIC CONCEPTUAL DECISIONS

LAYOUT

The two primary factors driving the layout decisions were the requirement for the vehicle to carry eight personnel as well as make a return through earth's atmosphere.

The decision was made to model the LST after vehicles such as the US Space Shuttle in order to take advantage of aerodynamic braking and a lifting body design. An enlarged version of the Soyuz or Apollo capsules did have some advantages. However, the lifting body design was chosen with the idea that structural weight could be reduced by limiting g-forces on re-entry and relying more heavily on the earth's atmosphere to decelerate.

One disadvantage of the space shuttle design is the complexity and weight of the landing gear system. The LST was designed to take advantage of a parachute system for the final phase of the landing. This would alleviate the increased training requirements for pilots and would also allow a reduction in complexity and weight. The obvious disadvantage of the choice is the reliance of the proper operation of the parachute system. However, the demonstrated reliability of systems such as the Soyuz parachute system makes this choice a likely reliable option for return to earth.

Since the LST will fulfill two different missions, the decision was made to develop one shuttle vehicle and to use different staging options depending on the mission requirements. When performing a lunar mission, the LST would use additional staging to enter a translunar orbit.

PROFILES

The initial analysis of the three profiles showed that the lunar mission would require greater thrust requirements and therefore additional staging. Additionally, the decision was made to design a vehicle that could rely on its internal fuel for the lunar landing evolution. The use of landing skids would be used during landings on the moon and earth. The vehicle also needed a means to provide thrust for the lunar landing and launch evolutions. Therefore, the engines needed to have sufficient thrust and propellant to land on and launch from the moon.

INITIAL SIZING ESTIMATES

MASS

Historical examples of space vehicles were used to approximate the mass of the LST. Historical examples of proposed and actual space vehicles are presented in table D-4.

The HL-42, a proposed replacement to the US Space Shuttle, was deemed to be the vehicle closest to the LST in size and mass. The LST had the same requirement as the HL-42 to transport personnel to and from the International Space Station. Since the LST would require additional propellant for the lunar mission and four additional crew, the first estimation for the LST mass was 25,000 kg.

Table D-4: Examples of Space Launch Vehicles

Vehicle	On Orbit Mass	Payload Mass	Purpose	Crew
HL-20[1]	10,884 kg	545 kg	Proposed Space Station Crew Return Vehicle	8
HL-42[1]	21,144 kg	4,300 kg	Proposed Space Shuttle Replacement	4
Apollo Lunar Transfer Vehicle[3]	47,000 kg	---	Lunar Mission	3
US Space Shuttle[2]	90,000 kg (approximate)	22,500 kg	Science and Space Station Support	7

Notes: 1. Source: http://www.friends-partners.ru/partners/mwade/craft/hl42.htm
2. Source: http://spacelink.nasa.gov
3. Translunar trajectory, source: http://users.commkey.net/Braeunig/space/specs.htm

After analyzing the fuel required to complete a lunar landing with adequate margin, the second estimate was increased by 5,000 to 30,000 kilograms. The first refined mass estimate produced an on-orbit mass of 29,773 kilograms. Next, calculations were performed to determine required wing and tail areas. Additionally, a math error was discovered in the initial mass estimates. The final mass estimate for the lunar configuration was 30,240 kilograms. The LST mass estimates determined during the design process are presented in table D-5.

DIMENSIONS

The initial estimates for the vehicle dimensions were determined using historical examples and the internal propellant volume estimates. Historical and proposed vehicle dimensions are presented in table D-6.

The LST dimensions were estimated by scaling up the HL-42 by a factor of 1.3. The dimension estimate iterations of the LST are presented in table D-7.

Table D-5: LST Mass Estimate Iterations

Iteration Number	On Orbit Mass	Description / Reason for New Iteration
One	25,000 kg	Initial estimate using historical examples.
Two	30,000 kg	Propellant required for lunar mission needed to be increased. Initial calculations performed using this estimate.
Three	29,773 kg	First refined mass estimate performed; several parameters estimated
Four/Final	**30,240 kg**	Calculations showed that wing area could be reduced and the vertical tail area had to be increased. Additionally, a math error was corrected in the mass estimate calculations.

Table D-6: Historical Examples of Space Craft Dimensions

Vehicle	On Orbit Mass	Length (m)	Wingspan (m)	Height (m)	Crew
HL-20[1]	10,884 kg	8.9	7.1	---------	8
HL-42[1]	21,144 kg	12.8	10.2	5.5 (with landing gear)	4
US Space Shuttle	91,000 kg (approximate)	37.24	23.79	17.27	7

Notes: 1. Source: http://www.friends-partners.ru/partners/mwade/craft/hl42.htm
2. Source: http://users.commkey.net/Braeunig/space/specs.htm

Table D-7: LST Dimension Estimate Iterations

Iteration Number	Length (m)	Wingspan (m)	Height (m)	Description/Reasoning
One	16.6	13.3	7 (skids deployed)	Initial estimate (HL 42 scaled up by factor of 1.3)
Two	17.5	13.3	7	Extra space required for propellant
Three	17.5	11	7	Wing area was greater than required (reduced from 72 m^2 to 60 m^2)
Four/Final	17.5	11	7.8	Vertical tail area increased after calculations.

INITIAL PROPULSION REQUIREMENTS (SPACE STATION PROFILE)

CONCEPT OF OPERATIONS

The primary focus of the design was on the LST crew vehicle not necessarily the ideal boosting system. There were several options available to insert the LST into a low earth orbit (LEO). Possible options were to use staging similar to the Saturn V or to use solid rocket boosters similar to the US Space Shuttle. If the payload capabilities were increased, launch systems similar to Ariane V would provide another option. For this design, staging was used to for calculating launch masses and propellant requirements.

The basic concept for the space station profile was to rely on some type of launch vehicle to place the LST in LEO and then rely on the LST main engines for orbit phasing with the space station. The LST main engines will also be used for the reentry burns. The LST will be equipped with a separate attitude maintenance system for the final phases of shuttle docking and attitude maintenance while in orbit.

For the station profile, the LST is designed to be launched using a two-stage liquid rocket system. The first stage uses RP-1 as a propellant and the second stage uses liquid hydrogen. Space Shuttle staging was used as a rough order of magnitude to evaluate the calculated staging requirements for the LST. The space shuttle staging parameters are presented in table D-8 and the calculated LST staging parameters are presented in table D-9. Staging calculations are presented in appendix E.

Table D-8: Space Shuttle Staging

Stage	Propellant Mass[1] (kg)	Propellant (Oxidizer)	I_{sp} (s)
Solid Rocket Boosters	1,008,000	TB-H1148 HB Polymer	268
Main Engines	730,000	Liquid Hydrogen (Liquid Oxygen)	453 (vac)

Note: 1. Source: http://users.commkey.net/Braeunig/space/specs.htm

Table D-9: LST Space Station Profile Staging

Stage	Propellant Mass[1] (kg)	Propellant (Oxidizer)	I_{sp}[2] (s)
One	844,820	RP-1 Hydrocarbon (Liquid Oxygen)	350
Two	204,770	Liquid Hydrogen (Liquid Oxygen)	450

Notes: 1. See appendix E for calculations
2. Source for specific impulse values: Space Analysis and Design

ΔV BUDGET/ PROPELLANT REQUIREMENTS AND BUDGET

A "ΔV" budget was developed for the profiles. The space station ΔV budget is presented in table D-10. To minimize the required fuel, the LST will be launched directly into an orbit with the same inclination (51.8 degrees) as the space station. Additionally, the LST will be launched to the east to take advantage of the earth's rotation to minimize the ΔV requirement. The LST launch time will also be coordinated in a similar manner as the current US Space Shuttles launches are timed.

The required ΔV for the LEO insertion was 10,200 m/s. The value of 9,700 m/s was found in a reference (www.pma.caltech.edu). This value was verified using calculations to estimate the drag and gravity losses during a launch. The drag loss for the LST was estimated to be 3 percent or 230 m/s. The gravity loss was estimated to be 1,300 m/s. The justification for these values can be found in appendix E. Delta V loss comparison are presented in table D-11.

A 10 percent "ΔV" margin was established for the LST launch. The 10 percent margin was added to account for propellant system inefficiencies and to ensure the LST was inserted into the LEO. The 10 percent margin was also added for the lunar profile mission. Stages 1 and 2 will be used for insertion into LEO and the LST main engines will be used for orbital phasing with the ISS and returning to earth.

A space station propellant budget was calculated using the "ΔV" budget. Calculations are presented in appendix E and the propellant budget is presented in table D-12.

Table D-10: Space Station Profile ΔV Budget

Item	ΔV requirements[1] (m/s)	Stage Providing ΔV
Burn 1 (earth surface to LEO)	9,700[1]	
Burn 2 (plane change)	3,100[2]	
Combined Burn (1+ 2)	10,200[2]	Stage 1 and Stage 2
10 % margin for LEO insertion	1,020	Stage 1 and Stage 2
Burn 3 (space station rendezvous)	150	LST Main Engines
Burn 4 (Re-entry)	152[3]	LST Main Engines
ΔV available for LST main engines	2,112	LST Main Engines
Total[4] ΔV	26,430	

Notes: 1. Source: http://www.pma.caltech.edu/~chirata/deltav.html
2. Calculated from plane change (28.5 to 51.8 degrees). Appendix E.
3. Source: Shuttle Crew Operations Manual.
4. Total ΔV does not include attitude maintenance.

Table D-11: ΔV Loss Comparisons for Space Shuttle, Titan IV, and LST

Parameter	Space Shuttle[1] m/s	Titan IV[1] m/s	LST[2] m/s
ΔV drag loss	125	65	230
ΔV gravity loss	780	750	1300
Total losses	905	815	1530

Notes: 1. Source: Space Mission Analysis and Design.
2. See calculations in appendix E.

Table D-12: Space Station Profile Propellant Budget

Item	Stage ΔV (m/s)	Initial Mass[1] (kg)	Propellant Mass[1] (kg)	Stage I_{sp}[2] (s)
Stage 1	4,207	1,213,670	857,250	350
Stage 2	7,013	261,180	207,840	450
Space Station Rendezvous	150	30,240	1,010	
Deorbit bun	152	29,230	990	450
ΔV available from LST main engines	2,112	30,240	11,500	450
Total Propellant			1,076,590	

Notes: 1. See appendix E for calculations.
2. Source for specific impulse values: Space Mission Analysis and Design.

EARTH REENTRY THRUST REQUIREMENTS

US Space Shuttle deorbit burns typically require 100 to 500 ft/s. It was assumed that the LST would require a 500 ft/s (152 m/s) deorbit burn. Calculations are presented in appendix E for propellant requirements.

INITIAL PROPULSION REQUIREMENTS (LUNAR PROFILE)

CONCEPT OF OPERATIONS

The basic concept of the LST lunar mission is to use a three-stage rocket system similar to Saturn V to place the LST into a translunar orbit. The LST will use its primary engines for the insertion into a low lunar orbit and then eventually for the lunar landing.

The LST will not carry additional fuel for a launch from the moon. It is assumed that an energy source such as hydrogen will be available at the lunar site. The LST is designed to carry sufficient fuel to bypass a lunar landing and return to earth.

The Saturn V was capable of delivering 118,000 kg into a LEO and 47,000 kg into a translunar orbit (reference 9). Calculations are presented in appendix E detailing staging requirements for the LST. The spreadsheet calculations were also performed using the Saturn V basic data to determine if the calculations provided a reliable rough estimate for staging calculations. The Saturn V staging parameters are presented in table D-13 and the LST lunar staging parameters are presented in table D-14.

The LST will be equipped with an attitude maintenance system (AMS) similar to the US Space Shuttle.

Table D-13: Saturn V Staging

Stage	Propellant Mass[1] (kg)	Propellant (Oxidizer)[1]	Calculated Weight using spreadsheet	I_{sp}[2] (s)
One	2,077,000	RP-1 Hydrocarbon (Liquid Oxygen)	2,072,653	350
Two	427,300	Liquid Hydrogen (Liquid Oxygen)	719,783	450
Three	105,200	Liquid Hydrogen (Liquid Oxygen)	49,455	450

Notes: 1. Source: http://users.commkey.net/Braeunig/space/specs.htm
2. Source: Space Analysis and Design.
3. Stage III had two burns. Initial burn aided in insertion into LEO. Second burn placed vehicle into translunar orbit.

Table D-14: LST Lunar Profile Staging

Stage	Propellant Mass[1] (kg)	Propellant (Oxidizer)	I_{sp}[2] (s)
One	1,480,100	RP-1 Hydrocarbon (Liquid Oxygen)	350
Two	421,450	Liquid Hydrogen (Liquid Oxygen)	450
Three	34,840	Liquid Hydrogen (Liquid Oxygen)	450
LST Main Engines	13,940	Liquid Hydrogen (Liquid Oxygen)	450
LST Attitude Maintenance System	1,100	NTO/MMH	289

Notes: 1. See appendix E for calculations
2. Source for specific impulse values: Space Analysis and Design.

ΔV BUDGET/ PROPELLANT REQUIREMENTS AND BUDGET

The lunar mission ΔV budget is presented in table D-15. A 10 percent “ΔV” margin was established for the LST launch and a 15 percent “ΔV” margin was established for the lunar landing.

Stages 1 and 2 will be used for insertion into LEO and the LST’s main engines will be used for orbital phasing and return to earth. Stage 3 will insert the LST into a translunar orbit. The LST main engines will be used for insertion into the low lunar orbit and the lunar landing.

A lunar propellant budget was calculated using the “ΔV” budget. Calculations are presented in appendix E and the propellant budget is presented in table D-16.

A lunar return mission propellant budget was calculated with the assumption that the starting mass of the LST would be 27,740 kilograms. Once at the lunar base, the internal payload would be offloaded and the LST would then be refueled. The LST would rely only on its main engines for the return to earth.

The LST will transfer from the low lunar orbit to a return orbit to the earth. The LST will rely on aero-braking to enter a LEO. The LST main engines will provide the deorbit burn to reenter the atmosphere. The lunar return mission propellant budget is presented in table D-17.

Table D-15: Lunar Profile ΔV Budget

Item	ΔV requirements[1] (m/s)	Stage Providing ΔV
Burn 1 (earth surface to LEO)	9,700	Stage 1 and Stage 2
10 % margin for LEO insertion	970	Stage 1 and Stage 2
Burn 2 (LEO to C3 transfer orbit)	3,107	Stage 3
Burn 3 (C3 to low lunar orbit)	700	LST Main Engines
Burn 4 (low lunar orbit to lunar surface)	1,600	LST Main Engines
15 % margin for lunar landing	345	LST Main Engines
Total[2] ΔV	16,420	

Notes: 1. Source: http://www.pma.caltech.edu/~chirata/deltav.html
2. Total ΔV does not include attitude maintenance.

Table D-16: Lunar Profile Propellant Budget

Item	Stage ΔV (m/s)	Initial Mass (kg)	Propellant Mass (kg)	Stage I_{sp}[2] (s)
Stage 1	3895	2,181,790	1,480,100	350
Stage 2	6775	537,230	421,450	450
LEO to C3 orbit – Stage 3	3107	68,960	34,840	450
C3 to lunar orbit	700	30,240	4,430	450
Lunar orbit to lunar surface	1600	25,810	7,850	450
15 % margin for Δ V for lunar landing	345	17,960	1,350	450
Remaining LST propellant			310	
Total Propellant			1,950,330	

Notes: 1. See appendix E for calculations.
2. Source for specific impulse values: Space Mission Analysis and Design.

Table D-17: Lunar Return Profile Propellant Budget

Item	Stage ΔV (m/s)	Initial Mass[1] (kg)	Propellant Mass[1] (kg)	Stage I_{sp}[2] (s)
Launch to a low lunar orbit	1,600	27,740	8,430	450
10 % margin	160	19,306	690	450
Low lunar orbit to C3 transfer orbit	700	18,619	2,730	450
Aero-braking (establish earth orbit)	0	15,889	0	N/A
Deorbit burn for reentry	152	15,889	540	450
Remaining propellant			1,550	
Total Propellant			12,390	

Notes: 1. See appendix E for description of calculations.
2. Source for specific impulse values: Space Mission Analysis and Design.

LUNAR LANDING THRUST REQUIREMENTS

The lunar landing will be accomplished using the LST main engines in combination with the attitude maintenance system (AMS). The LST will approach the lunar surface with the nose of the vehicle pointed vertically away from the surface. The engines will be used to control the rate of descent. Just prior to landing the vehicle will rotate or pitch nose down to land in a flat attitude. The AMS rockets will provide the pitching moment. Prior to landing the lunar skids will deploy.

The LST was designed with four main engines to provide stability during the vertical lunar landing.

The lunar landing thrust requirements were the primary factor in sizing the LST main engines. The main engines had to provide an adequate thrust to weight ratio during the lunar descent. To estimate the LST engine mass and thrust requirements, the US Space Shuttle and Apollo Lunar Descent vehicles were used. The US Space Shuttle main engine and Apollo Lunar Descent vehicle engine parameters are presented in table D-18.

VEHICLE ATTITUDE MAINTENANCE AND CONTROL

An attitude maneuvering system was required for the following tasks:

1. Docking and rendezvous with the ISS
2. Orienting the vehicle prior to and during rocket burns
3. Orienting the vehicle for communications
4. Provide attitude control during final portion of lunar landing

A system similar the US Space Shuttle will be used to fulfill these requirements. Thrusters will be positioned to ensure precise and proper control of the vehicle is possible. The system will use NTO/MMH as the propellant. Using the space shuttle as a reference, the propellant requirements for the LST were approximated to be 1,100

Table D-18: Lunar Profile Thrust Requirement

Space Craft	Number of Engines	Single Engine Mass (kg)	Thrust per engine (kN)	Lunar Thrust to Weight Ratio (minimum)	Space Craft Mass[1] (kg)
US Space Shuttle (main engines)	3	3,180[2]	2,090[1]	N/A	91,000
US Space Shuttle (Orbital engines)	2	118[2]	26.7[1]	N/A	91,000
Lunar Descent Module (with Ascent module)	1	N/A	43.90	1.8	14,696
LST	4	95[3]	21.9[3]	1.8	30,000

Notes: 1. Source: http://users.commkey.net/Braeunig/space/specs.htm
2. Source: Shuttle Crew Operations Manual.
3. See calculations in appendix E.

Table D-19: Attitude Maintenance Propulsion System Propellant Requirements

Item	Vehicle On Orbit Mass (kg)	Propellant Mass (kg)	Specific Impulse
US Space Shuttle Reaction Control System[1]	91,000	3,300	289 s
Lunar Shuttle Transport	30,000	1,100	289 s

Note: 1. Source: http://users.commkey.net/Braeunig/space/specs.htm

kilograms. The estimate was made by assuming the LST would require approximately one third of the space shuttle's propellant mass because the mass of the LST was approximately one third that of the space shuttle. A comparison of the two vehicles is presented in table D-19.

EXTERNAL VEHICLE LAYOUT

AERODYNAMIC BRAKING DRAG

The primary reason the LST is a delta wing/lifting body shape is to increase the aerodynamic braking of the vehicle during re-entry. Acceleration forces on re-entry could be significantly reduced and the crew will also have an alternative if the parachute system fails. The vehicle can be flown to the desert floor in CA and land on the lunar skids using the attitude maintenance system to provide additionally deceleration prior to landing.

LIFTING BODY AIRFOILS

The vehicle structure of the LST was modeled after the US Space Shuttle and proposed lifting body designs similar to the Assured Crew Return Vehicle. This was done to take advantage of aerodynamic braking upon reentry into earth's atmosphere.

One of the primary disadvantages of the choice was the location of the engines nozzles during lunar landings. This forced a vertical landing profile during the descent to the moon.

PARACHUTES

For earth landings, aerodynamic braking will be used for the initial phase. As the vehicle approaches a slower speed regime, the vehicle will rely on parachutes for the final phase prior to landing. This would be accomplished much in the same manner as the Russian Soyuz capsules and the Apollo capsules. Since the LST is a much larger vehicle than the Soyuz Descent module, it will use a system of three parachutes similar to the Apollo capsule. The descent rate of the LST will be 8 meters per second similar to the Soyuz module.

The primary parachute system specifications are presented in table D-20. The calculations are presented in appendix E.

Table D-20: Soyuz and LST Parachute Systems

Vehicle	Landing Mass (kg)	Descent Rate (m/s)	Parachute C_D	Parachute Radius[3] (m)
Soyuz Descent Module[1]	2,850	8	0.7	17.75
Lunar Shuttle Transport	17,000	8	0.7[2]	24.7

Notes: 1. Source: http://users.commkey.net/Braeunig/space/specs.htm
2. C_D was assumed to be the same as that calculated for Soyuz system. See appendix E.
3. Soyuz has one parachute. The LST will have three parachutes.

Table D-21: US Space Shuttle and LST Airlock Parameters

Vehicle	Airlock Mass (kg)	Airlock Hatch Mass (kg)	Internal Diameter (m)	Hatch Diameter (m)	EVA Capable
US Space Shuttle[1]	374	33	1.6	1.0	Yes
Lunar Shuttle Transport	146	33	1.2	1.0	No

Note: 1. Source: http://www.asi.org/adb/06/07/04/10/airlock-pictures.html

The parachute system will be augmented with the vehicle attitude control system prior to landing. The vehicle attitude system (AMS) will fire to cushion the landing. The parachute system is located on the upper surface of the LST aft of the airlock and just forward of the center of gravity. Locating the parachute attach points just forward of the center of gravity will provide a slightly nose-up attitude during the landing. This will increase safety if the LST has a forward velocity at impact with the surface.

DOCKING REQUIREMENTS

The LST is designed to dock with the ISS and a future lunar base. The LST airlock will be used to equalize pressure between the ISS and the LST cabin. The airlock opening is positioned on the upper surface of the LST. The LST attitude maneuvering system will be used for control during the final phase of the rendezvous. The docking procedure will be controlled from the pilot station using a system of cameras and hand controls to manipulate the attitude maneuvering system.

As a reference the space shuttle airlock parameters are compared to the LST airlock parameters in table D-21. Unlike the space shuttle airlock, the LST air lock is not designed to allow astronauts to depressurize prior to space walks. Therefore, the LST airlock volume will not accommodate a crewmember. The LST airlock estimated total mass was 146 kg. The two hatches were 33 kg and the remaining structure was estimated to be 80 kilograms.

HEAT/RADIATION REQUIREMENTS

The LST is designed with fibrous refractory composite insulation tiles.[5] The tiles on the lower surface and vertical tail weigh 12 lb/ft^3. The upper surface tiles weigh 9 lb/ft^3. The average thickness of the tiles is approximately 3 inches. The leading edges of the LST are covered with a reinforced carbon-carbon material. The US Space Shuttle was used to determine likely requirements for the LST.

WING SIZING

Wing sizing was estimated by making visual comparisons with similar vehicles such as the US Space Shuttle. The initial wing area approximation was 72 square meters. This was found to be excessive for the mission requirements. The LST wing was required to allow for deceleration to 200 KTAS during the landing phase on earth. See appendix E for calculations. The wing was sized to provide a stall speed of 180 KTAS at 4,000 MSL for a vehicle with a mass of 20,000 kilograms. The second estimate for the wing area was 54 square meters. The estimated angle of attack for this wing shape was determined to be 15 degrees. After evaluating data for sample airfoils, it was determined that an angle of attack of 15 degrees would likely be above stall angle of attack for the wing. Therefore, the wingspan and area was increased to lower the coefficient of lift required at 180 KTAS and to lower the angle of attack at 180 KTAS. The three wing sizing choice parameters are presented in table D-22. Figure D-1 is an illustration of the wing design choices.

VERTICAL TAIL SIZING

The vertical tail was sized using the wing area, wingspan, and approximate moment arm for the vertical tail. The vertical tail sizing calculations are presented in appendix E. The required area for the vertical tail was determined to be 12.5 square meters using a method presented in Aircraft Design: A Conceptual Approach (p 124-125). Typical tail volume coefficients were used for a "jet fighter."

A visual comparison was made between the US Space Shuttle and the LST comparing vertical tail area to wing surface area. This was performed to determine if the calculated vertical tail area of 12.5 m^2 would likely be sufficient. An illustration of this comparison is presented in figure D-2. The LST vertical tail area to wing surface area ratio was greater than the space shuttle ratio. Therefore it was concluded that 12.5 m^2 would be sufficient.

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[5] Source: http://science.ksc.nasa.gov.

Table D-22: Wing Iteration Choices

Wing Design Choice	Span (m)	Area (m^2)	Aspect Ratio	Estimated AOA at 180 KTAS	$C_{l\alpha}$ (/radian)	C_L at 180 KTAS
A	13.3	72	2.5	7°	3.46	0.55
B	9.6	52	1.8	15°	2.95	0.76
C / Final	11	60	2.0	12°	3.15	0.66

Note: See appendix E for sample calculations.

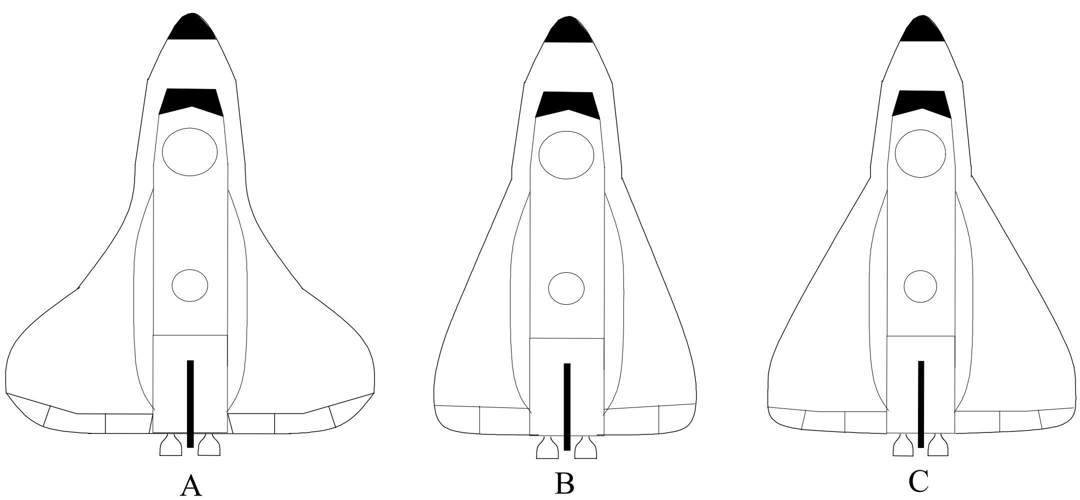

Figure D-1: LST Wing Design Sequence

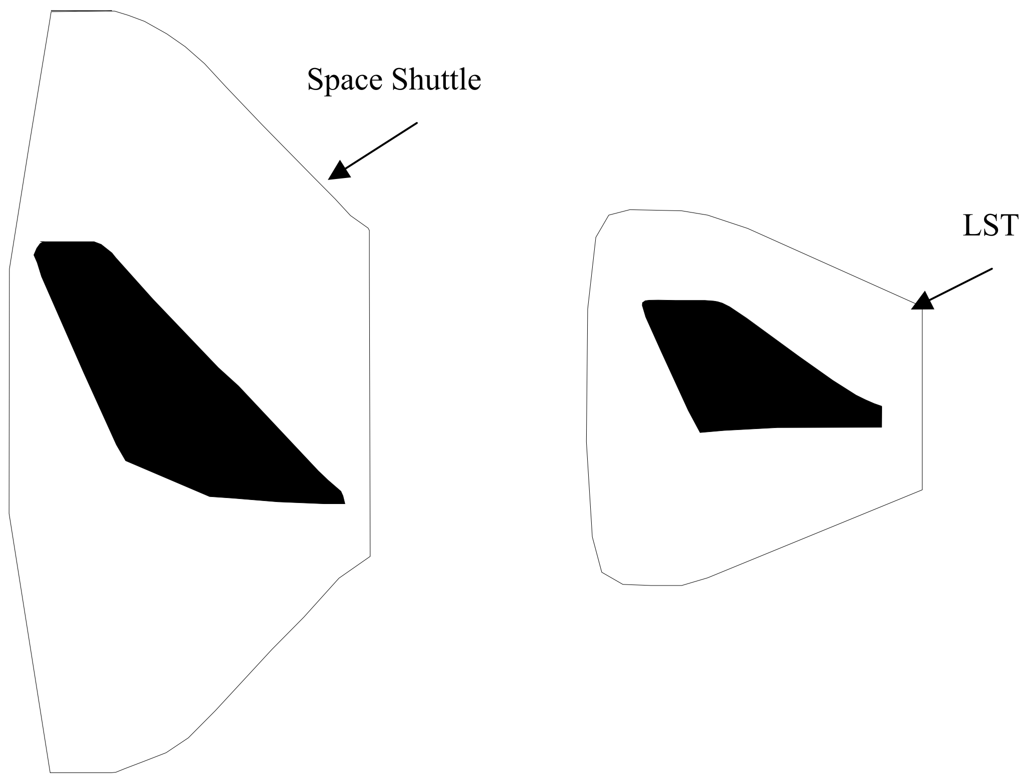

Figure D-2: LST and Space Shuttle Vertical Tail Area to Wing Area Comparison

INTERNAL VEHICLE LAYOUT

CREW ACCOMMODATION/INTERFACE

The LST is designed to carry a crew of eight. Two crew are designated as pilots and the remaining crew are designated as passengers. An illustration of crew seating layout, figure A-1, is presented in appendix A. The pilot seats are positioned in the forward portion of the crew cabin where they will control and monitor life support systems and vehicle parameters. The pilots will monitor the autopilot systems during launch and landing. The pilots will have controls at each seat to maneuver the LST during the landing phases should the primary autopilot system fail. The pilots will maneuver the vehicle during the rendezvous and docking phase with the space station.

The LST habitable volume is approximately 45.3 cubic meters. Sufficient habitable volume was designed into the LST to provide for a relatively comfortable environment. Other space vehicles designs were used as a guide in determining the habitable volume of the LST (see table D-23). The mission duration and crew responsibilities were used to determine the requirements. For example the US Space Shuttle internal volume is 71.50 cubic meters. The crew is highly involved in carrying out science experiments and the mission may last up to 16 days. The LST crew will not be involved in executing science experiments and nominal missions should last only 3 to 7 days.

The passenger seats will fold into the floor once the LST is in a micro-gravity environment. During launch, the crew will be seated such that the vertical acceleration will be from front to back. The crew will experience an approximate transverse g-force of 3 g's. The crew seats will be stroking seats to prevent injury during hard landings on either the earth or moon.

Table D-23: Internal Volume Requirements for Crew

Vehicle	Design Life (days)	Volume (m^3)	Number of Crew
Space Shuttle Columbia[1]	9-16	71.50	7
HL-20 PLS[2]	3	16.40	8
Soyuz VI[2]	3	11.00	3
Apollo Lunar Module[2]	10	6.65	2
Apollo Command Module[2]	10	6.17	3
Lunar Shuttle Transport	3-11	45.3	8

Notes: 1. Source: http://users.commkey.net/Braeunig/space/specs.htm
2. Source: http://www.friends-partners.ru/partners/mwade/craft/hl42.htm

Table D-24: LST Main Propellant Internal Volume Requirements

Profile	Propellant Mass (kg)	Propellant Volume (m^3)
Lunar Profile	13,940	40
Space Station Profile	11,500	33

PROPELLANT VOLUME REQUIREMENTS

To perform the lunar mission, the LST is required to carry 13,940 kg of propellant. Since the LST and US Space Shuttle are both designed to use a liquid hydrogen and oxygen mixture, the space shuttle external tank was used as a reference to estimate the volume of the LST's propellant. The LST propellant volume requirements are presented in table D-24. Calculations are presented in appendix E.

The attitude maneuvering system (AMS) propellant mass is 1,100 kg. The AMS system uses NTO/MMH. The volume required for the AMS propellant was calculated to be 0.96 cubic meters. Calculations are presented in appendix E.

PAYLOAD ARRANGEMENT

The internal payload can be arranged in two different configurations. The payload is located below the cabin floor and in the aft portion of the cabin. In the space station configuration, a propellant pallet is removed for the additional payload.

The payload is limited in size by the internal diameter of the airlock. All portions of the payload need to be loaded and removed via the airlock. Illustrations of the payload arrangement, figures A-5 and A-8, are presented in appendix A.

FLIGHT CONTROL SYSTEMS

The LST flight control system consists of two redundant flight control computers, trailing edge flaps, ailerons, and a rudder on the vertical tail. The control surfaces are "fly-by-wire" and also use lightweight electric actuators. Electric actuators were chosen to minimize weight and remove the requirement for a high-maintenance and heavy hydraulic system.

The flight controls can be manually controlled through the fight control computers by using a side-stick located on the right console at both pilot stations. The flight control computers measure stick force and displacement to determine the desired control surface deflections. To reduce weight no rudder pedals are provided to the pilots. The pilots can control vehicle pitch, roll, and yaw with the side-stick controllers. The flight control computers provide coordinated rudder deflections.

During the docking maneuver, the side-stick is used with a translation hand controller at the pilot station. The side-stick and translation hand controllers use the AMS system to control the attitude and position of the LST. The side-stick is used to

control pitch, roll and yaw. The translation hand controller is used to control the LST in the x, y, and z axes.

STABILITY

LAUNCH ENVIRONMENT

During the launch and insertion into the low earth orbit, the LST engines and attitude maintenance systems will not be used. Primary attitude and stability will be controlled using the lower stages and nozzle positioning.

During the lunar launch the main engines will provide the required thrust as well as the attitude control. The LST main engines thrust angles can be varied to provide stability during launch and orbital maneuvers.

SPACE ENVIRONMENT

LST attitude control during orbital transfer maneuvers, space station docking, de-orbit burns and general attitude maintenance will be maintained by the Attitude Maintenance System (AMS).

EARTH LANDING

The LST recovery is designed with three primary stages. Following reentry, the LST will decelerate to approximately 200 KTAS. At approximately 15,000 MSL, the LST parachute system will deploy slowing the descent rate to 8 meters per second. Just prior to touchdown on the skids, the AMS (attitude maintenance system) will perform a coordinated burn to cushion the landing. See figure A-6 in appendix A for an illustration of the landing phases.

LUNAR LANDING

An illustration of the lunar landing profile, figure A-7, is presented in appendix A. The LST main engines will provide the thrust required for a vertical approach to the landing surface. The main engines are designed to provide 89.5 kN of thrust. During the initial descent the thrust to weight ratio will be greater than 1.8 and will continue to increase as propellant is burned. As the LST nears the surface, the craft will be rotated using a combination of the AMS jets and the main engines. The LST will be landed on the deployed skids. Calculations for the main engine requirements for the lunar landing are presented in appendix E.

REFINED SIZING ESTIMATES

The LST mass estimate calculations are presented in table E-7 in appendix E. The methods used to estimate the individual component masses are presented in appendix E as well. A list of the vehicle mass estimates is presented in table D-25.

The on-orbit mass for the lunar profile was calculated to be 30,241 kilograms. The on-orbit mass for the space station was calculated to be 29,801 kilograms. The lunar return launch mass was calculated to be 27,741 kilograms.

REFINED PROPULSION ESTIMATES

Following the final vehicle mass estimation, the propellant requirements were recalculated. The final calculations are presented in tables E-2, 3, 5, and 6 in appendix E.

Table D-25: LST Vehicle Mass Estimates

Item	mass (kg)
Air volume	54
Aircrew	640
Aircrew flight gear	80
Aircrew seats	120
Airlock	146
AMS propellant	1100
Attitude maintenance system (AMS)	165
Avionics weight	490
Clamps and miscellaneous structure	337
Cooling/insulation components for propellant	500
Crew life support system	1500
Electric or hydraulic actuators for flight controls	137
Electrical	326
Electronic systems (sensors, computers, etc)	200
Engine systems	50
Firewall	14
Flight control system	254
Food	200
Fuselage	1162
Landing skids	264
Lockers for flight crew gear/food	14
Main engine propellant (ISS Mission)	11500
Main engine propellant (Lunar Mission)	13940
Main engines	380
Parachute system	612
Payload for ISS mission	4500
Payload for lunar mission	2500
Payload for lunar return mission	0
Tail	446
Tiles (lower surface)	1350
Tiles (tail)	380
Tiles (upper surface)	1010
Toilet	20
Windows	100
Wings	1760
Lunar Mission Total Mass	**30241**
Space Station Mission Total Mass	**29801**
Lunar Return Trip Mission Total Mass	**27741**

APPENDIX E: PERFORMANCE CALCULATIONS

The following calculations or sample calculations are contained in this appendix:

1. Lunar Transfer Orbit Calculations
2. LEO Insertion ΔV Requirements
3. Space Station Plane Change Calculations
4. Staging Calculations
5. Parachute Calculations
6. Thrust Requirements for Lunar Landing
7. Thrust Requirements for Earth Reentry
8. Wing Aerodynamic Requirements for Earth Landing
9. Tail aerodynamic Requirements for Earth Landing
10. LST Mass Estimates
11. Specific Mass Calculations
12. LST Internal Propellant Volume Requirements
13. AMS System Propellant Volume Requirements

1. LUNAR TRANSFER ORBIT CALCULATIONS

Variables

R_1	low earth orbit radius (6750 km)
R_2	radius 2 of transfer orbit
Mean distance from earth to moon	384,400 km
μ_E	3.986012×10^5 km^3/s^2
mean radius of moon	1738 km
altitude for low lunar orbit	50 km
a_t	semi-major axis of transfer orbit
TOF	time of flight
E_t	orbit energy
Vcs_1	orbital velocity of LEO
$V_{mooncircular}$	orbital velocity of LLO
ΔV	change in velocity
V_1	orbital velocity of low earth orbit
V_2	orbital velocity of low lunar orbit

Assumption:

- The following calculations were performed to obtain an estimate of the ΔV requirements. The LST will not perform a simple Hohmann transfer for the mission. A C3 orbit similar to the Apollo mission will be used.
- Two body orbital problem (not three body problem).

Find:

- ΔV requirements for a Hohmann transfer and compare to values from other sources for C3 orbit (ΔV_1 = 3.107 km/s and ΔV_2 = 0.7 km/s)
- Time of flight to moon

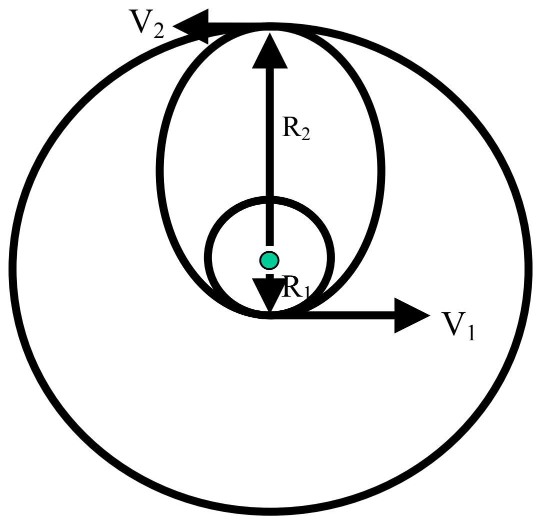

Figure E-1: Hohmann Transfer Orbit Illustration

<u>*Calculations*</u>:[6]

R_2 = Mean distance from earth to moon + mean radius of moon + altitude of low lunar orbit

R_2 = 386,188 km

$$E_t = \frac{-\mu_E}{R_1 + R_2}$$ $E_t = -1.0144\ km^2/s^2$

$$V_1 = \sqrt{2[\frac{\mu_E}{R_1} + E_t]}$$ $V_1 = 10.77\ km/s$

$\Delta V_1 = V_1 - Vcs_1$ $$\Delta V_1 = V_1 - \sqrt{\frac{\mu_E}{R_1}}$$ $\Delta V_1 = 3.09$ km/s

$$V_2 = \sqrt{2[\frac{\mu_E}{R_2} + E_t]}$$ $V_2 = 2.023$ km/s

$$V_{mooncircular} = \sqrt{\frac{\mu_M}{R_{mooncircular}}}$$ $V_{mooncircular} = 2.74$ km/s

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[6] Equation source for Hohmann transfer: <u>Fundamentals of Astrodynamics</u>, p. 163-166.

Table E-1: ΔV Requirements for Lunar Missions

Event	C3 orbit (from source)	Hohmann Transfer Calculations
ΔV for transfer from LEO to Lunar Transfer Orbit	3.107 km/s[1]	3.09 km/s
ΔV for transfer from lunar transfer orbit to low lunar orbit	0.7 km/s[2]	0.717 km/s
Time of flight for LEO to low lunar orbit	73 hours[3]	120.4 hours

Notes: 1. Source: http://spacecraft.ssl.umd.edu
2. Source: http://www.pma.caltech.edu/~chirata/deltav.html
3. Source: http://visitor.broaddaylight.com/spacekids/FAQ 44 359.shtm

$\Delta V_2 = V_{mooncircular} - V_2$ $\qquad \Delta V_2 = 0.717$ km/s

Calculate time for orbital transfer from LEO to lunar orbit (half time of transfer orbit)

$$a_t = \frac{R_1 + R_2}{2} = 196,469 km \qquad TOF = \pi \sqrt{\frac{a_t^{\,3}}{\mu_E}} = 120.37 hours$$

---

2. LEO INSERTION ΔV REQUIREMENTS

Variables

R	low earth orbit radius (6630 km)
μ_E	3.986012×10^5 km^3/s^2
ΔV_{des}	ΔV design
$\Delta V_{burnout}$	velocity at burnout (orbital velocity)
$\Delta V_{gravity}$	ΔV loss due to gravity
ΔV_{drag}	ΔV loss due to drag
V_1	orbital velocity of low earth orbit

Assumption:

- ΔV_{drag} is 3 % of $\Delta V_{burnout}$ [7]
- Drag and gravity losses are similar to the US Space Shuttle and Titan IV launch systems.
- $\Delta V_{gravity}$ is 1300 m/s [8]

[7] Source for assumption: Space Mission Analysis and Design (p. 669). "For the current inventory of large, expendable launch vehicles, velocity losses due to drag are less than 3% of the total change in velocity required, . . . "

Find:

- Velocity of LEO (V)
- Approximate ΔV requirement for LEO insertion and compare to referenced values.
- Compare with referenced value of 9.7 km/s[9]

Calculations:[10]

$$V_{burnout} = V = \sqrt{\frac{\mu_E}{R}} = 7.75\frac{km}{s}$$

$$\Delta V_{drag} = 0.03V_{burnout} = 0.03(7.75\frac{km}{s}) = 0.23\frac{km}{s}$$

$$\Delta V_{des} = \Delta V_{burnout} + \Delta V_{gravity} + \Delta V_{drag}$$

$$\Delta V_{des} = (7.75 + 1.3 + 0.23)\frac{km}{s} = 9.28\frac{km}{s}$$

Conclusions:

9.7 km/s was used as the ΔV requirement for the staging calculations. The above calculations showed that this should be sufficient to insert the LST into a LEO. Additionally, a 10 percent margin for propellant mass was added to account for inefficiencies and unburned propellant.

3. SPACE STATION PLANE CHANGE CALCULATIONS

Variables

R_1	6750 km
μ_E	3.986012×10^5 km^3/s^2
i_{init}	28.5 degrees (initial orbit inclination-KSC)
i_{final}	51.8 degrees (ISS orbit inclination)
θ	plane change angle
V	orbital velocity
ΔV for LEO insertion	9.7 km/s

[8] Source for assumption: Space Mission Analysis and Design (p. 668). ". . . for medium-to-large launch vehicles on nominal trajectories the velocity losses due to gravity fall between 750 to 1500 m/s."
[9] Source for ΔV requirement: http://www.pma.caltech.edu/~chirata/deltav.html.
[10] Source for equations: Space Mission Analysis and Design and Fundamentals of Astrodynamics.

Find:

- Plane change ΔV requirement if LST is established in orbit with inclination of 28.5 degrees
- Determine DV requirement to launch directly into 51.8 degree inclination orbit

Calculations:

$$\Delta V = 2V\sin\frac{\theta}{2} \qquad \Delta V = 2[\frac{\mu_E}{R_1}]^{\frac{1}{2}}\sin\frac{i_{final} - i_{init}}{2} \qquad \Delta V = 3.10 km/s$$

$$\Delta V_{direct} = \Delta V^2{}_{LEO} + \Delta V^2{}_{planechange} \qquad \Delta V_{direct} = 10.2 km/s$$

4. STAGING CALCULATIONS

A spreadsheet was developed using the basic rocket equation to determine propellant mass requirements, optimum staging, and stage mass requirements. The rocket equation used in the calculations was:

$$\Delta V = g_o I_{sp}\ln[\frac{W_o}{W_f}]$$

"ΔV" is the required change in velocity. "g_o" is a constant (9.81 m/s^2). "I_{sp}" is the specific impulse of the propellant (units seconds). "W_o" is the initial weight of the stage. "W_f" is the final weight of the stage after the propellant has been consumed.

For staging calculations the assumption was made that 10 percent of the masses of stages 1, 2, and 3 was structure. The remaining 90 % of the mass was assumed to be propellant[11].

The calculations used a specific impulse of 350 seconds for the first stage in both profiles. The second and third stages and the LST main engines used a specific impulse of 450 seconds. It was assumed that stage 1 would use RP-1 hydrocarbon as a propellant and stages 2,3 and the LST would use liquid hydrogen and oxygen.

The Saturn V staging was calculated to verify that the spreadsheet could be used as an effective tool. The spreadsheet results are presented in appendix D in table D-12.

Samples of the spreadsheet calculations are presented in tables E-2 through E-6. The following is a description of the data in tables E-2 through E-6.

Table E-2: Space Station ΔV and Propellant Budget Calculations
Optimum space station staging was determined as well as required propellant weights. It was assumed that a two-stage rocket system would place the LST into a low earth orbit in the same plane as the International Space Station. A 10 percent ΔV margin requirement was added which resulted in a ΔV requirement of 11,220 m/s.

[11] Source for assumptions and equations: Logsdon, Tom. Orbital Mechanics: Theory and Applications. (p. 117).

Table E-3: Lunar ΔV and Propellant Budget Calculations

Optimum lunar staging was determined as well as required propellant weights. It was assumed that a two-stage rocket system would place the LST and a third stage (translunar insertion stage) into a low earth orbit. A 10 percent ΔV margin requirement was added which resulted in a ΔV requirement of 10,670 m/s. A 15 percent ΔV margin requirement was added for the lunar landing phase.

Table E-4: Lunar Return ΔV and Propellant Budget Calculations

Propellant and ΔV requirements for a return from the moon to the earth were calculated. The initial mass of the LST on launch is 27,740 kg. This is without the internal payload of 2,500 kg.

Table E-5: Spreadsheet Sample Calculations

Sample rocket equation calculations are presented in table E-4.

Table E-6: Space Station Profile Optimum Staging Calculations

A portion of the space station staging calculations is presented in table E-5.

Table E-7: Lunar Profile Optimum Staging Calculations

A portion of the lunar staging calculations is presented in table E-6.

Table E-2: Space Station ΔV and Propellant Budget Calculations

Space Station Calculations ---ΔV / Propellant Budget						
Item/Event	**Requirement m/s**	**Initial Mass kg**	**Final Mass kg**	**Stage Mass kg**	**Propellant Mass kg**	**I_{sp} s**
LEO Insertion	10,200					
10 % margin for ΔV	1,020					
Stage 1	4,207	1,213,673	356,426	952,497	857,247	350
Stage 2	7,013	261,176	53,334	230,936	207,843	450
Space station rendezvous	150	30,240	29,230	N/A	1,010	450
Deorbit burn	152	29,230	28,241	N/A	989	450
ΔV available from LST main engines	2,112	30,240	18,740		11,500	450
Total at Launch		1,213,673			1,076,590	

Table E-3: Lunar ΔV and Propellant Budget Calculations

Lunar Calculations ---ΔV / Propellant Budget						
Item/Event	**Requirement m/s**	**Initial Mass kg**	**Final Mass kg**	**Stage Mass kg**	**Propellant Mass kg**	**I_{sp} s**
LEO Insertion	9,700					
10 % margin for ΔV	970					
Stage 1	3,895	2,181,787	701,688	1,644,554	1,480,099	350
Stage 2	6,775	537,232	115,783	468,277	421,450	450
LEO to C3 Orbit (stage 3)	3,107	68,955	34,111	38,715	34,844	450
C3 to Lunar Orbit	700	30,240	25,806	N/A	4,434	450
Lunar Orbit to Lunar Surface	1,600	25,806	17,960	N/A	7,846	450
15 % margin for ΔV	345	17,960	16,610	N/A	1,350	450
Total for LST	2,760	30,240	16,610	N/A	13,630	
Remaining propellant					310	450
Total at Launch		2,181,787			1,950,333	

Table E-4: Lunar Return ΔV and Propellant Budget Calculations

Lunar Calculations ---ΔV / Propellant Budget (Return Trip)						
Item/Event	**Requirement** **m/s**	**Initial Mass** **kg**	**Final Mass** **kg**	**Stage Mass** **kg**	**Propellant Mass** **kg**	**I_{sp}** **s**
Lunar Surface to LLO	1,600	27,740	19,306	N/A	8,434	450
10 % margin for ΔV	160	19,306	18,619	N/A	687	450
LLO to C3 Orbit	700	18,619	15,889	N/A	2,730	450
Aero-braking	0	15,889	15,889	N/A	0	
Deorbit burn	152	15,889	15,351	N/A	538	450
Total	2,612				12,389	
Remaining propellant					1,551	

Table E-5: Spreadsheet Sample Calculations

Rocket Equation

Given:

I_{sp}	W_{final}	$W_{initial}$	g_o
311	2,183	14696	9.81

Find:

ΔV
5818

Given:

I_{sp}	$W_{initial}$	ΔV	g_o
311	14696	2470	9.81

Find:

W_{final}
6540

Given:

I_{sp}	W_{final}	ΔV	g_o
311	25,000	3107	9.81

Find:

$W_{initial}$
69218

Given:

I_{sp}	Stage Weight	ΔV	g_o
439	34000	3107	9.81

Assume:

Boosting Stage has a 10 % weight penalty

$W_{initial}/W_{final}$ **= ($W_{boosting\ stage}$ + stage weight)/(stage weight + 0.1*$W_{boosting\ stage}$)**

Find:

$W_{initial}$	W_{final}	$W_{boosting\ Stage}$
79265	38527	45265

stage weight is what we want to accelerate

Table E-6: Space Station Profile Optimum Staging Calculations

Given:	Δ V Total	Stage Weight	$I_{sp\text{-stage }2}$	$I_{sp\text{-stage }1}$	g_o	step
	11220	30240	450	350	9.81	0.005

Δ V Contribution Stage 2	Stage 2 Weight	Wfinal (LST+Stage 3+Empty Stage 2)	Winitial (Stage 2,3,LST)	Δ V Contribution Stage 1	Stage 1 Weight	Wfinal (LST+Stage 3,2+Empty Stage 1)	Winitial (Stage 1,2,3,LST)
5891	136424	43882	166664	5329	1174862	284150	1341526
5947	139888	44229	170128	5273	1157839	285912	1327967
6003	143452	44585	173692	5217	1141637	287856	1315330
6059	147121	44952	177361	5161	1126210	289982	1303572
6115	150899	45330	181139	5105	1111516	292291	1292655
6172	154791	45719	185031	5049	1097516	294782	1282547
6228	158801	46120	189041	4992	1084174	297458	1273215
6284	162935	46533	193175	4936	1071459	300320	1264634
6340	167197	46960	197437	4880	1059340	303371	1256778
6396	171595	47400	201835	4824	1047791	306614	1249626
6452	176134	47853	206374	4768	1036786	310052	1243160
6508	180820	48322	211060	4712	1026302	313690	1237362
6564	185661	48806	215901	4656	1016319	317532	1232220
6620	190663	49306	220903	4600	1006817	321585	1227720
6676	195835	49823	226075	4544	997779	325853	1223854
6733	201185	50358	231425	4487	989188	330344	1220613
6789	206721	50912	236961	4431	981031	335065	1217993
6845	212454	51485	242694	4375	973295	340024	1215989
6901	218394	52079	248634	4319	965967	345231	1214601
6957	224551	52695	254791	4263	959037	350694	1213828
7013	230936	53334	261176	4207	952497	356426	1213673
7069	237563	53996	267803	4151	946337	362437	1214141
7125	244445	54684	274685	4095	940552	368740	1215237
7181	251596	55400	281836	4039	935136	375349	1216972
7237	259031	56143	289271	3983	930083	382280	1219354
7294	266768	56917	297008	3926	925391	389547	1222399
7350	274825	57722	305065	3870	921057	397170	1226122
7406	283220	58562	313460	3814	917080	405168	1230540
7462	291975	59437	322215	3758	913460	413561	1235675
7518	301113	60351	331353	3702	910199	422373	1241552
7574	310659	61306	340899	3646	907298	431628	1248197
7630	320639	62304	350879	3590	904762	441355	1255641
7686	331084	63348	361324	3534	902596	451584	1263920
7742	342026	64443	372266	3478	900806	462347	1273072

Optimum staging

Table E-7: Lunar Profile Optimum Staging Calculations

Given:	Δ V Total	Stage Weight	$I_{sp\text{-stage 2}}$	$I_{sp\text{-stage 1}}$	g_o	step
	10670	68955	450	350	9.81	0.005

Δ V Contribution Stage 2	Stage 2 Weight	Wfinal (LST+Stage 3+Empty Stage 2)	Winitial (Stage 2,3,LST)	Δ V Contribution Stage 1	Stage 1 Weight	Wfinal (LST+Stage 3,2+Empty Stage 1)	Winitial (Stage 1,2,3,LST)
5708	286803	97635	355758	4962	2003634	556121	2359392
5761	293656	98321	362611	4909	1978553	560466	2341163
5815	300691	99024	369646	4855	1954436	565089	2324081
5868	307914	99746	376869	4802	1931236	569993	2308105
5921	315334	100488	384289	4749	1908910	575180	2293199
5975	322957	101251	391912	4695	1887418	580654	2279330
6028	330791	102034	399746	4642	1866721	586418	2266467
6081	338845	102840	407800	4589	1846784	592479	2254585
6135	347128	103668	416083	4535	1827576	598840	2243659
6188	355648	104520	424603	4482	1809064	605510	2233668
6242	364416	105397	433371	4429	1791222	612493	2224593
6295	373441	106299	442396	4375	1774022	619798	2216418
6348	382735	107228	451690	4322	1757439	627434	2209129
6402	392309	108186	461264	4268	1741451	635409	2202715
6455	402174	109172	471129	4215	1726036	643733	2197166
6508	412345	110190	481300	4162	1711174	652418	2192475
6562	422834	111238	491789	4108	1696847	661474	2188636
6615	433656	112321	502611	4055	1683037	670915	2185648
6668	444827	113438	513782	4002	1669728	680754	2183509
6722	456361	114591	525316	3948	1656905	691007	2182221
6775	468277	115783	537232	3895	1644554	701688	2181787
6828	480593	117014	549548	3842	1632664	712815	2182212
6882	493329	118288	562284	3788	1621222	724406	2183505
6935	506505	119605	575460	3735	1610217	736481	2185677
6988	520143	120969	589098	3682	1599640	749062	2188738
7042	534268	122382	603223	3628	1589483	762171	2192705
7095	548904	123845	617859	3575	1579737	775833	2197596
7148	564079	125363	633034	3522	1570396	790074	2203430
7202	579822	126937	648777	3468	1561453	804922	2210230
7255	596164	128571	665119	3415	1552905	820410	2218024
7309	613139	130269	682094	3361	1544746	836569	2226840
7362	630783	132033	699738	3308	1536974	853436	2236712
7415	649135	133869	718090	3255	1529586	871049	2247676
7469	668238	135779	737193	3201	1522581	889451	2259774

Optimum staging

5. PARACHUTE CALCULATIONS

Variables:

Soyuz Descent Module Mass	2,850 kg
LST landing mass	17,000 kg
Soyuz descent rate with primary chute	V = 8 m/s
g_o	gravity at earth's surface (9.81 m/s^2)
C_D	coefficient of drag
r	radius of parachute (Soyuz = 17.75 m)
ρ	density (sea level STD = 1.29 kg/m^3)

Assumptions:

- Drag is proportional to the area of the parachute opening disc area..
- Coefficient of drag is the same for the Soyuz and LST parachutes.
- LST will use a three parachutes similar to the Apollo capsule.

Find:

- Drag coefficient for Soyuz chute and use as a baseline to determine parachute requirements for LST.

Calculations:

$$Force = 0 = mg_o - C_D \frac{1}{2}\rho V^2 S = mg_o - C_D \frac{1}{2}\rho V^2 S\pi r^2$$

$$C_D = 0.7$$

Find:

Radius of three LST parachutes that will provide a 8 m/s descent rate.

Calculations:

$$Force = 0 = mg_o - C_D \frac{1}{2}\rho V^2 S = mg_o - C_D \frac{1}{2}\rho V^2 S 3\pi r^2$$

$$C_D = 0.7, m = 17,000kg, V = 8 \, {}^{m}\!/\!{}_{s}$$

$$r = 24.7m$$

6. THRUST REQUIREMENTS FOR LUNAR LANDING

Variables:

g	1.62 m/s^2 (surface gravity on moon)
thrust of Apollo Lunar Descent Module	43.90 kN
mass of Apollo Lunar Descent Module	14,696 kg
mass of US Space Shuttle OMS Engine	118 kg ($Mass_{SSOE}$)
thrust of US Space Shuttle OMS Engine	26.7 kN ($Thrust_{SSOE}$)
LST mass (starting)	30,000 kg
LSTME	LST main engine
$Thurst_{LST}$	LST main engine total thrust

Find:

- LST engine thrust requirements so that thrust to weight ratio is the same as Apollo Descent Module
- LST engine mass comparing US Space Shuttle as a guide

Assumptions:

LST will have four engines similar to US Space Shuttle (scaled down)

Calculations:

$$\frac{Thrust}{Weight} = \frac{Thrust}{(mass)(gravity)} = \frac{43.90kN}{(14{,}696kg)(1.62\,{}^{m}\!/\!{}_{s^2})} = \frac{Thrust_{LST}}{(30{,}000kg)(1.62\,{}^{m}\!/\!{}_{s^2})}$$

$$\frac{Thrust}{Weight} = 1.84 \qquad Thrust_{LST} = 89.6kN$$

$$\frac{Mass_{SSOE}}{Thrust_{SSOE}} = \frac{Mass_{LSTME}}{Thrust_{LSTME}} \qquad \frac{118kg}{26.7kN} = \frac{Mass_{LSTME}}{89.6kN/4}$$

$$Mass_{LSTME} = 95kg$$

---

7. THRUST REQUIREMENTS FOR EARTH REENTRY

Variables:

g_o	9.81 m/s^2
ΔV required for deorbit burn	152 m/s

W_o	ignition mass: 29,800 kg
I_{sp}	450 s (specific impulse)
W_f	burnout mass (final mass)
$Mass_{prop}$	propellant mass

Find:
- W_f
- $Mass_{prop}$

Calculations:

$\Delta V = g_o I_{sp} \ln[\frac{W_o}{W_f}]$ $\qquad W_f = 28,791kg$ $\qquad Mass_{prop} = 1009kg$

---

8. WING AERODYNAMIC REQUIREMENTS FOR EARTH LANDING

The final wing calculations using the refined wing area and span are presented below. Calculations performed using the initial wing estimates are not presented below. A discussion of the initial calculations is presented in appendix D.

Variables:

S	wing area (60 m^2)
C_L	coefficient of lift
q	dynamic pressure
V	vehicle velocity (m/s)
W	vehicle mass (20,000 kg)
ρ	density at 4000 ft = 1.15 kg/m^3
A	aspect ratio (2.0)
V_S	stall speed
$C_{l\alpha}$	lift curve slope
L	lift (force)

Find:
- C_L requirement to allow a V_S of less than 180 KTAS at 4,000 MSL
- Determine if wings will be able to generate this coefficient of lift
- Determine estimated $C_{L/\alpha}$ for wings using aspect ratio using flat plate approximation.

Assumptions:

LST mass will be 20,000 kg during re-entry (most propellant expended)

Calculations:

Table E-8: Sample Airfoil Characteristics

Airfoil[1]	Maximum section lift coefficient	Stall angle of attack
NACA 64-006	0.81	10
NACA 65(216)-415	1.4	18
NACA 23015	1.5	16

Note: 1. Source: Aircraft Design: A Conceptual Approach

$$180KTAS = 180KTAS(\frac{6076ft}{nm})(\frac{hr}{3600s})(0.3048\, {}^{m}\!/_{ft}) = 93\frac{m}{s}$$

$$C_L = \frac{L}{qS} = \frac{mg}{\frac{1}{2}\rho V^2 S}$$

$$C_L = \frac{(20,000kg)(9.81\frac{m}{s^2})}{(\frac{1}{2})(1.15kg/m^3)(93m/s)^2(60m^2)} = 0.66$$

$$C_{L_\alpha} = 2\pi(\frac{A}{A+2}) \qquad C_{L_\alpha} = 2\pi(\frac{2.0}{2.0+2}) = 3.14/radian$$

$$(\frac{0.66}{3.14/radian})(\frac{180}{\pi}\frac{\deg ree}{radian}) = 12\deg$$

Approximate angle of attack will be 12 degrees at 180 KTAS. Twelve degrees should be less than the stall angle of attack. Several airfoils were evaluated for maximum lift coefficients and approximate stall angles of attack. Typical airfoil data is presented in table E-8.

Conclusion:
Wings will be able to support a coefficient of lift of 0.66.

9. TAIL AERODYNAMIC REQUIREMENTS FOR EARTH LANDING

Variables:

S_{VT}	Surface area of vertical tail required
c_{VT}	vertical tail volume coefficient (assume 0.07)
b_W	wingspan (11.0 m)
S_W	surface area of wing (60 m^2)
L_{VT}	tail quarter chord to wing quarter chord distance (3.7 m)

Find:

- S_{VT} required

Assumptions:

- C_{VT} is equal to 0.07 for typical jet fighter (Aircraft Design: A Conceptual Approach, p. 125). Estimated c_{VT} for space shuttle is 0.05.

Calculations:

$$S_{VT} = \frac{c_{VT} b_W S_W}{L_{VT}} \qquad S_{VT} = \frac{(0.07)(11.0m)(60m^2)}{3.7m^2} = 12.5m^2$$

---

10. LST MASS ESTIMATES

LST final mass estimates are presented in table E-9. The final estimated mass of the LST was 30,241 kilograms. Masses were estimated using the following methods:

1. Known material masses were applied to designed vehicle values (i.e. Lower surface tile area and tile density).

2. Equations from Aircraft Design: A Conceptual Approach (Raymer, 1999) were applied using either a fighter or transport aircraft as appropriate. Scale factors were applied for composites and advanced future materials.

3. Systems on other spacecraft with known masses were compared to proposed systems on the LST. For example, the US Space Shuttle reaction control system propellant mass is 3,300 kilograms. The LST will be equipped with a very similar system using the same propellant. The LST propellant requirements were estimated to be 1,100 kilograms.

4. Calculations were performed to determine values using basic equations (i.e. propellant masses).

5. Best guess estimates were used when the above methods were not available or appropriate data was not found (i.e. toilet mass estimated to be 20 kilograms).

Table E-9: LST Final Mass Estimate (1 of 3)

Item	Notes and Justification	mass (kg)
Structure		
Fuselage	Raymer table 15-2. 24 kg/m^3--scaled down by factor of 0.85 (advanced materials). Wetted area 57 m^2	1162
Wings	Raymer table 15-2. 44 kg/m^2--scaled down by factor of 0.80 (advanced materials). Exposed area 44 m^2.	1760
Tail	Raymer table 15-2. 44 kg/m^2--scaled down by factor of 0.78 (advanced materials). Exposed area 13 m^2.	446
Tiles (lower surface)	Used heavy tiles (12 lb/ft^3). Lower surface area = 92 m^2, tiles 3 inches thick.	1350
Tiles (upper surface)	Used lighter tiles (9 lb/ft^3). Upper surface about the same as lower=92 m^2,tiles 3 in.	1010
Tiles (tail)	Used heavy tiles (12 lb/ft^3). Tail area = 26 m^2, tiles 3 inches thick.	380
Windows	Estimated	100
Firewall	Raymer equation 15.8. Area of firewall approximately 28 m^2	14
Landing skids	Raymer Table 15-2. Assumed 0.033 for fighter aircraft, 20,000 kg landing mass. Used 40% of value since skids were being used instead of landing gear.	264
Parachute system	Estimated using PRS K-500 parachute system as a reference	612
Clamps and miscellaneous structure	1.1 factor on structure (wings, fuselage, tail)	337
Airlock	Shuttle airlock is 272 kg, scale down size for LST	146
Engines Systems and Propellant		
Main engines	See appendix E for calculations, thrust requirement for lunar landing.	380
Engine systems	Estimated	50
Main engine propellant (Lunar Mission)	See appendix E for calculations	13940
Main engine propellant (ISS Mission)	Volume required for ISS payload. O_2 and H_2 tanks removed to allow for payload.	11500
Attitude maintenance system (AMS)	Estimated AMS system mass to be approximately 15 percent of propellant mass.	165
AMS propellant	US Space Shuttle carries 3,300 kg. LST is approximately 1/3 the size of shuttle.	1100
Cooling/insulation components for propellant	Liquid O_2 and H_2 tanks require insulation/cooling components. Estimated.	500

Table E-9: LST Final Mass Estimate (2 of 3)

Item	Notes and Justification	Mass (kg)
Aircrew and Life Support Systems		
Aircrew	8 aircrew at 80 kg each	640
Aircrew flight gear	8 aircrew at 10 kg each (estimate 10 kg for boots, helmets, flight suits, etc)	80
Aircrew seats	Typical passenger seat weighs 15 kg, assume seat is more robust but uses lighter materials (15 kg/seat)	120
Crew life support system	CO_2 scrubber, life support systems, water generation, cabin air. US Space Shuttle systems are approximately 2,800 kg. Assume improvements in technology, shorter mission for LST-11 days vice 16 for space shuttle.	1500
Toilet	Estimated	20
Food	Assumed 5 lbs/day per crew for a 11 day mission (food is dehydrated)	200
Air volume	Volume is approximately = 45 m^3 (1.20 kg/m^3 density for 70 deg C)	54
Lockers for flight crew gear/food	US Space Shuttle locker weight for food lockers is 30 lbs, use same value for LST	14
Vehicle Systems		
Electrical	Raymer equation 15.20. Rating 160, 4 generators, 25 ft for length, 8 crew, Kmc=1.0	326
Flight control system	Raymer equation 15.17. Mach 1.0 at first application of controls, surface area 10.5 m^2, 2 systems, 2 crew	254
Electric or hydraulic actuators for flight controls	Equation 15.38. Number of functions = 7, fuselage length = 16.6, wing span 13.3 m.	137
Avionics	Equation from p 476, typical uninstalled weight is 360 to 640 kg	490
Electronic systems (sensors, computers, etc)	Estimated mass	200
Payloads		
Payload for lunar mission	Requirement for LST	2500
Payload for ISS mission	Requires removal of propellant tank (less approximately 2,500 kg of propellant)	4500
Payload for lunar return mission	None. Payload is removed at lunar base for return trip.	0

Table E-9: LST Final Mass Estimate (3 of 3)

Item	Notes and Justification	Mass (kg)
Lunar Mission Total Mass		**30241**
Space Station Mission Total Mass	Propellant and systems removed. Additional internal payload added.	**29801**
Lunar Return Trip Mission Total Mass	Internal payload removed.	**27741**

11. SPECIFIC MASS ESTIMATE CACULATIONS

PARCHUTE MASS ESTIMATE:

The LST parachute mass was estimated by making a comparison with the PRS K-500, a parachute system designed for light civil aircraft. The PRS K-500 is designed to provide a rate of descent of less than 6 m/s for an aircraft of 500 kg.[12] The PRS R-500 mass is 18 kg.

Variables:

$Mass_{PRS}$	mass of PRS (18 kg)
$PRS_{VehicleMass}$	design mass for PRS system (500 kg)
$Mass_{LST}$	LST mass (17,000 kg)
$Mass_{LSTparachute}$	LST parachute mass

Find:

- Estimate for LST parachute system total mass

Assumptions:

- Parachute masses are proportional for LST and the PRS system

Calculations:

$$\frac{Mass_{PRS}}{PRS_{VehicleMass}} = \frac{18kg}{500kg} = \frac{Mass_{LSTparachute}}{Mass_{LST}} = \frac{Mass_{LSTparachute}}{17,000kg}$$

$$Mass_{LSTparachute} = 612kg$$

12. LST INTERNAL PROPELLANT VOLUME REQUIREMENTS

Variables:[13]

Space Shuttle External Tank H_2 Volume	53,518 ft^3
Space Shuttle External Tank H_2 Weight	227,641 lbs
Space Shuttle External Tank O_2 Volume	19,563 ft^3
Space Shuttle External Tank O_2 Weight	1,361,936 lbs

Find:

- Propellant volume requirement for lunar profile
- Propellant volume requirement for space station profile

[12] Source for PRS K-500 parachute system: http://mven.netfirms.com/inst.htm.
[13] Source for US Space Shuttle external tank parameters: Shuttle Crew Operations Manual (1993).

Assumptions:

- LST main engines will use same propellant as space shuttle and mixture ratio and pressures will be the same
- LST propellant density if the same as Space Shuttle propellant density
- 6 to 1 oxidizer to propellant ratio[14]

Calculations:

$$\frac{H_2 mass + O_2 mass}{H_2 vol + O_2 vol} = \frac{(227{,}641 + 1{,}361{,}936) lbs}{(53{,}518 + 19{,}563)\, ft^3} = 21.75\, {}^{lb}\!/_{ft^3} = 350\, {}^{kg}\!/_{m^3}$$

$$Lunarpropvolume = \frac{13{,}940kg}{350\, {}^{kg}\!/_{m^3}} = 40m^3$$

$$Stationpropvolume = \frac{11{,}500kg}{350\, {}^{kg}\!/_{m^3}} = 33m^3$$

13. AMS SYSTEM PROPELLANT VOLUME REQUIREMENTS

Variables:

V_{NTO}	NTO volume (oxidizer)
V_{MMH}	MMH volume (propellant)
ρ_{NTO}[15]	density of NTO (1.43 g/cm^3)
ρ_{MMH}	density of MMH (0.86 g/cm^3)
Total NTO/MMH mass	1,100 kg

Find:

- AMS Propellant Volume Requirements (NTO and MMH volumes) given total mass of propellant/oxidizer is 1,100 kilograms.

Assumptions:

- 1.6 to 1 oxidizer to propellant ratio[16]

Calculations:

$$mass_{NTO} = 1.6 mass_{MMH} = \rho_{NTO} V_{NTO} = 1.6 \rho_{MMH} V_{MMH} \qquad V_{NTO} = 0.96 V_{MMH}$$

[14] Same oxidizer to propellant ratio as US Space Shuttle, Shuttle Crew Operations Manual (p. 2.16-8).
[15] Source for NTO and MMH average bulk densities: Space Mission Analysis and Design (p. 644).
[16] Same oxidizer to propellant ratio as US Space Shuttle, Shuttle Crew Operations Manual (p. 2.20-3).

$$TotalMass = \rho_{NTO} V_{NTO} + \rho_{MMH} V_{MMH}$$

$$1{,}100kg = (1{,}430\frac{kg}{m^3})(0.96V_{MMH}) + (860\frac{kg}{m^3})V_{MMH}$$

$V_{MMH} = 0.49m^3$ $V_{NTO} = 0.47m^3$

$mass_{MMH} = 423kg$ $mass_{NTO} = 677kg$

APPENDIX F: REFERENCE DATA

Table F-1: Lunar Orbital Parameters

Parameter	Value
semi-major axis (a) – mean	384,400 km
Perigee	363,300 km
Apogee	405,500 km
eccentricity (e)	0.0549
Inclination (I) – mean	5°8’ (4°59’ to 5°18’)

Note: Source: Fundamentals of Astrodynamics,

Table F-2: ΔV Requirements for Lunar Missions

To: / From:	Low earth Orbit	Lunar Transfer Orbit	Low Lunar Orbit	Lunar Descent Orbit	Lunar Landing
Low Earth Orbit		3.107 km/s			
Lunar Transfer Orbit	3.107 km/s		0.837 km/s		3.140 km/s
Low Lunar Orbit		0.837 km/s		0.022 km/s	
Lunar Descent Orbit			0.022 km/s		2.684 km/s
Lunar Landing		2.890 km/s		2.312 km/s	

Note: Source: http://spacecraft.ssl.umd.edu

Table F-3: International Space Station Approximate Orbital Parameters

Parameter	Value
semi-major axis (a)	6750 km
eccentricity (e)	0.002
Inclination (I)	51.8

Note: 1. Source: http://spacelink.nasa.gov

Table F-4: Propellant Densities

Propellant	Isp	Avg Bulk Density (g/cm^3)
O_2 and H_2	450	1.14 and 0.07
O_2 and RP-1	350	1.14 and 0.80
N_2O_4 and MMH	300-340	1.43 and 0.086

Note: Source: Space Mission Analysis and Design.

Table F-5: Earth and Lunar Parameters

Parameter	Moon	Earth
Mass	0.07349×10^{24} kg	5.9736×10^{24} kg
Surface gravity	1.62 m/s^2	9.81 m/s^2
μ (gravitational parameter)	0.049×10^5 km^3/s^2	3.986012×10^5 km^3/s^2
Mean Equatorial Radius	1738 km	6378.145 km
Escape Velocity	2.38 km/s	11.2 km/s

Note: Source: http://www.lunarrepublic.com/

Table F-6: Apollo Lunar Module Specifications

Module	Mass (kg)	Propellant Mass (kg)	Δ V (m/s)	Propellant	Isp (s)
Descent Module	10,149	8,165	2,470	NTO/Aerozine-50	311
Ascent Module	4,547	2,358	290	NTO/Aerozine-50	311

Notes: 1. Source: http://users.commkey.net/Braeunig/space/specs.htm

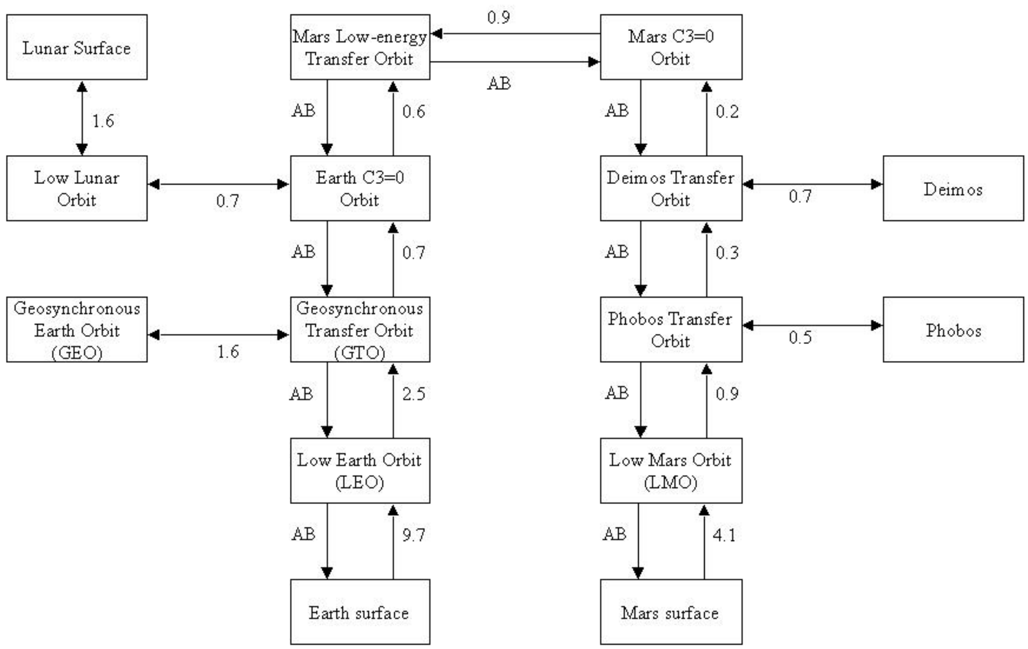

Figure F-1: Delta V Requirements for Lunar and Mars Missions
(Source: http://www.pma.caltech.edu/~chirata/deltav.html)

OTHER SPACE CRAFT ILLUSTRATIONS

Figure F-2: HL-20 PLS
(Source: http://www.friends-partners.ru/partners/mwade/graphics/h/hl20hl42.jpg)

Figure F-3: Orbital Space Plane Proposal
(Source: http://www.orbital.com/LaunchVehicle/SpacePlane)

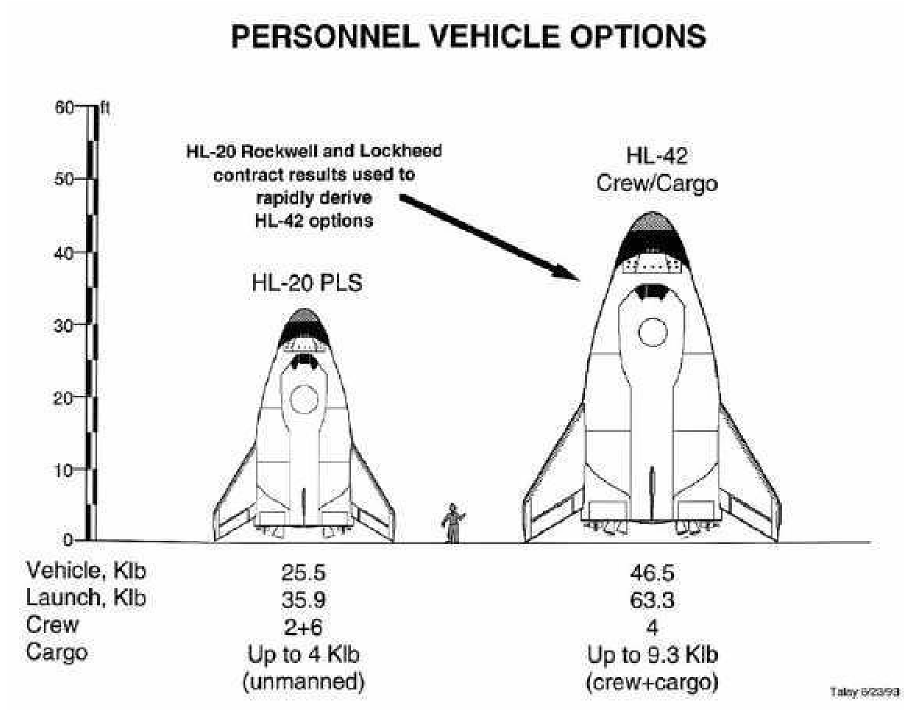

Figure F-4: HL-20 and HL-42
(Source: http://www.friends-partners.ru/partners/mwade/graphics/h/hl20hl42.jpg)

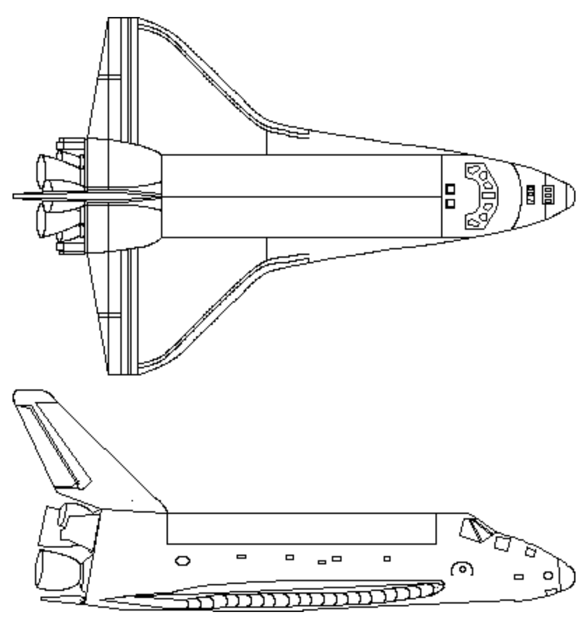

Figure F-5: US Space Shuttle Two View Illustration
(Source: http://www.friends-partners.ru/partners/mwade/graphics/h/hl20hl42.jpg)

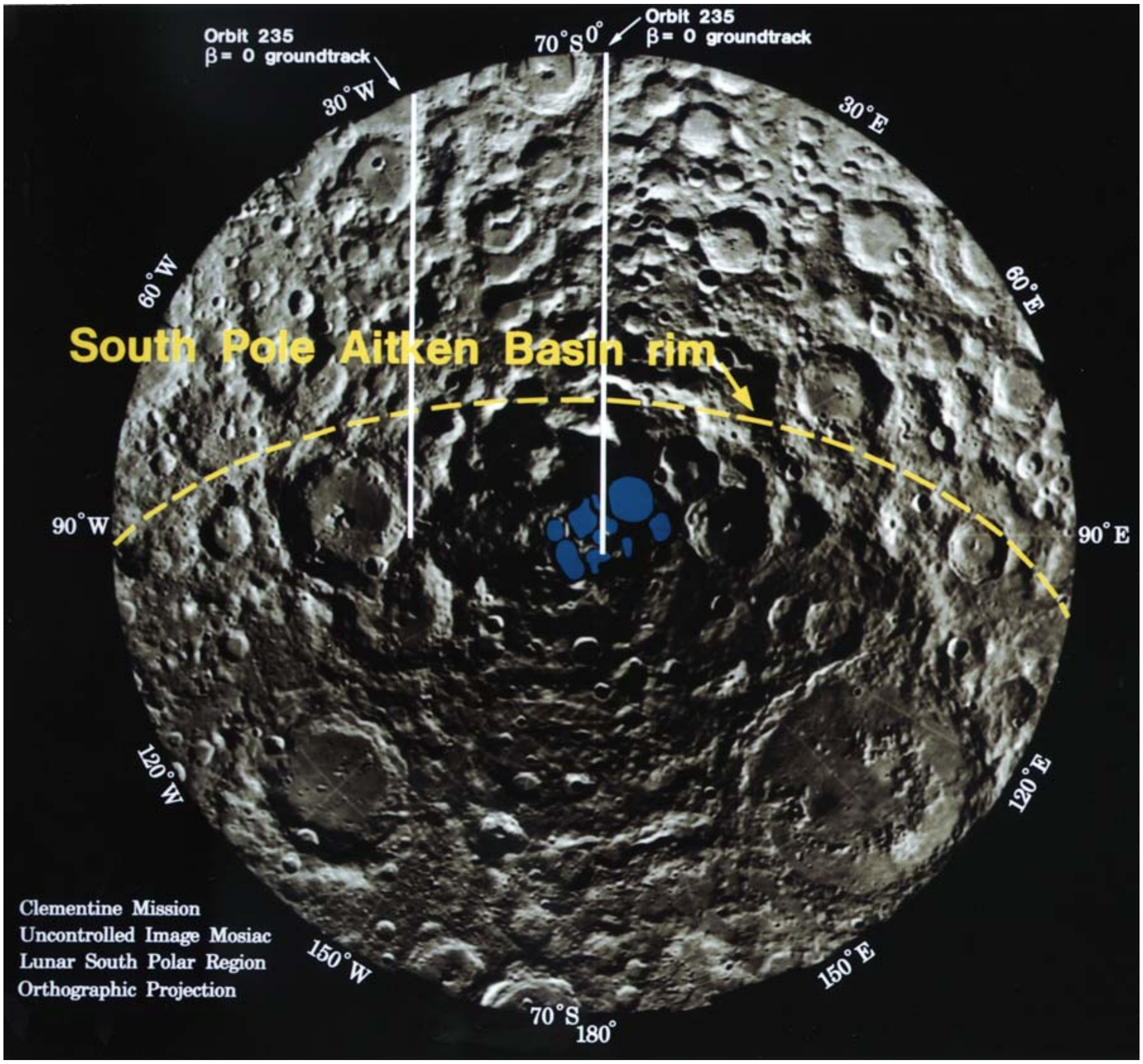

Figure F-6: Clementine Lunar Ice Discovery Illustration
(Source: http://www.cmf.nrl.navy.mil/clementine/clementine.html)

APPENDIX G: DEFINITIONS AND VEHICLE DESCRIPTIONS

DEFINITIONS AND VEHICLE DESCRIPTIONS

Assured Crew Return Vehicle (ACRV) – The ACRV was a “lifeboat” designed to be used by the space station crew to abandon the shuttle in the event of a station or space shuttle emergency. The project was cancelled due to cost. The crew on the space station depend on the Soyuz capsule as an escape vehicle. The ACRV design was based on a lifting body shape. The design parameters were: Crew Size: 10. Design Life: 3 days. Orbital Storage: 3.00 days. Total Length: 8.9 m. Maximum Diameter: 7.2 m. Total Habitable Volume: 16.40 m3. Total Mass: 10,884 kg. Total Payload: 545 kg. Electrical System: Batteries.[17]

C3=0 Orbit –A parabolic orbit where the spacecraft has just enough energy to escape earth's gravity. It is used to send spacecraft (e.g. the Lunar Prospector) to the moon. Once the spacecraft reaches the moon, it can enter lunar orbit or land on the lunar surface.[18]

Hohmann Transfer orbit – The lowest energy orbit requiring the least speed change (ΔV) between two coplanar circular orbits.[19]

RP-1- The petroleum used as rocket fuel is kerosene, or a type of highly refined kerosene called RP-1 (refined petroleum). It is used in combination with liquid oxygen as the oxidizer.[20]

HL-20 PLS- HL-20 PLS was the designation of NASA’s ACRV (Assured Crew Return Vehicle) and PLS (Personnel Launch System).[21]

HL-42 – The HL-42 was a scaled up version of the HL-20 PLS. The HL-42 was scaled up by a factor of 42 percent. The HL-42 reference vehicle was a reusable, lifting body spacecraft designed to be placed into low-Earth orbit by an expendable booster. The design parameters were Crew Size: 4. Total Length: 12.8 m. Maximum Diameter: 5.5 m. Total Mass: 21,093 kg. Total Payload: 4,300 kg. Total Propellants: 2,000 kg. Total RCS Impulse: 5,800,000.00 kgf-sec. Main Engine Propellants: Lox/Methane. Main Engine Isp: 300 sec. Total spacecraft delta v: 290 m/s. Electrical System: Fuel cells.[22]

Orbital Space Plane – The orbital space plane is an ongoing design being developed as a replacement to the space shuttle. Ideas being considered are a lifting body shape as well as a capsule similar in shape to the Soyuz and Apollo capsules. Initially, the design was

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[17] Source: http://www.friends-partners.ru/partners/mwade/graphics/h/hl20hl42.jpg
[18] Source: http://www.pma.caltech.edu/~chirata/deltav.html
[19] Source: Fundamentals of Astrodynamics.
[20] Source: Space Mission Analysis and Design.
[21] Source: http://www.friends-partners.ru/partners/mwade/graphics/h/hl20hl42.jpg
[22] Source: http://www.friends-partners.ru/partners/mwade/graphics/h/hl20hl42.jpg

leaning towards a vehicle with wings that would fly back to earth. Recently, NASA has been leaning towards a capsule capable of carrying four crew.

VITA

Matthew D. Finney was born in Monterey, CA on February 5, 1971. In 1987 his family settled in Friendswood, Texas. He graduated from Friendswood High School in June of 1989. From July 1989 to May 1993, he attended the United States Naval Academy in Annapolis, MD graduating with a Bachelor of Science in Aerospace Engineering. He was commissioned an Ensign in the United States Navy and briefly worked at NASA Johnson Space Center from July to October 1993 before reporting to Pensacola, Florida for flight training as a Student Naval Aviator. After receiving his Naval Aviator wings in December 1994, he was selected to be a pilot in the F/A-18 Hornet. After completing initial F/A-18 training at Marine Corps Air Station El Toro, California, he served with the VFA-113 "Stingers" in Lemoore, California. During this tour, his squadron was based aboard the USS Abraham Lincoln aircraft carrier. In July of 2000, he reported to the U.S. Naval Test Pilot School at Patuxent River Naval Air Station, Maryland, and graduated with class 119 in June 2001. Upon graduation, he reported to Naval Weapons Test Squadron in China Lake, California. He served as the lead test pilot for the Advanced Mission and Computer and Display integration into the F/A-18EF Super Hornet. He will be joining the VFA-14 "Top Hatters" based in Lemoore, California for his second operational tour in December 2003.