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FINAL DESIGN REPORT OF A PERSONNEL LAUNCH SYSTEM AND A FAMILY OF HEAVY LIFT LAUNCH VEHICLES

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Executive Summary

During the course of the year 1990, numerous questions were raised regarding the ability of the current Shuttle Orbiter to provide reliable, on-demand support of the planned space station. Besides being plagued by reliability problems, the Shuttle lacks the ability to launch some of the heavy payloads required for future space exploration, and is too expensive to operate as a mere passenger ferry to orbit. Therefore, additional launch systems are required to compliment the Shuttle in a more robust and capable Space Transportation System.

In addition to this, the December 1990 Report of the Advisory Committee on the Future of the U.S. Space Program, headed by Norm Augustine, advised NASA of the risks of becoming too dependant on the Space Shuttle as an all-purpose vehicle. Furthermore, the committee felt that reducing the number of Shuttle missions would prolong the life of the existing fleet. In their suggestions, the board members strongly advocated the establishment of a fleet of unmanned, heavy lift launch vehicles to support the space station and other payload intensive enterprises.

Another committee recommendation was that a space station crew rotation/rescue vehicle be developed as an alternative to the Shuttle, or as a contingency if the Shuttle is not available. The committee emphasized that this vehicle be designed for use as a personnel carrier, not a cargo carrier. This recommendation was made to avoid building another version of the existing Shuttle, which is not ideally suited as a passenger vehicle only.

The objective of this project was to design both a Personnel Launch System (PLS) and a family of Heavy Lift Launch Vehicles (FHLLVs) that provide low-cost and efficient operation in missions not suited for the Shuttle.

The PLS vehicle is designed primarily for space station crew rotation and emergency crew return. Therefore, a nominal compliment of eight passengers is provided for. Studies have indicated that a small, reusable, lifting-body spacecraft can operate at greater cost effectiveness, reliability and safety than the Shuttle. The personnel vehicle is carried into low Earth orbit by a partially reusable, man-rated version of the heavy lift vehicles co-designed in this project.

The final design of the PLS vehicle is depicted in Figure 1. It has an overall length of 36 ft and an overall width of 27 ft. The weight of this vehicle is 30,000 lbs. The vehicle has provisions for eight passengers and a flight crew two for a maximum mission duration of three days.

The interior of the craft is shown in Figure 2. Although it is meant to be a payload intensive vehicle, the PLS is designed to carry a minimum of space station resupply with specific cargo area designed into the craft. More cargo area can be gained by removing the passenger seats when the PLS vehicle does not have a full crew compliment.

The PLS vehicle is designed to be boosted into orbit by launching it serially from a man-rated rocket. To ensure crew safety during ascent, the final design provides for an on-pad abort, as well as an abort during ascent if an emergency situation arises.

The mission of the Family of Heavy Lift Launch Vehicles (FHLLVs) is to place large, massive payloads into Earth orbit with payload flexibility being considered foremost in the design. Because of this concern, the final design of three launch vehicles was found to yield a payload capacity range from 20 mt to 200 mt. These designs include the use of multi-staged, high-thrust liquid engines mounted on the core stages of the rocket. Payload flexibility is provided by the use of multiple strap-on solid rocket boosters. The final design of the FHLLV project consists of three basic configurations: the SR-1, the SR-2 and the SR-3. These vehicles are shown in comparison in Figure 3.

The SR-1 is the smallest vehicle in the launch vehicle family. It has a payload capacity of

20 mt to 95 mt depending on the number of SRB's used, and whether or not a second stage is employed. Figure 4 illustrates the basic dimensions of the SR-1 in the 72 mt configuration. This configuration employs 2 SRB's and the second stage. The SR-1 can mount two or four SRB's as required to increase the payload capacity.

The first stage of the all liquid-propelled core utilizes three SSME-35's for propulsion, and is a cylindrical structure that houses the oxidizer and fuel for the first stage in separate tanks. The first stage is 31 ft in diameter and 149 ft tall. The second stage of the SR-1 relies on two, unmodified SSME's for thrust. It has a diameter of 24 ft, and a length of 82 ft without the payload shroud.

Overall, the SR-1 stands 357 ft tall, and has a width of nearly 70 ft. The gross lift-off weight (GLOW) and stage dimensions for the SR-1 are shown in Figure 4.

The SR-2 is the medium capacity vehicle in the launch vehicle family. It has a payload capacity of 40 mt to 150 mt depending on the number of SRB's used, and whether or not the second stage is employed. Figure 5 illustrates the basic dimensions of the SR-2 in the 100 mt configuration. This configuration employs 2 SRB's and the second stage. The SR-2 can employ two, four or six SRB's as required to increase the payload capacity. The first stage of the all liquid-propelled core utilizes five SSME-35's for propulsion. The first stage is 40 ft in diameter and 149 ft tall. The second stage of the SR-2 relies on two or three, unmodified SSME's as needed for thrust. The second stage has a diameter of 31 ft, and a length of 82 ft without the payload shroud.

Overall, the SR-2 stands 384 ft tall, and has a width of nearly 76 ft. The gross lift-off weight (GLOW) and stage weights for the SR-2 are shown in Figure 5.

The SR-3 is the largest vehicle in the launch vehicle family. It has a payload capacity of 140 mt to 200 mt depending on the number of SRB's used. Figure 6 illustrates the basic dimensions of the SR-3 in the 200 mt configuration. This configuration employs six SRB's. The SR-3 can mount two, four, six or eight SRB's as required to increase the payload capacity.

The first stage of the all liquid-propelled core utilizes eight SSME-35's for propulsion. The first stage is 50 ft in diameter and 149 ft tall. The second stage relies on two or three, unmodified SSME's as needed for thrust. The second stage has a diameter of 40 ft, and a length of 82 ft without the payload shroud.

Overall, the SR-3 stands 440 ft tall, and has a width of nearly 86 ft. The gross lift-off weight (GLOW) and stage weights for the SR-3 are shown in Figure 6.

Both the PLS and FHLLV systems designed by Spacely's Rockets fit neatly into the planned evolution of the U.S. space program. The PLS, if actually constructed, would provide more efficient manned access to space on a routine schedule of flights. This in turn, alleviates fears that the Space Station Freedom will be built without a guarantied crew return vehicle.

The construction of the Family of Heavy Lift Launch Vehicles would give the U.S. unprecedented launch capacity for any program being pursued, and potentially provide the inexpensive commercial access to space. Thus, the hopes of the Space Exploration Initiative and other projects can be realized by finally having a heavy lift system available.

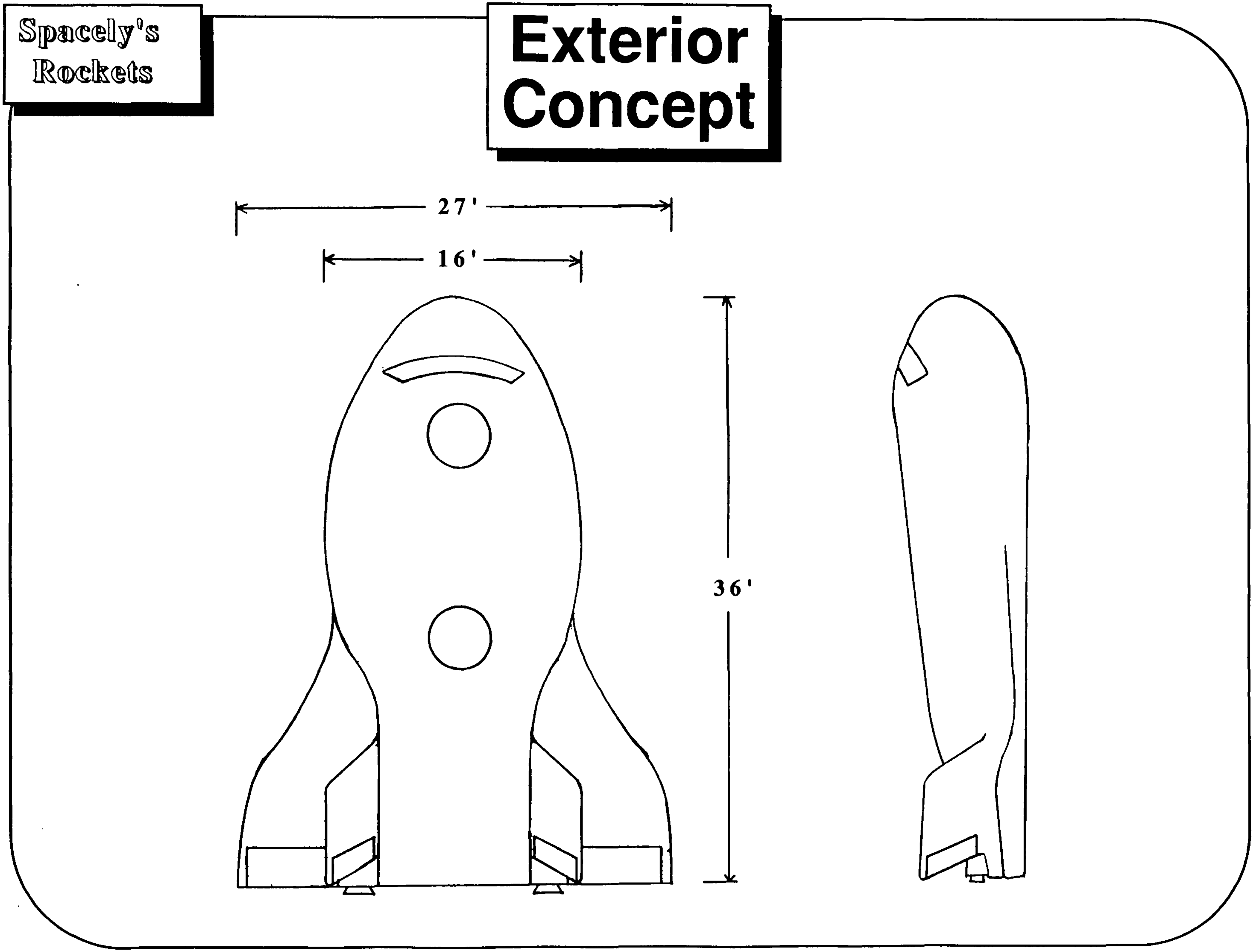

Figure 1. PLS Exterior Concept

Spacely's Rockets

Interior Diagram

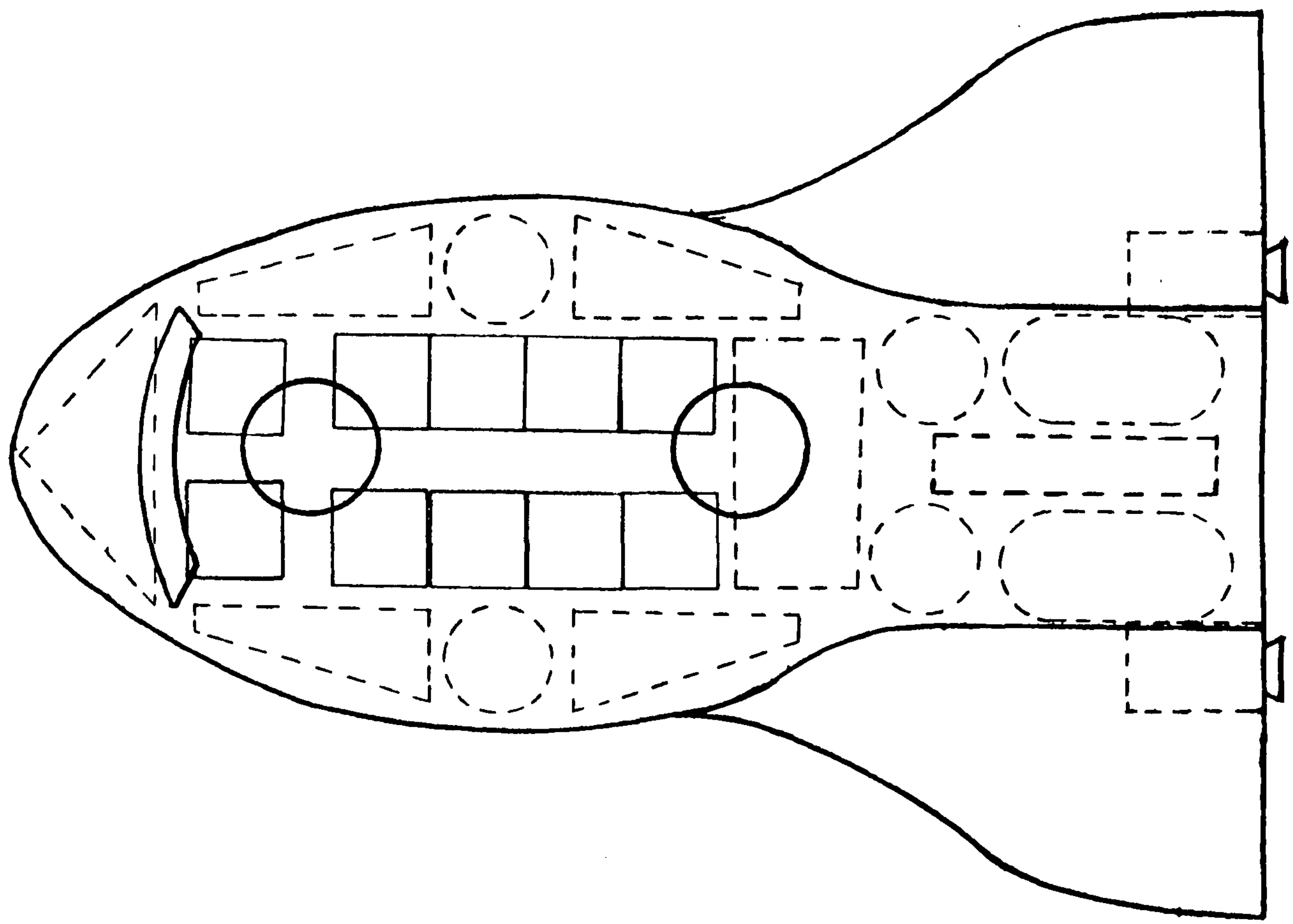

Figure 2. PLS Interior Concept

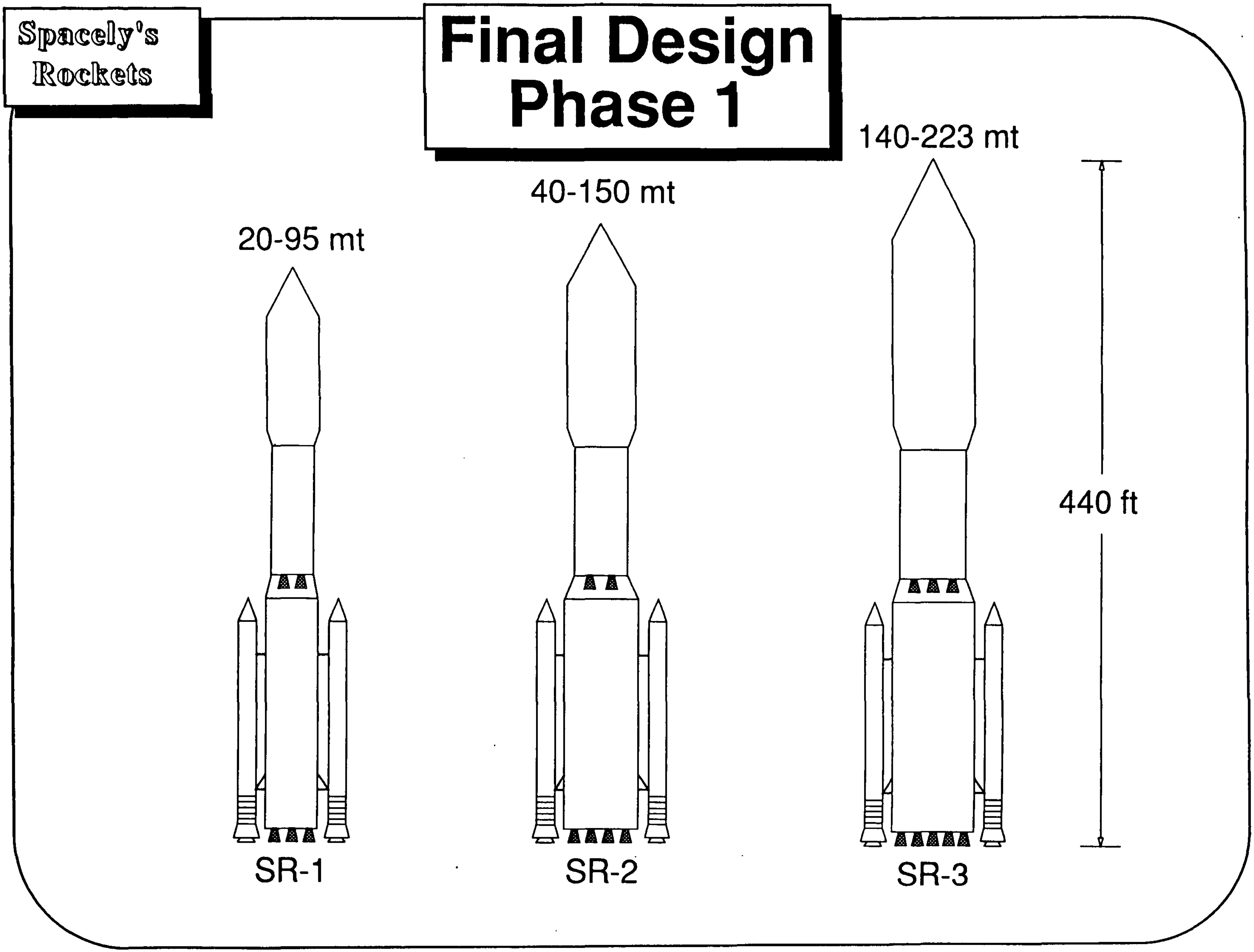

Figure 3. Launch Vehicle Family

Spacely's Rockets

72 MT Capacity

GLOW	**5,328,660 lb**
Payload Shroud	**3307**
<u>2nd Stage</u>	
Structural Mass	**47,058**
Propellant Mass	**306,004**
2 SSME's	
<u>1st Stage</u>	
Structural Mass	**122,355**
Propellant Mass	**1,178,083**
3 SSME-35's	
2 SRB's	

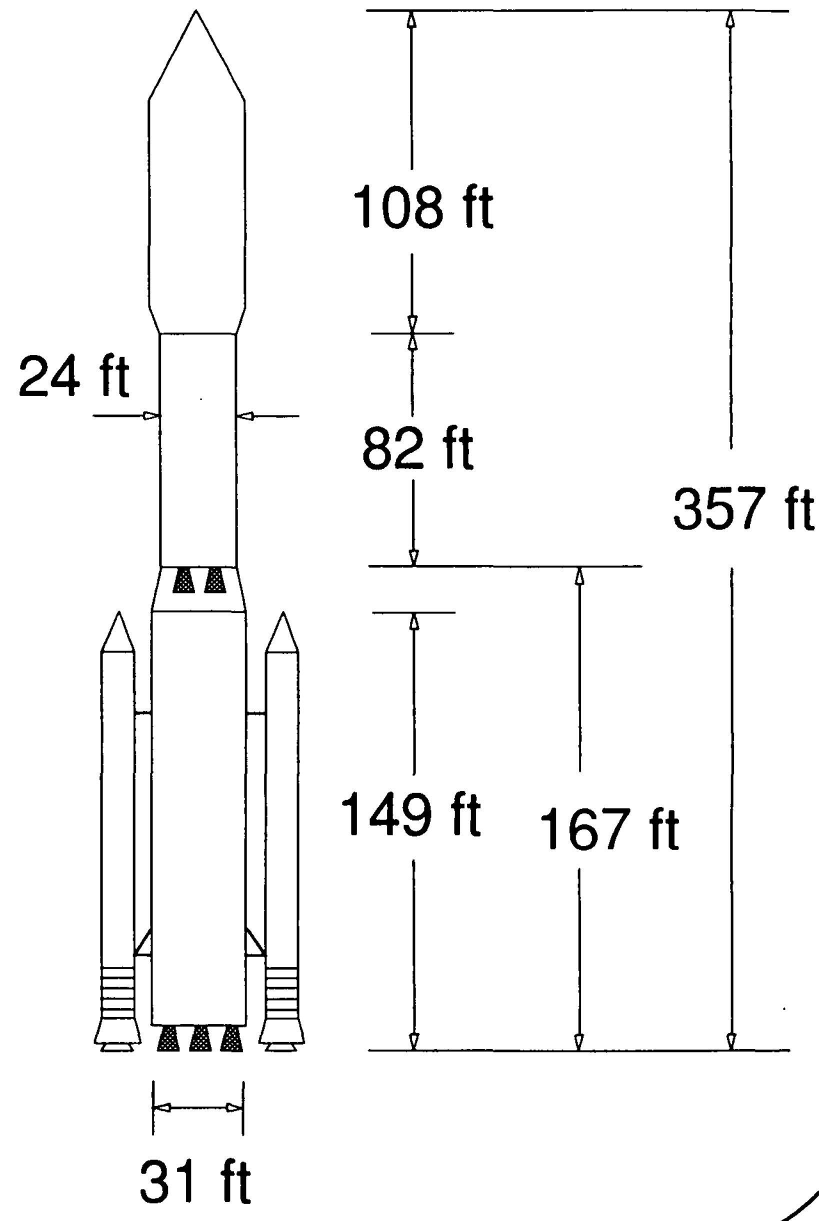

Figure 4. SR-1 Launch Vehicle

Spacely's Rockets

100 MT Capacity

GLOW	**5,496,064 lb**
Payload Shroud	**6,614**
<u>2nd Stage</u>	
Structural Mass	**78,446**
Propellant Mass	**509,915**
2 SSME's	
<u>1st Stage</u>	
Structural Mass	**203,925**
Propellant Mass	**1,963,472**
5 SSME-35's	
2 SRB's	

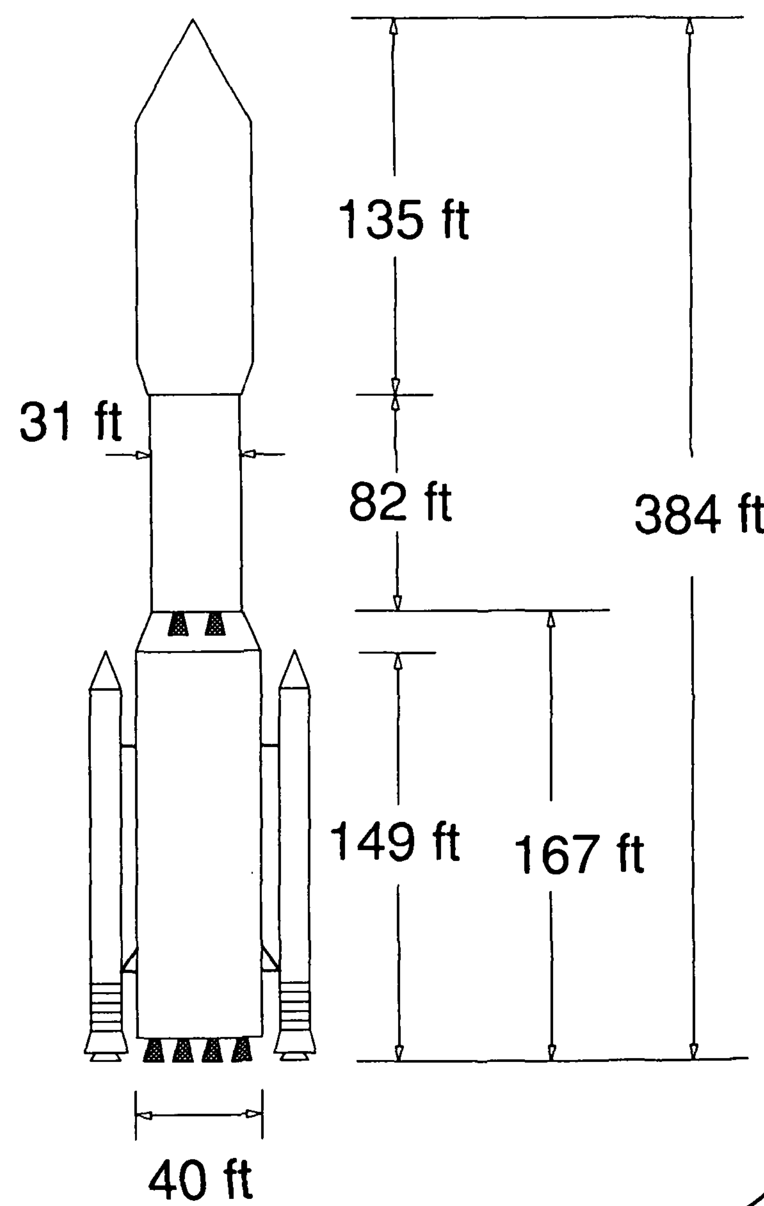

Figure 5. SR-2 Launch Vehicle

Spacely's Rockets

200 MT Capacity

GLOW	12,399,460 lb
Payload Shroud	11,023
2nd Stage	
Structural Mass	125,662
Propellant Mass	817,702
3 SSME's	
1st Stage	
Structural Mass	326,281
Propellant Mass	3,141,555
8 SSME-35's	
6 SRB's	

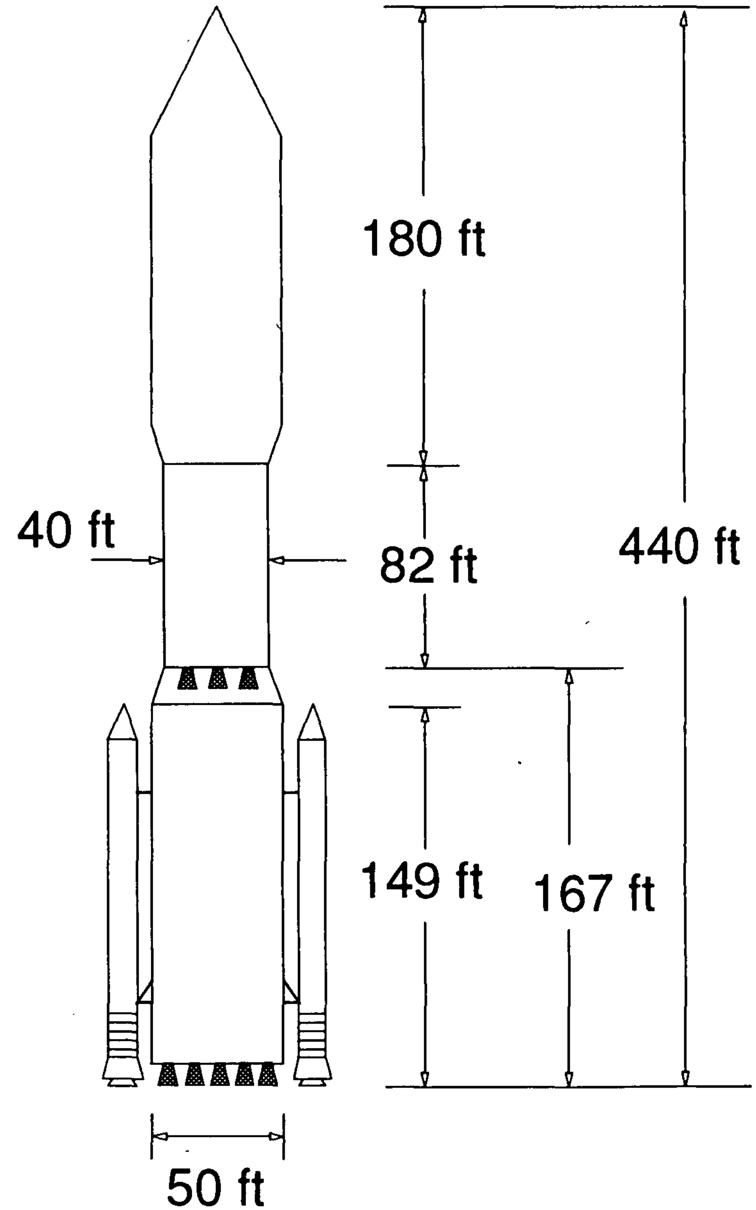

Figure 6. SR-3 Launch Vehicle

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List of Acronyms

ALS	Advanced Launch System
APU	Auxiliary Power Unit
ASRM	Advanced Solid Rocket Motor
BIA	Bus Interface Adapter
CRT	Cathode Ray Tube
ECLSS	Environmental Control and Life Support Systems
EDO	Extended Duration Orbiter
ELV	Expendable Launch Vehicle
EVA	Extravehicular Activity
FHLLV	Family of Heavy Lift Launch Vehicles
GNC	Guidance Navigation and Control
GPC	General Purpose Computer
GPS	Global Positioning System
IMS	Inertial Measurement System
INS	Inertial Navigation System
LEO	Low Earth Orbit
LES	Launch Escape System
LiOH	Lithium Hydroxide
LRB	Liquid Rocket Booster
LRU	Line Replacement
MDM	Multiplexer/Demultiplexer
NIU	Network Interface Unit
OMS	Orbital Maneuvering System
PDR	Preliminary Design Review
PLS	Personnel Launch
PRSD	Power Reactant Storage and Distribution System
RCRS	Regenerative CO2 Removal System
RCS	Reaction Control System
SAATCOM/LEASAT	Military UHF Satellite Relay Systems

SDOF	Single Degree of Freedom
SDP	Standard Data Processor
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine
TDRSS	Tracking and Data Relay Satellite System
TVC	Thrust Vector Control
UHF	Ultra High Frequency
WCS	Waste Collection Subsystem
WEARM	Wet Engine and Avionics Recovery Module

1 General Summary

Currently the United States Space Transportation System consists of the Space Shuttle and several low to medium payload launch vehicles, such as the Delta and Titan. However, to help build and maintain the proposed space station Freedom or a future lunar base, a Personnel Launch System (PLS) and Family of Heavy Lift Launch Vehicles (FHLLV) will be needed. The following summary will outline why a PLS and FHLLV are needed, and how Spacely's Rockets will accomplish the design requirements. In addition, the following document includes a review of the technical design completed to date, a cost analysis of the cost of the project, and a management review explaining the administration of the project.

1.1 Assumptions

Spacely's Rockets made the following assumptions in order to design the PLS and FHLLV systems:

1. Space Station Altitude: 220-250 nmi
2. Space Station Inclination: 28.5 deg
3. Typical Low Earth Orbit: 160 nmi.
4. Current launch facilities adequate for PLS and FHLLV missions.

1.2 PLS Project Background

Currently the only personnel launch vehicle available to the U.S. is the Space Shuttle. However, since the Space Shuttle is designed to carry crew as well as large payloads it is not cost effective for the shuttling of crew only. In addition, the Space Shuttle's complex design and 1970's technology diminish its ability to perform missions due to maintenance, reliability, and long launch buildup times. Therefore, Spacely's Rockets is designing a PLS that can shuttle personnel with incidental cargo to and from space frequently and efficiently. The PLS will have a simpler design than the Shuttle because of its more limited mission. The simpler design will allow higher reliability, less maintenance, lower manufacturing costs, and higher launch frequency than the Shuttle.

1.3 FHLLV Project Background

Currently the U.S. does not have a heavy lift launch vehicle, and the Space Shuttle is not very flexible since the same structure and launch system is used no matter how heavy the payload. Furthermore, additional expense is incurred with the Shuttle since it must be man-rated for its crew. Therefore, Spacely's Rockets has designed a FHLLV that can launch various payloads (up to 200 metric tons) into various orbits frequently and efficiently. By varying the launch configuration, the FHLLV is able to tailor itself to various missions, thus reducing the waste of unneeded structure. The FHLLV has a simple design to help facilitate high launch frequencies, low maintenance, simple manufacture, and cost effectiveness.

2 PLS Project Final Design Review

The PLS Preliminary Design Review details the current level of PLS design. The Preliminary Design Review contains a brief descriptive overview of the division of tasks within the PLS Division of Spacely's Rockets, a listing of the mission scenarios that the PLS will likely fulfill, a description of basic system requirements, and a general description of the major technical areas, such as body configuration and subsystems, that have been researched.

2.1 Overview

The PLS group of Spacely's Rockets has the task of defining the PLS as well as designing it. The two working groups within the PLS division reflect this need to be both general and specific. The two groups are the Body group and the Subsystems group. The Body group is concerned with body configuration, sizing, interior configuration, thermal protection, docking mechanisms and crew escape. The Subsystems group is responsible for power, GNC & communications, ECLSS & medical, avionics, and vehicle propulsion.

2.2 Mission Scenario

The exact mission scenario for PLS has been left up to the PLS division to define. The major school of thought about the PLS is that it should provide an economical and simple way for man to access space. It should, in conjunction with the FHLLV, enhance and eventually move beyond the Space Shuttle. This school of thought also dictates that a man rated system should be used for accessing man to space, while an unmanned system, such as the FHLLV, be used for the majority of space-bound cargo. Therefore, the PLS will be mainly a people carrier with limited resupply capability. The PLS division has determined primary mission scenarios and possible secondary mission scenarios to shoot for in the design.

2.2.1 Primary Missions

The primary mission visualized for the PLS will be to service the Space Station. That includes mainly crew rotation since it will be assumed that major resupply will be accomplished by unmanned cargo vehicles. The PLS should also be able to rescue Space Station and Space Shuttle crews in case of emergency. This dictates that the PLS have the proper reentry environment for injured personnel. To make the PLS more efficient, it should have the capability to carry small amounts of cargo when not carrying a full load of passengers (via removable seats.) The number of vehicles needed and the expected frequency of launch will be determined in the design process.

2.2.2 Secondary Missions

Some alternate secondary missions will be considered for the PLS, although they will not be a driving factor in the design. The secondary missions now being considered are lunar and planetary crew rotation (via rendezvous with orbital transfer vehicles), minor LEO platform repair, and minimal Space Station resupply. Another task that may be considered for the PLS is to leave one at the Space Station at all times to act as a lifeboat.

2.3 System Requirements

The following is a list of basic criteria that the PLS division feels that the PLS must meet:

1. The PLS must have the capacity to carry 2 crew members and 8 passengers.
2. The PLS must be able to attain Space Station altitude at a minimum (approx. 220 nautical miles).
3. The PLS must be designed for a mission duration of no less than and probably no longer than 3 days.
4. The PLS must have a simple, cost effective design.

5. The PLS must be able to sustain a high launch frequency.

6. The PLS must incorporate emergency escape systems.

7. The PLS must be able to dock universally with all manned space systems (such as the Space Station and Space Shuttle).

8. The PLS must have minimal environmental impact.

While some of these criteria may be altered during the design process, every effort will be made to meet them.

2.4 Technical Research Areas

The PLS division focused the majority of its design efforts in the technical areas of research described below.

2.4.1 Body Configuration

Spacely's Rockets considered three different types of body configurations: a winged body, a fuselage with attached wings much like an airplane; a lifting body, only a fuselage tapered to provide lift; and a capsule, like those used in early spacecraft missions.

2.4.1.1 Primary Design

After consideration of the three basic body types, a hybrid combining two of the types was chosen - a winged lifting body. A winged lifting body incorporates the good points of both the winged and the lifting bodies. Figure 2.1 is a conceptual sketch of the chosen design.

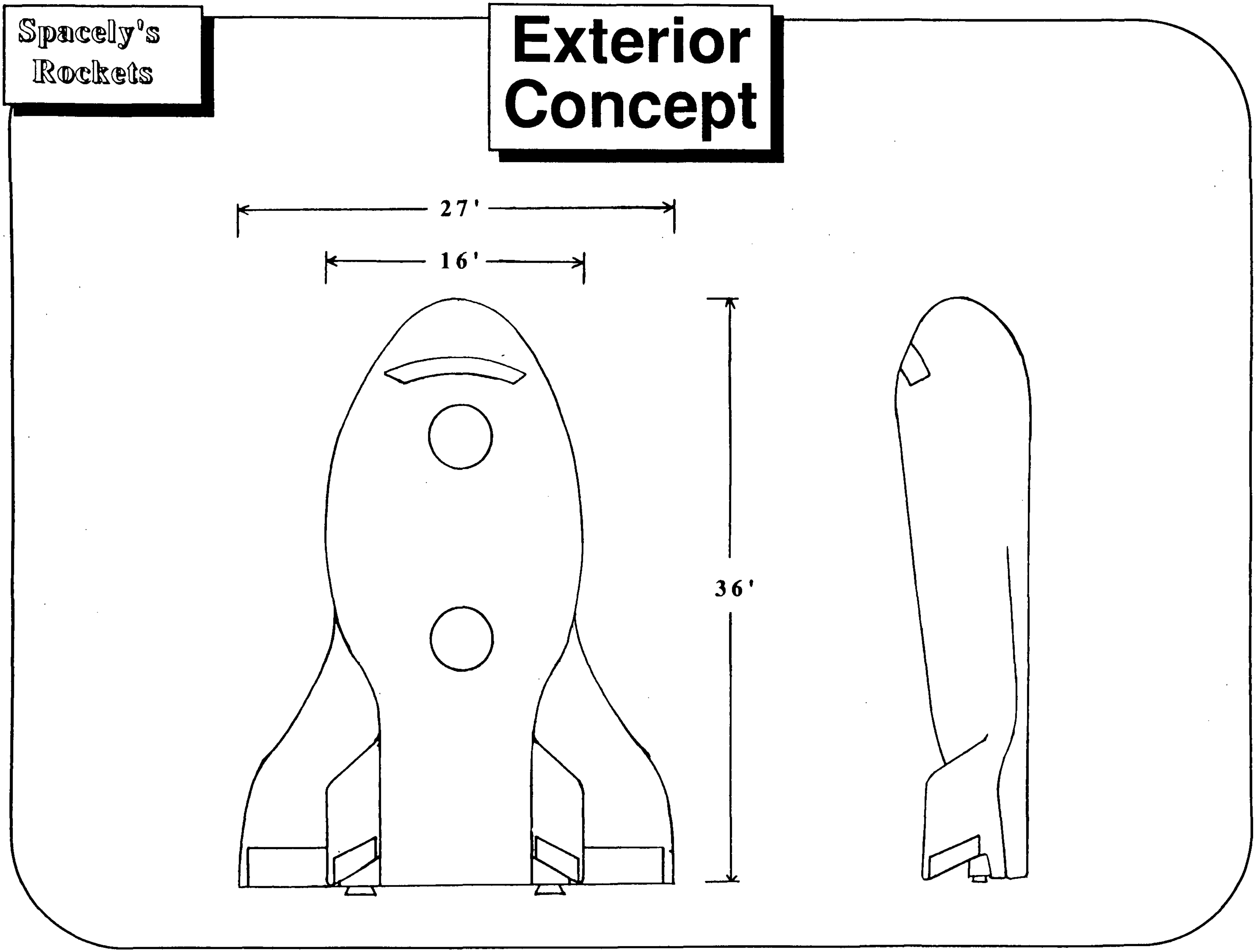


Figure 2.1 Exterior Concept

The fuselage provides lift and tapers into small wings. This configuration provides for horizontal landing so that landing at or near the launch site is possible, thus shortening the launch buildup time. With a capsule, a water landing is almost the only option because of attenuation problems. After a splashdown, the craft must be retrieved before reuse is possible. The winged lifting configuration enables a pilot to control the craft during landing and gives the craft cross range capability in case of an emergency. Because the wings provide lift with only a slight drag penalty, the lift to drag ratio is better than that of a pure lifting body. The tapered design creates less aerodynamic heating than a pure winged body, which tends to have sharp edges where the wing and the fuselage meet, causing high pressure gradients. Structure is used more efficiently with a winged lifting body than with a pure winged body. The fuselage of the latter creates a lot of drag and very little, if any, lift, where the fuselage of the winged lifting body provides lift, therefore, the structure is not wasted.

Though a capsule is simple and inexpensive to design, the advantages of the chosen design are of greater importance, and in the long term will provide a cheaper, more reliable means of meeting the mission objectives.

2.4.1.2 Aerodynamic Performance

Determination of the aerodynamic characteristics of the craft was difficult without the empirical data that wind tunnel tests can provide. Therefore, these characteristics were approximated analytically. Only subsonic characteristics were calculated since landing was the primary flight regime where performance was crucial.

Since the entire fuselage provides lift, the craft was considered a wing without a fuselage. This assumption seems a bit optimistic, but other approximations were made very conservatively. This simplified the performance calculations significantly, and without empirical data, the performance values will lack accuracy.

With a length of 36 ft and a span of 27 ft in a delta configuration, the maximum lift over drag ratio was determined to be about 8.1. This is much higher than that of the shuttle - about 4.8 - so there remains plenty of room for error. The planform area was calculated to be 472 sq ft and the maximum coefficient of lift was conservatively approximated to be 1. For a landing weight of 25,000 lb, the landing velocity was calculated to be 210 ft/sec, low compared to the shuttle's landing velocity of 290 ft/sec.

Because the performance values determined were much better than required for a smooth, safe landing, the fact that the fuselage was considered a part of the wing should not affect the final performance too adversely. The high performance values obtained will easily allow the craft to be landed at the launch site.

2.4.1.3 Interior

Figure 2.2 is a sketch of the projected interior design of the PLS. The crew of two is to be seated side by side and the eight passenger seats will be arranged with four rows of two adjacent seats each down the center of the craft. On either side of the passenger seats, space is provided for any needed subsystem components. Aft of the passenger seats is a cargo area (4 x 7 x 5 ft) just large enough for critical supplies.

Spacely's Rockets

Interior Diagram

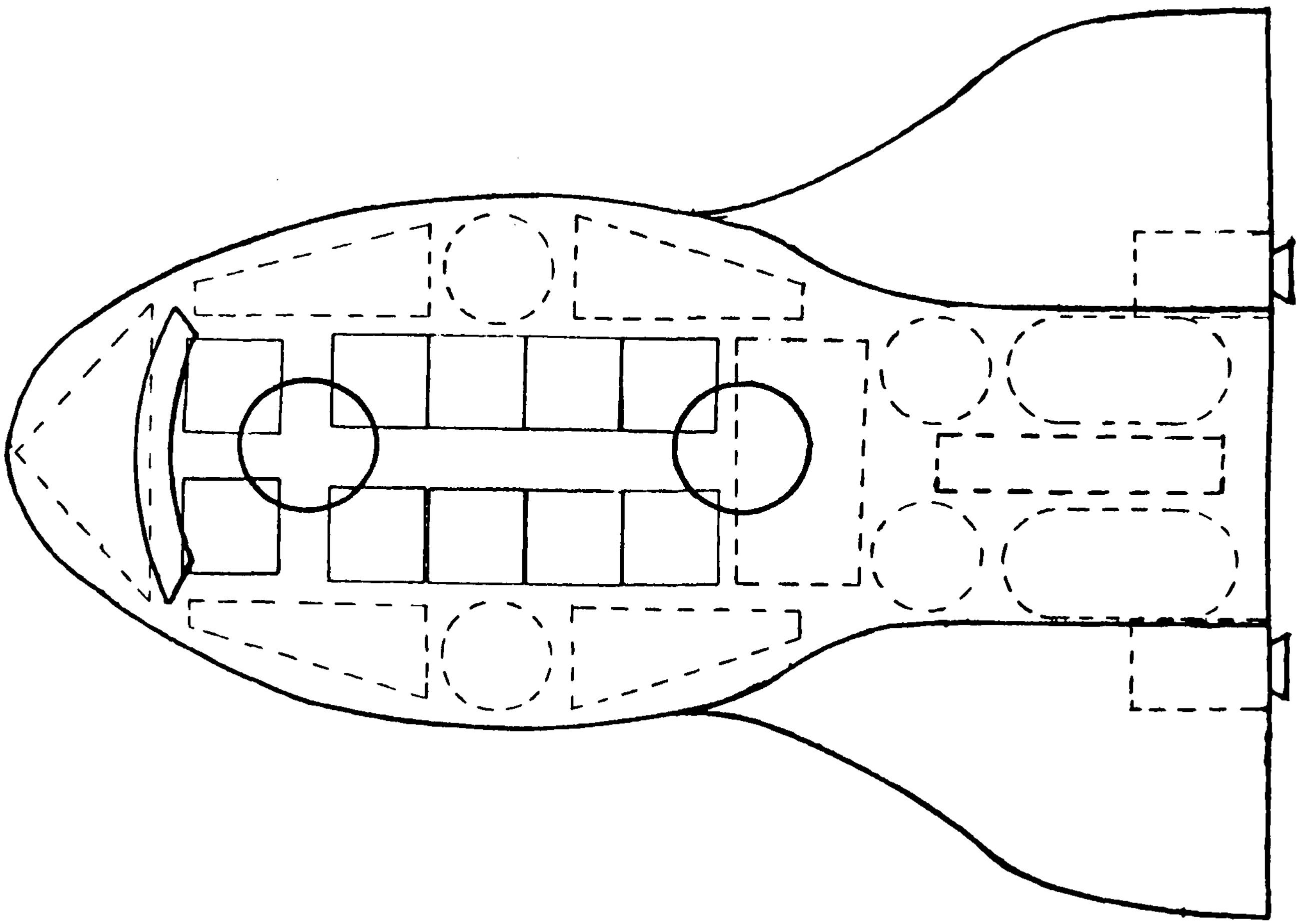


Figure 2.2 Interior Concept

Since the purpose of this craft is primarily for personnel, a large cargo bay is not necessary and would be a waste of structure if not used. If, however, there are less than eight passengers and more supplies are needed, seven of the seats will be designed so that they can be individually removed to provide necessary space. The dimensions of any cargo are limited to the size of the two hatches which are located on the top of the craft just aft of the crew seats and the passenger seats, respectively, and have a diameter of 4 ft.

2.4.2 Launch and Landing Scenarios

In order to insert the PLS into orbit, it is necessary to have a launch vehicle for the PLS, since it will not have on-board main engines like the Space Shuttle. Figure 2.3 depicts the chosen launch configuration as well as the landing configuration. The primary choice for the PLS ascent and orbit insertion is a serial Expendable Launch Vehicle (ELV). The launch vehicle will insert the PLS into a 180 nmi orbit, where it will perform an OMS burn to attain Space Station orbit. A lift-off weight breakdown is presented in Table 2.1. Because of its simplicity and current availability, the serial ELV launch configuration was chosen over parallel ELV launch, piggyback atop an aircraft or an air-breathing platform, and an air-breathing first stage. Another consideration in the launch vehicle choice is the availability of a manned rated version of a FHLLV. However, due to its simple design, the PLS can be easily modified to be launched from other launch vehicles as they are developed.

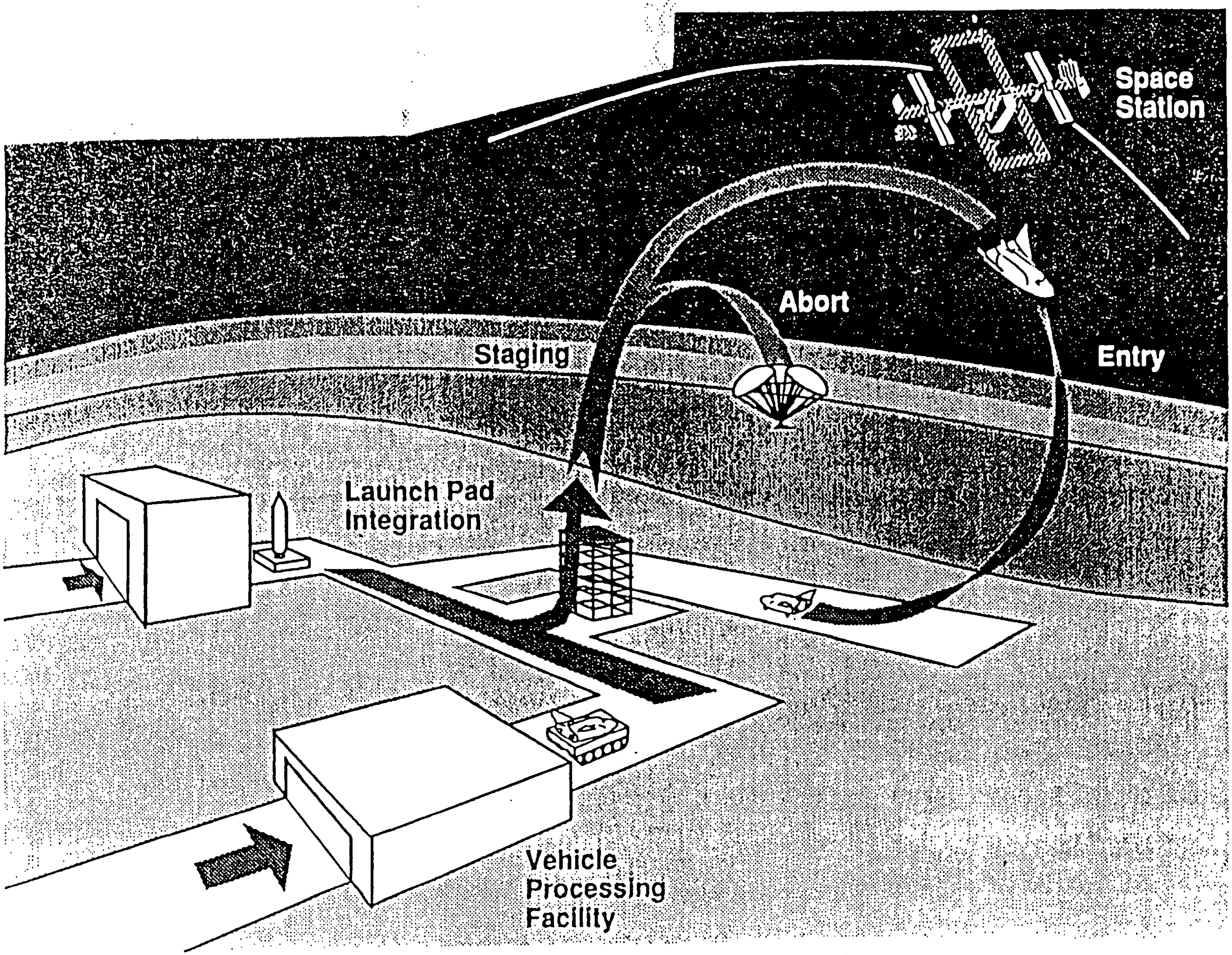


Figure 2.3 PLS Mission Scenario

Table 2.1 Breakdown of PLS Launch Weights

Component	Weight (lbs)
PLS	30,000
LES	10,500
Launch Vehicle Adaptor	7,500
Liquid Core	1,313,705
SRB (2)	2,512,772
Total Lift-off Weight	3,874,477

Since the body design chosen for the PLS is a winged lifting body, the primary landing scenario chosen is a horizontal land landing at the launch site. This scenario was chosen over vertical (parachute) land or water landing because the horizontal landing facilitates the reusability of the system, since the problems of corrosion due to salt water and shock attenuation are not encountered. In addition, having the PLS land primarily at the launch site reduces turnaround time and transportation costs.

2.4.3 Abort Scenarios

For the safety of the crew while on-pad, a launch escape system (LES) has been integrated into the PLS. The LES consists of an 8-g pitch motor and PLS adaptor. The motor will fire to separate the PLS from the launch vehicle and propel the PLS to 4000 ft. Once the altitude has been attained, parachutes will be deployed allowing the PLS to land at 30 ft/s. The LES scenario is depicted in Figure 2.4. In addition, the same parachute system will be used for emergency vertical landing during ascent or after reentry.

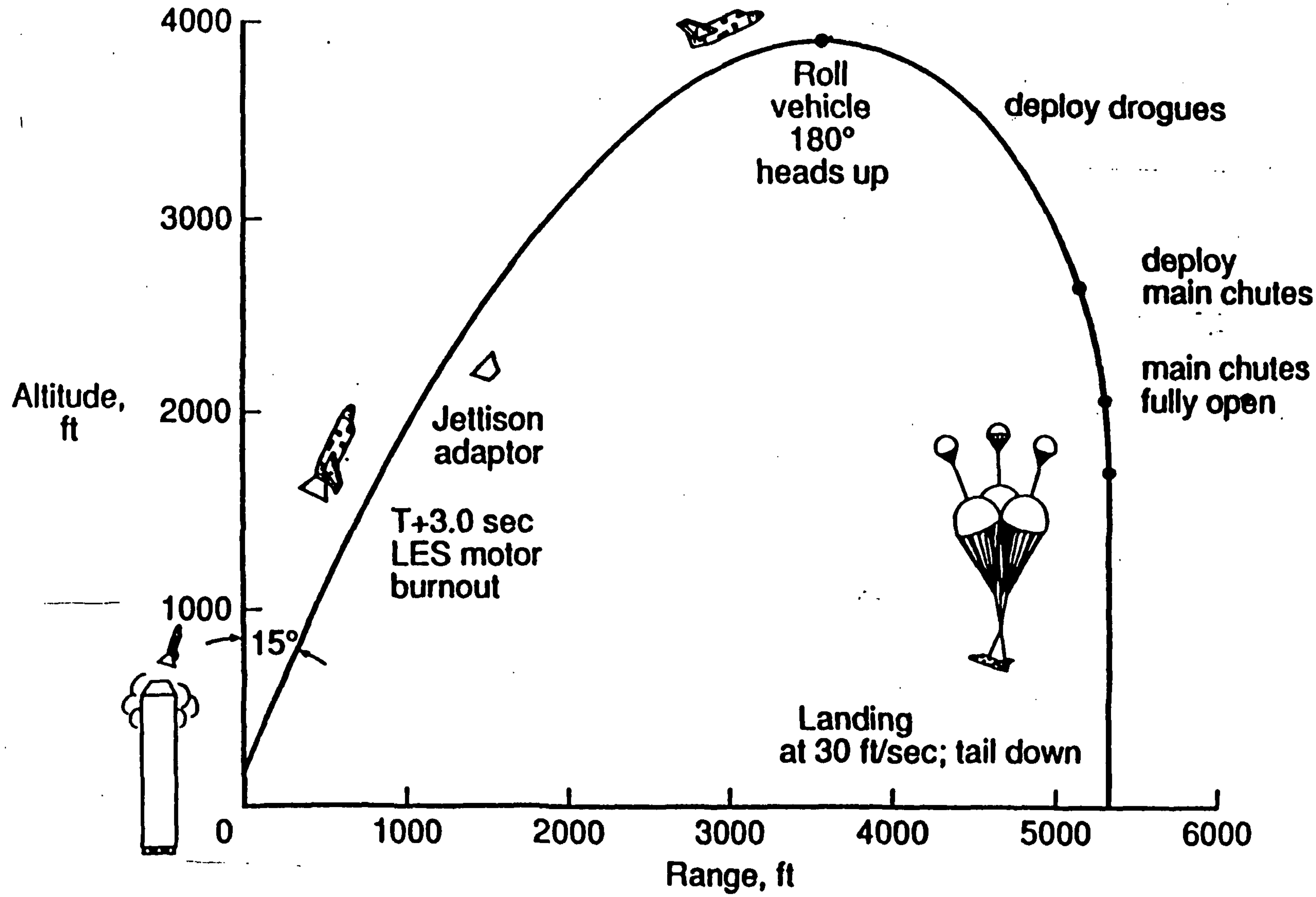


Figure 2.4 PLS On-Pad Abort

One landing abort scenario considered was the use of OMS or an on-board solid motor to enable the PLS to retry a landing. However, OMS engines are not effective at low altitudes due to back pressure and a solid motor would be too large and heavy to fit within the PLS configuration. Therefore, the parachute system was chosen for landing abort above 4000 ft.

2.4.4 Materials

The materials researched include those considered for use on the hull and for thermal protection during reentry.

2.4.4.1 Hull

The primary structural component is the hull. The material making up the hull must be light, have a high modulus of elasticity, low thermal conductivity, and a low coefficient of thermal expansion.

Both aluminum and titanium alloys were considered for the primary material of the hull. Aluminum is ductile, inexpensive, and light weight - density of 0.101 lb/in^3. However aluminum has a low yield strength - 76 x 10^3 psi at room temperature - and high thermal conductivity, causing heat removal problems after landing. Though somewhat more difficult to machine and slightly heavier - density of 0.164 lb/in^3, titanium has a higher yield strength - 170 x 10^3 psi at room temperature, and has much lower thermal conductivity than aluminum. The linear coefficient of thermal expansion is also much less for titanium than for aluminum.

The advantages of titanium overshadowed those of aluminum. The low thermal conductivity of titanium would significantly reduce the need for heat removal equipment used after landing, and increase landing flexibility by allowing for more landing site possibilities. The linear coefficient of thermal expansion of titanium would relieve much of the shear stresses created by the interface between the heat shielding and the hull. The heaviness of titanium would not be a problem since its high strength would allow for less material to be used. Over time, these advantages would more than make up for the added expense of having to machine titanium.

2.4.4.2 Thermal Protection

Three thermal protection methods were considered for use on the PLS: a one piece heat shield, a few large panels, and a shuttle type tile system. The panel system was chosen as the primary design. This method is much simpler than the tile system, and consists of only a few large, molded sheets as opposed to more smaller panels or tiles which are prone to fall off, requiring more maintenance. The panel system was chosen over the one piece heat shield method because of the shear stress created between the hull and the shield itself. The use of titanium helps alleviate this stress, but not to the extent of allowing the use of the one piece heat shield.

The panels will consist of a low density silica ceramic material. A nylon felt material will be used as an interface between the panels and the hull to reduce the shear stresses created by the difference in thermal expansion. The nose and wing leading edges will be coated with reinforced carbon-carbon. These areas become extremely hot and will require more protection than will be provided by the panel material.

2.4.5 Propulsion

The propulsion subsystem will consist of the Orbital Maneuvering System (OMS). PLS will not require main engines because it is a small, lightweight vehicle and the launch vehicle can provide the needed thrust. Without main engines, the PLS is a simple, low cost vehicle that has high reliability and maintainability. The turn around and launch buildup time will be reduced without main engines which allows for the high launch frequency that is desired.

The OMS must provide thrust to transfer the PLS from one orbit to another, rendezvous with the Space Station and Shuttle, and to deorbit. The system will consist of 2 OMS pods with each containing an engine, the propellants, and the propellant tanks. The propellants consist of monomthylhydrazine as the fuel and nitrogen tetroxide as the oxidizer. They are hypergolic, which means they ignite when they are brought together. Hypergolic fuels can be dangerous, but they are easily handled and do not require the cooling that cryogenic fuels do. Therefore they do not pose greater problems than any other propellants would. They produce 313 seconds of specific impulse and about 6,000 pounds of thrust in vacuum. These numbers coincide with the specifications of the Space Shuttle OMS which uses the same propellants.

The OMS will provide 1,500 feet per second change in velocity for each mission. From simple analysis, this is enough velocity change to ascend from LEO to Space Station altitude, perform about 2 degrees of plane change, and deorbit with 300 feet per second to spare. The launch vehicle will place the PLS within 2 degrees of the desired orbit on all missions. The propellant weight to provide 1,500 feet per second is approximately 5,000 pounds. The propellant weight is divided into 1,890 pounds of fuel and 3,120 pounds of oxidizer.

Powered landing scenarios were considered. The options were atmospheric engines, OMS with atmospheric capability, and a solid motor. Atmospheric Engines are unneeded since they would be dead weight during ascent, orbit, and descent. The OMS engines were looked at to deliver enough thrust for some thrust for a powered landing, but the OMS engines are ineffective below 70,000 feet altitude due to back pressure. The lift to drag ratio is high enough so that the vehicle could provide adequate cross range capabilities; therefore, a powered landing scenario was not chosen.

2.4.6 Power

The main requirements of the power subsystem is to provide the needed power to all the subsystems and to fit within the vehicle constraints. The choice of power subsystem is dependent upon the amount of power required, mission duration, size constraints, and system's impact on the PLS. Extra consideration to the impact must be taken since the vehicle will be man-rated. The four types of power generation that were examined for the PLS are fuel cells, batteries, solar power, and nuclear power systems.

The primary power system choice is a non-regenerative liquid hydrogen and liquid oxygen fuel cell system. They have been used on manned missions from the Gemini program to the Space Shuttle. The combination of hydrogen and oxygen releases energy which is converted to electrical power. This choice provides many positive aspects. The combination of the two creates water as a by-product which can by used by the Environmental Control and Life Support System. It can be used as a coolant for thermal management also.

The fuel cell system has many advantages over the other types of power generation. Batteries are heavy and provide low levels of power. Solar Arrays require large collection areas and would not be easily integrated into the vehicle design. Nuclear power systems would require extra shielding for the personnel and are not allowed in LEO. The extra shielding would unnecessarily increase the weight of the vehicle. Therefore fuel cells are the most logical choice for the power subsystem.

The chosen power subsystem is similar to the one used by the Space Shuttle. It consists of three main components: power reactant storage and distribution, fuel cell power plants, and electrical power distribution. The power reactant storage and distribution will hold the liquid hydrogen and oxygen and supply them to the fuel cell power plants. There will be two power plants with each producing about 7 KW. This power output should be ample for the needs of the PLS. The electrical power distribution will supply the power from the power plants to each of the subsystems.

2.4.7 Environmental Control and Life Support Systems

The driving impetus behind the design of the Environmental Control and Life Support Subsystem (ECLSS) is simplicity. The ECLSS must be as simple as possible in order to attain the PLS requirements of a short launch buildup time, a high frequency launch rate, and low cost. Because the PLS is being designed for a maximum contingency flight duration of 3 days, a mostly open-loop ECLSS can be utilized. For example, a closed-loop ECLSS would call for the recycling of urine into useable water. This type of system would be inappropriate for a mission duration as short as 3 days. A comparison was made between the current Space Shuttle ECLSS and that of the Extended Duration Orbiter (EDO) in order to choose the proper ECLSS for PLS. The extended duration orbiter is expected to go on line in 1992 (Columbia is being refitted) and will have the ability to achieve a 28 day mission. While this is longer than the mission duration being designed into PLS, it is expected that some of the technology being developed for EDO will be better suited for PLS than that of the current Space Shuttle. The ECLSS subsystems that have been considered are the CO_2 removal subsystem, the N_2 supply system, the O_2 supply system, the waste collection subsystem (WCS), and the cabin thermal control system.

2.4.7.1 CO_2 Removal Subsystem

Two CO_2 removal subsystems were considered for PLS. They are the existing Space Shuttle Lithium Hydroxide system (LiOH) and the EDO Regenerative CO_2 Removal System (RCRS).

The Space Shuttle LiOH system removes CO_2 from the cabin air supply via a chemical reaction with cartridges of LiOH. Two LiOH canisters are placed in the cabin air loop with approximately 112 pounds of air forced through each canister. Each canister weighs 6.2 pounds, occupies 0.24 cubic feet and will last a minimum of 48 man-hours varying with the number of crew, length of the crew's day cycle, and finding the most convenient time for the crew to change the canisters. The support structure for the LiOH system occupies approximately 0.25 cubic feet. For a 3 day mission with 10 personnel on board the total LiOH system would need approximately 4.6 cubic feet of volume and would weigh approximately 200 lbs. The reaction causes excess heat and water to be fed into the cabin which must later be removed by the cabin humidity separator and heat exchanger. The maintenance required for the LiOH system is the changing of the canisters. Power requirements for the LiOH system are unavailable at this time, but it might be assumed that they are negligible since the canisters are placed in the cabin air loop.

The EDO RCRS is a self-contained system that removes CO_2 from the cabin air supply via adsorption into a solid amine media. The RCRS adsorbs CO_2 from cabin air at atmospheric pressure and when exposed to space vacuum desorbs the CO_2 thus regenerating itself for future adsorption. Two beds of solid amine are utilized (one adsorbing while the other is desorbing) and operate on approximately 30 minute cycles. The RCRS unit weighs approximately 328 pounds, requires 340 Watts of power at peak with an average of 200 Watts, and has a useful life of about 5 years. Backup LiOH canisters are employed in case of total RCRS failure. No reactants are fed into the cabin air supply and no consumables are required.

After comparison, the LiOH system was chosen for use in PLS. For a 3 day mission duration the LiOH system is more volume and weight efficient (at least an 8 day mission would be required for the RCRS to be more weight efficient). While no on-flight attention is required for the RCRS, it is conceivable, because of its complexity, that pre-flight checkouts would be more time-consuming than for the LiOH system. While no power requirements are available for the LiOH system at this time, it is assumed that they would be less than or equal to the power requirements for the RCRS. With the exception of the reaction products being fed back into the cabin atmosphere, the LiOH system appears to be the superior system for the PLS mission. It is concluded that the LiOH system is superior for a 3 day mission because it weighs less, occupies less volume, uses equal or less power, and is much more simple and therefore easy to maintain and refurbish during the life of PLS. The only reason for PLS to consider using the RCRS would be if longer duration missions were to be considered.

2.4.7.2 N_2 Supply System

The Space Shuttle cabin continuously bleeds air overboard at a rate of 6 lbs. per day of air due to seepage and intentional outgassing. It is assumed that the PLS will bleed air at a lower rate because only docking with pressurized entities is expected to occur with only emergency EVA's. Oxygen will be resupplied via the Power Reactant Storage and Distribution (PRSD) system but gaseous Nitrogen must be brought from Earth. The EDO will carry a total a six 3300 psia tanks of gaseous nitrogen for a 28 day mission. It can be extrapolated that PLS will need only one tank of gaseous N_2 for its 3 day mission. The gaseous nitrogen would weigh approximately 200 lbs. including tank support structure, tank, and gaseous nitrogen.

2.4.7.3 O_2 Supply System

Oxygen will supplied by the PRSD system. It is estimated that the 3 day contingency mission would require 670 lbs. of gaseous O_2 for the full compliment of 2 crew and 8 passengers.

2.4.7.4 Waste Collection Subsystem

To begin the discussion of the WCS for the PLS it should be noted that only in contingencies would the flight duration be the maximum three days. One should also note that the available volume of the PLS crew cabin is very limited. For the above reasons, the WCS unit will be placed under the front starboard passenger seat. However, only in the event that fecal waste must be excreted would the crew have to exchange places with the passenger of the front starboard seat. Urine tubes connected with the WCS unit will be provided for each passenger seat as well as the crew seats. The WCS unit will incorporate as much of the present day Shuttle WCS design as possible in order to speed its development. However, it will take ingenuity to fit the WCS into the volume provided under the seat and still have the seat meat all comfort and safety criteria.

Airflow is used for urine collection to entrain the fluid and transport it through a prefilter, to remove hair, etc., and then to the phase separator, which removes the urine from the airflow by means of a centrifugal fan. The urine is then pumped to the wastewater collection tank while the air flows through an odor/bacteria filter and then back into the cabin. Fecal collection is performed using the same fan separator for airflow. However, the bolus is transported by the air into a storage container with a fabric lining that acts much as a vacuum cleaner bag. The air flows out of the bag, through an odor/bacteria filter and back into the cabin air stream.

2.4.7.5 Cabin Thermal Control

It is estimated that the cooling requirements for the crew would be 850 BTU/hr for nominal activity. Avionics would require approximately 2.5 kW (24 hr). There are essentially 4 phases that must be considered for cabin thermal control. Those phases are launch, on orbit, reentry, and post landing.

Crew cooling will take place via forced air movement coupled with a water coolant loop. The heat absorbed by the water in the coolant loop is rejected on orbit using a flash evaporator system similar to the Shuttle's. However, since the flash evaporator will not function below 200,000 ft., the coolant loop contains a small cold water reservoir to be used for launch and reentry. The water in the reservoir will be cooled prior to launch by an Earth based system, and prior to reentry by the flash evaporator system. Post landing ventilation will be required. Avionics cooling is accomplished primarily by cold plating interconnected with the coolant loop.

2.4.8 Avionics Subsystem

The avionics subsystem as defined here is the subsystem necessary for interaction between the crew and the PLS vehicle. Guidance, navigation, and control will be considered in Section 2.4.9.

2.4.8.1 Subsystem Requirements

The avionics subsystem, like every system on the PLS, is as simple as possible, while retaining maintainability and reliability. Launch buildup time is a major concern of the PLS, therefore the avionics subsystem incorporates modularity with the maximum number of components consisting of line replaceable units (LRU's). In other words, if a CRT fails the ground crew will not spend time trying to repair it, but will simply replace the entire component. The avionics subsystem is also compatible with the Space Station. The wisdom and necessity of this is obvious. If an LRU fails on orbit it can be easily replaced with a Space Station spare. Avionics bay cooling is discussed in Section 2.4.7.5.

2.4.8.2 Components

Table 2.2 lists the PLS avionics components along with their estimated weight power and volume. Several of the major components are described below.

The data net interface consists of a 250 channel multiplexer/demultiplexer unit (MDM), a network interface unit (NIU), and a bus interface adapter (BIA). The MDM is triply redundant because a multiplexer failure would be a point failure for any of those channels. However, both the NIU and BIA have an interface unit per string and therefore do not require interface redundancy.

The general purpose computer (GPC) will be a scaled down version of the Space Station standard data processor (SDP). The GPC will be about a third the size of the SDP. The present intent is to use a 286/020 16-bit microprocessor manufactured by Electronic Memories Corporation. The GPC will be triply redundant.

The memory storage unit contains backup programs of all the software and is doubly redundant.

Four interactive CRT screens constitute the PLS "all glass cockpit". The CRT's replace many of the displays, controls and switches that would normally be present. There are several reasons for incorporation of the "all glass cockpit" into the PLS. Electronics have been proven to have a longer mean time between failure than mechanical switches throughout the history of the space program. The CRT's would also reduce control panel area and clutter since only pertinent information need be brought on screen at any one time.

Table 2.2. Avionics Requirements for PLS.

Systems	Size, in.^{3}	Wt., lb.	Power, W
Data Net Interface	12096	302	553
GPC	1410	66	300
Memory Store	1140	50	100
Displays	6000	185	568
Controls	1672	92	9
Lighting	1140	44	750
Power Distribution	7344	295	350
Wiring	13000	390	NA
Total (rounded)	43800	1425	2650

2.4.9 GNC and Communications

This section describes the design of the systems chosen for the Guidance, Navigation, and Control (GNC) and communications subsystems. Table 2.3 presents a breakdown of equipment, weight, and power requirements for the GNC and communications systems.

Table 2.3. Breakdown of GNC and Communications Hardware.

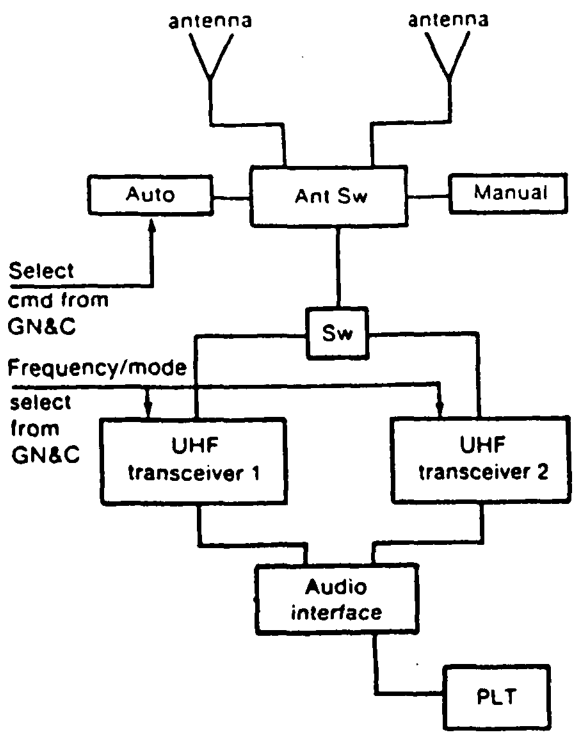
Hardware	Number	Weight (lbs)	Power (watts)
RCS (dry)	3	1,041	57
RCS Propellants	-	950	-
IMS	2	84	200
UHF/GPS	2	334	105
Horizon Sensor	2	20	40
Sun Sensor	2	16	24
Pitot-Static System	2	30	40
Radar Altimeter	2	20	26

2.4.9.1 Communications and Tracking

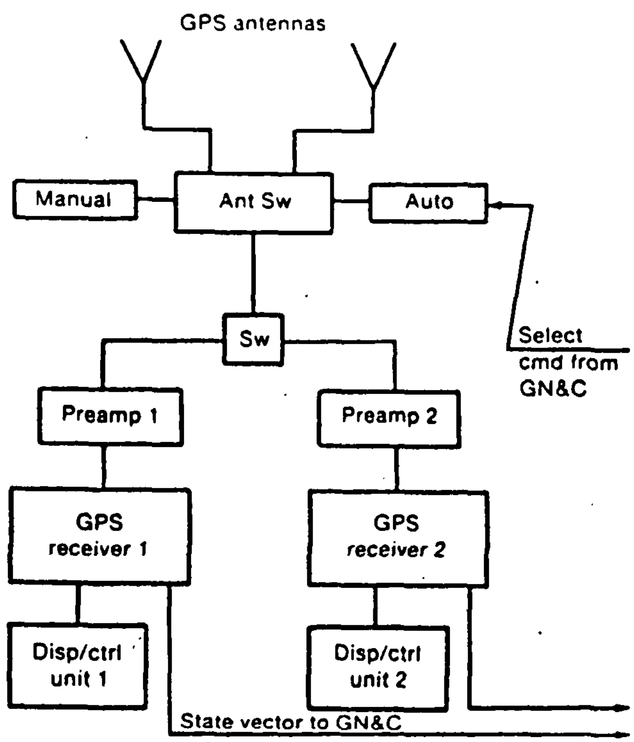
The Communications and Tracking subsystem of the PLS will be used to communicate with the ground station, Space Station, Global Positioning System (GPS), Space Shuttle, and other orbiting targets. The PLS will use UHF (200-300 MHz) for voice communication and data transfer and a GPS system to provide positioning data to the GNC system.

The current S-band communication system of the Space Shuttle was considered for communication link through the Tracking and Data Relay Satellite System (TDRSS), but was rejected because of its complexity, weight, and power requirements. In addition, a UHF system can be used for communications after deorbit where an S-band link cannot. Furthermore, the TDRSS can only provide an 85% on-orbit coverage, whereas the UHF system can provide 100% coverage using existing military UHF satellite relay networks (e.g. SATCOM, LEASAT). The PLS will be equipped with two UHF systems each consisting of two flush mounted antennas, an antenna switch, and two UHF transceivers. A schematic of one of the UHF systems is shown in Figure 2.5.

For tracking purposes the state vector will need to be continually updated to the GNC system. The GPS, with its constellation of satellites, can provide accurate and timely PLS position to the GNC system. The GPS equipment is lightweight, requires low power, and is currently available. The PLS will be equipped with two GPS systems each with two flush mounted antennas, an antenna switch, two GPS receivers and amplifiers. A schematic of one of the GPS systems is shown in Figure 2.5.



UHF System



GPS System

Figure 2.5 Schematics of Communication and Tracking Systems

2.4.9.2 GNC

The GNC subsystem determines the position, velocity, and attitude of the spacecraft. Furthermore, the GNC subsystem processes the state information to relay the necessary changes to the pilot or control systems. The control systems physically change the state of the spacecraft. The GNC subsystem can be divided into three areas: ascent, orbit, and descent and reentry.

In the ascent phase of the mission the GNC will be controlled by the launch vehicle with communication between the PLS, ground station, and launch vehicle for monitoring systems in case abort is necessary.

During the orbiting phase of the mission, communication with the GPS, the ground station, the space station, the space shuttle, or other orbiting targets, as well as horizon, sun, and inertial sensors will be used to determine the state of the spacecraft. These systems have redundancies in that many perform the same task, which is to provide the state of the spacecraft so it can rendezvous with orbital targets, maintain proper attitude with respect to the Earth and Sun, and deorbit.

The GPS and UHF communication equipment has already been discussed. However, the other equipment will consist of two integrated horizon sensors, two digital sun sensors, and two Inertial Measurement Systems (IMS). The IMS will consist of three strap-down ring laser gyros, three accelerometers, and two 16-bit microprocessors with associated input/output for providing the necessary data to the GPC.

The OMS engines (discussed in propulsion section of this report) and RCS jets will be used to change the position, velocity, and attitude of the spacecraft from the information processed by the on-board GNC computers or the pilot. The RCS subsystem will consist of three RCS pods (one fore and two aft) each capable of providing thrust in five directions (up, down, left, right, and fore or aft, depending on location). Each pod will have two independent drivers and its own fuel and oxidizer tanks. The RCS uses hypergolic propellants with nitrogen tetroxide as the oxidizer and monomethal hydrazine as the fuel. During the reentry/descent phase of the mission, communication with GPS and the ground station and inertial guidance sensors will be used to determine the state of the spacecraft relative to the earth and landing sight. The state of the spacecraft will be controlled by the on-board computer and pilot. For reentry, the RCS jets in the fore and aft of the spacecraft will be used to physically control the spacecraft.

Once the PLS is below Mach 3, two pitot-static systems (one on either side and close to the crew window) will be deployed. Each pitot-static system will provide airspeed, angle of attack, and barometric altitude to the on-board computers. In addition, from this velocity the aerodynamic surfaces will be used to control the PLS down to landing. Furthermore, the PLS will be equipped with a radar altimeter which will give altitude and change in altitude data to the computer from 5000 ft to 100 ft.

The GNC subsystems described above will have to perform all their tasks safely, efficiently, and reliably. Therefore, many proven shuttle-type systems or derivatives are used for the PLS.

2.4.10 Suggested Future Studies

The purpose of the PLS portion of the report was to present a conceptual design of a personnel "ferrying" system for Space Station crew rotation and Space Shuttle rescue.

The main areas that further study is needed in completing the PLS are in the areas of aerodynamics and launch vehicle. For this design, a basic aerodynamic study was done leading to a conceptual sketch of the PLS body design. In order to determine the actual body design, a detailed theoretical and experimental analysis is required. Next, the launch system chosen for the PLS was a man-rated version of the FHLLV. This choice was made to integrate the two divisions of our design group. In the future, a study of alternate launch vehicles such as all liquid boosters or air-breathing first-stage platform needs to be completed. These further studies would help to make the PLS more cost effective.

Some of the other areas that were studied in detail are materials, system integration, and docking. With the advent of new aerospace materials, a more aerodynamic and efficient PLS body can be designed. During the design of the PLS, the group did basic analysis on system integration with weight and power requirements. A detailed analysis of the subsystems is necessary for a fully designed PLS. In addition, the docking of the PLS was assumed to be universal, but a detailed analysis is needed to insure that the PLS will be able to dock with the Space Station, Shuttle, and other orbiting targets.

3 FHLLV Project Final Design Review

3.1 Overview

The technical PDR 2 will consist of a discussion on the mission scenarios, assumptions, and system requirements. To satisfy these requirements, the Launch Vehicle Division is separated into three groups: Propulsion Design, Vehicle Design, and Trajectory Design. The Propulsion Design group is responsible for the research and design of that subsystem, including propulsion system, propellant, engine, and feed system selection. The Vehicle Design group is concerned with the launch configuration and systems integration, tank design, weight and sizing analysis, structural analysis, power systems and the feasibility of reusable components. The third group, Trajectory Design is responsible for flight requirements, and the avionics subsystem, including navigation, guidance, controls, and communications.

3.2 Mission Scenarios

The Family of Launch Vehicles will be designed to provide launch services for the following near term missions: Space Station Support, Free Flying Platforms, Commercial Payloads and PLS Support, which will include a payload range of 20 to 100 metric tons. Far term missions include Lunar Base and Mars Mission Support with a payload range of 100 to 200 metric tons.

3.3 Assumptions

The following assumptions have been made for the FHLLV:

1. Approximately five year minimum development plan

2. Orbital Transfer Vehicle will be available to transfer payloads from low Earth orbit (150-250nm) to higher orbits
3. Launch vehicles will be cargo-carrying only (not man-rated, except for the PLS vehicle)
4. Launching from Kennedy Space Center

3.4 System Requirements

The Launch Vehicle Division has established the following system requirements for the Family of Launch Vehicle Design.

1. Safety and reliability
2. Minimal environmental impact
3. Geosynchronous orbit capability
4. Minimize launch readiness time
5. Standardized design for efficient production and component integration
6. Cost effective
7. Standard payload integration

3.5 Final Design Time Line

Figure 3.1 is the final design time line considered appropriate for the evolution of the FHLLV Project. The following subsections detail the several design phases to which the family of launch vehicles is expected to evolve.

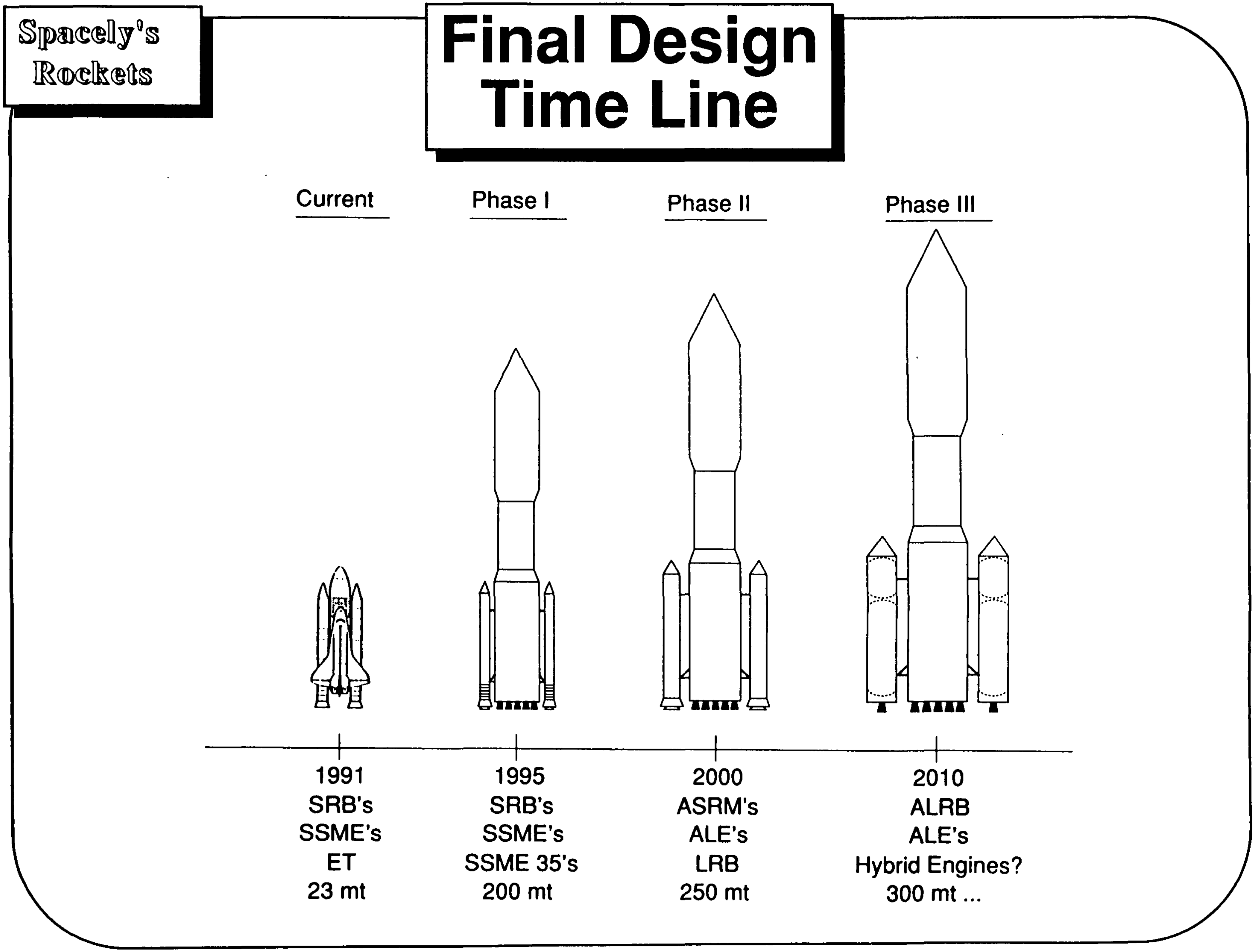


Figure 3.1 Final Design Time Line

3.5.1 Current

At the present time, there is no large, heavy lift options available. The Space Shuttle, with a maximum payload of 23 mt, is only one of a limited number of options for payload launches. However, none of these options can carry the size of payloads expected to be needed during the latter half of the decade.

3.5.2 Phase I

Design Phase I is the core of the FHLLV Project. It is expected to be available as soon as 1995 due to the use of "off-the-shelf" components. This design uses Space Shuttle Main Engines (SSME's) and solid rocket boosters (SRB's), which are propulsion systems currently in use. The addition of SSME 35's (modified SSME's) will boost the launch performance while low in the atmosphere with a minor modification of the existing SSME's. This phase of the design consists of a family of launch vehicles that can launch payloads from 20 mt to 200 mt. Design Phase I is the design detailed in this report.

3.5.3 Phase II

Although not a part of this design, Spacely's Rockets anticipates the evolution of its FHLLV system to include the use of Advanced Solid Rocket Motors (ASRM's). Near the turn of the century, their usage could become common, and augment the FHLLV system by providing a greater lift capacity. The use of Liquid Rocket Boosters (LRB's) and Advanced Liquid Engines is also expected to boost performance.

3.5.4 Phase III

Although it is far in the future, design Phase III is expected to employ all liquid propulsion elements, with a possible use of hybrid engines in ultra-heavy lift configurations. The payload capacity may be boosted to 300 mt or higher, depending on advances in propulsion technology.

3.6 Current FHLLV Design Configuration

Currently, Spacely's Rockets has completed its design for the launch vehicle system depicted in Design Phase I. This family of launch vehicles consists of three basic configurations: the SR-1, the SR-2 and the SR-3. These vehicles are shown in comparison in Figure 3.2.

3.6.1 SR-1 Launch Vehicle Design

The SR-1 is the smallest vehicle in the launch vehicle family. It has a payload capacity of 20 mt to 95 mt depending on the number of SRB's used, and whether or not the second stage is employed. Figure 3.3 illustrates the basic dimensions of the SR-1 in the 72 mt configuration. This configuration employs 2 SRB's and the second stage. The SR-1 can employ two or four SRB's as required to increase the payload capacity.

The first stage of the all liquid-propelled core utilizes three SSME-35's for propulsion. The first stage is a cylindrical structure that houses the oxidizer and fuel for the first stage in separate tanks. The first stage is 31 ft in diameter and 149 ft tall.

The second stage relies on two, unmodified SSME's for thrust. The second stage has a diameter of 24 ft, and a length of 82 ft without the payload shroud.

Overall, the SR-1 stands 357 ft tall, and has an overall width of nearly 70 ft.

The gross lift-off weight (GLOW) and stage weights for the SR-1 are shown in Figure

3.3.

3.6.2 SR-2 Launch Vehicle Design

The SR-2 is the medium capacity vehicle in the launch vehicle family. It has a payload capacity of 40 mt to 150 mt depending on the number of SRB's used, and whether or not the second stage is employed. Figure 3.4 illustrates the basic dimensions of the SR-2 in the 100 mt configuration. This configuration employs 2 SRB's and the second stage. The SR-2 can employ two, four or six SRB's as required to increase the payload capacity.

The first stage of the all liquid-propelled core utilizes five SSME-35's for propulsion. The first stage is a cylindrical structure that houses the oxidizer and fuel for the first stage in separate tanks. The first stage is 40 ft in diameter and 149 ft tall.

The second stage relies on two or three, unmodified SSME's as needed for thrust. The second stage has a diameter of 31 ft, and a length of 82 ft without the payload shroud.

Overall, the SR-2 stands 384 ft tall, and has an overall width of nearly 76 ft.

The gross lift-off weight (GLOW) and stage weights for the SR-2 are shown in Figure 3.4.

3.6.3 SR-3 Launch Vehicle Design

The SR-3 is the largest vehicle in the launch vehicle family. It has a payload capacity of 140 mt to 200 mt depending on the number of SRB's used. Figure 3.5 illustrates the basic dimensions of the SR-3 in the 200 mt configuration. This configuration employs six SRB's. The SR-3 can mount two, four, six or eight SRB's as required to increase the payload capacity.

The first stage of the all liquid-propelled core utilizes eight SSME-35's for propulsion. The first stage is a cylindrical structure that houses the oxidizer and fuel for the first stage in separate tanks. The first stage is 50 ft in diameter and 149 ft tall.

The second stage relies on two or three, unmodified SSME's as needed for thrust. The second stage has a diameter of 40 ft, and a length of 82 ft without the payload shroud.

Overall, the SR-3 stands 440 ft tall, and has an overall width of nearly 86 ft.

The gross lift-off weight and stage weights for the SR-3 are shown in Figure 3.5.

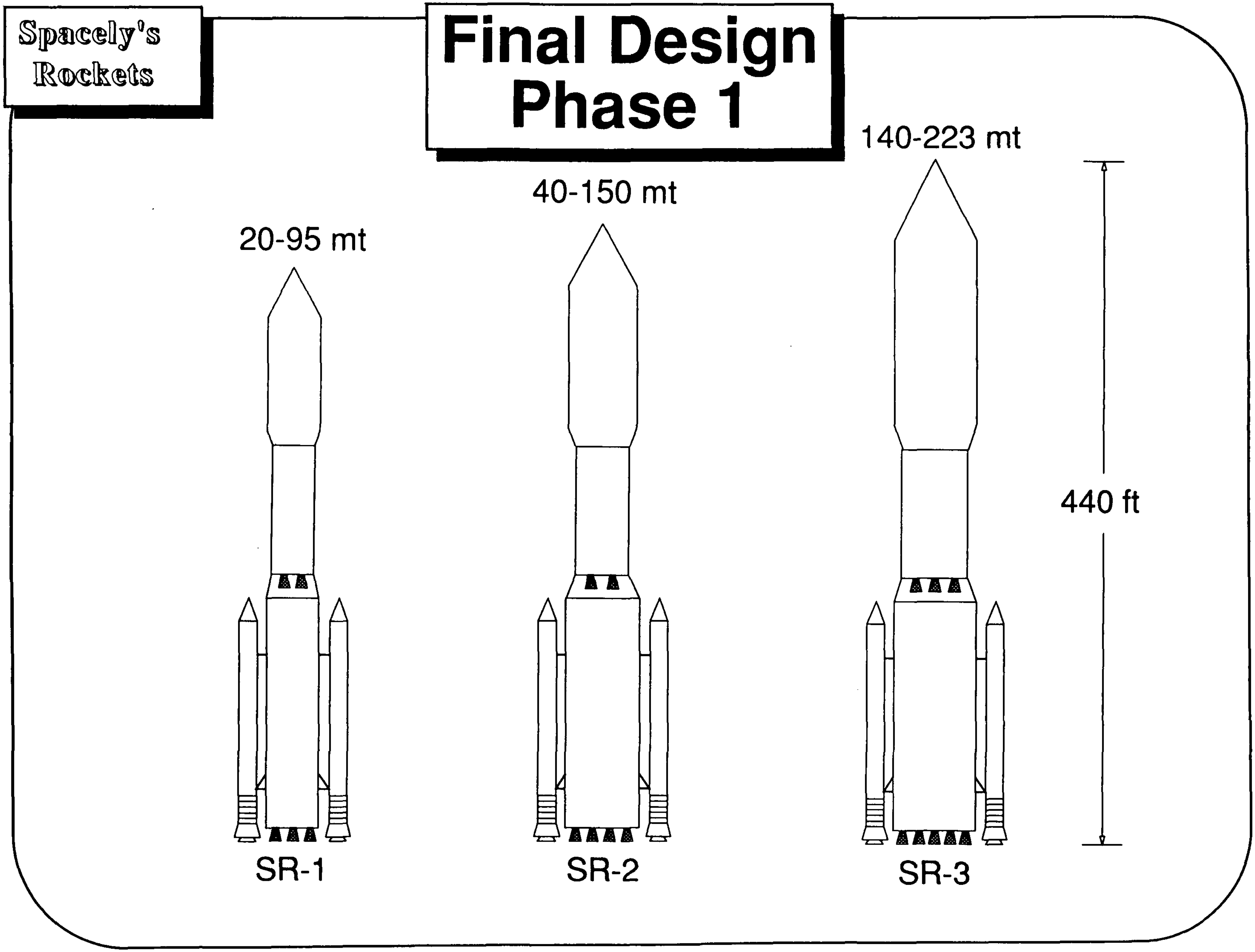


Figure 3.2 Launch Vehicle Family

Spacely's Rockets

72 MT Capacity

GLOW	5,328,660 lb
Payload Shroud	3307
2nd Stage	
Structural Mass	47,058
Propellant Mass	306,004
2 SSME's	
1st Stage	
Structural Mass	122,355
Propellant Mass	1,178,083
3 SSME-35's	
2 SRB's	

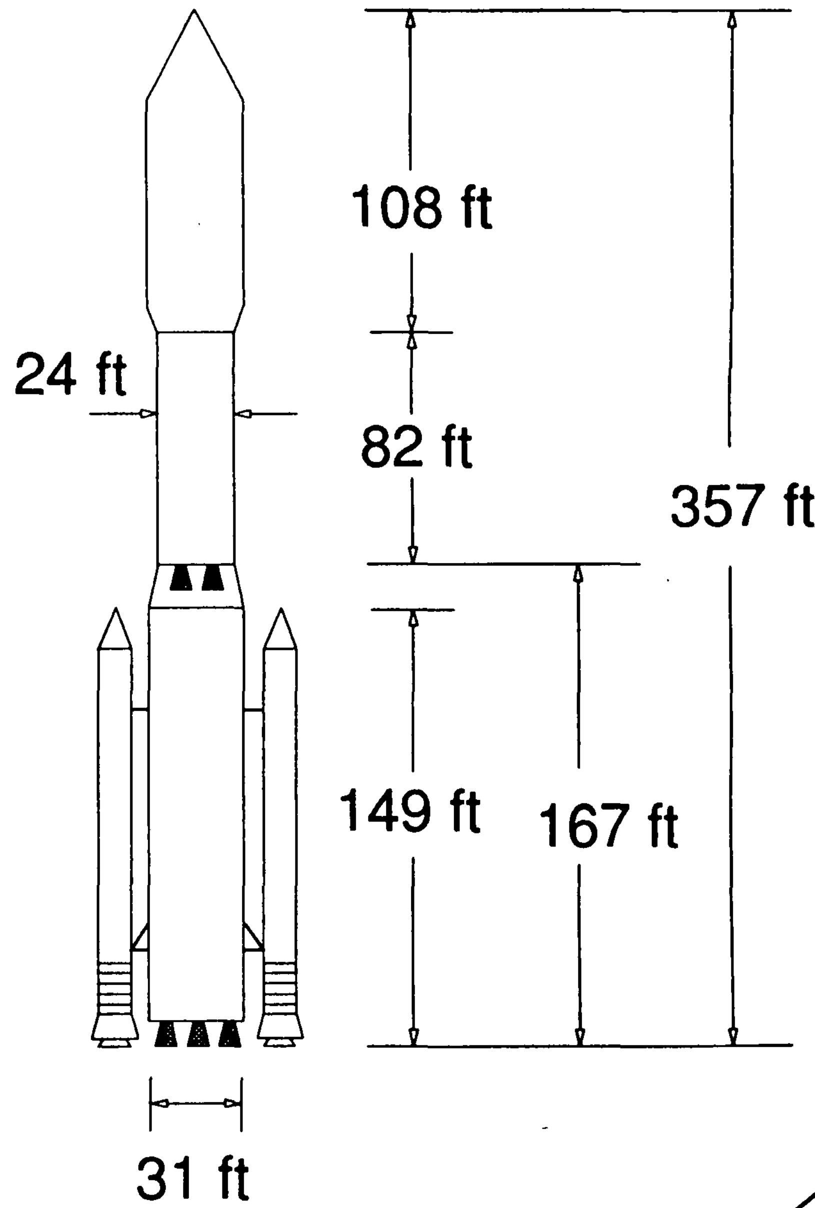


Figure 3.3 SR-1 Launch Vehicle

Spacely's Rockets

100 MT Capacity

GLOW	5,496,064 lb
Payload Shroud	6,614
2nd Stage	
Structural Mass	78,446
Propellant Mass	509,915
2 SSME's	
1st Stage	
Structural Mass	203,925
Propellant Mass	1,963,472
5 SSME-35's	
2 SRB's	

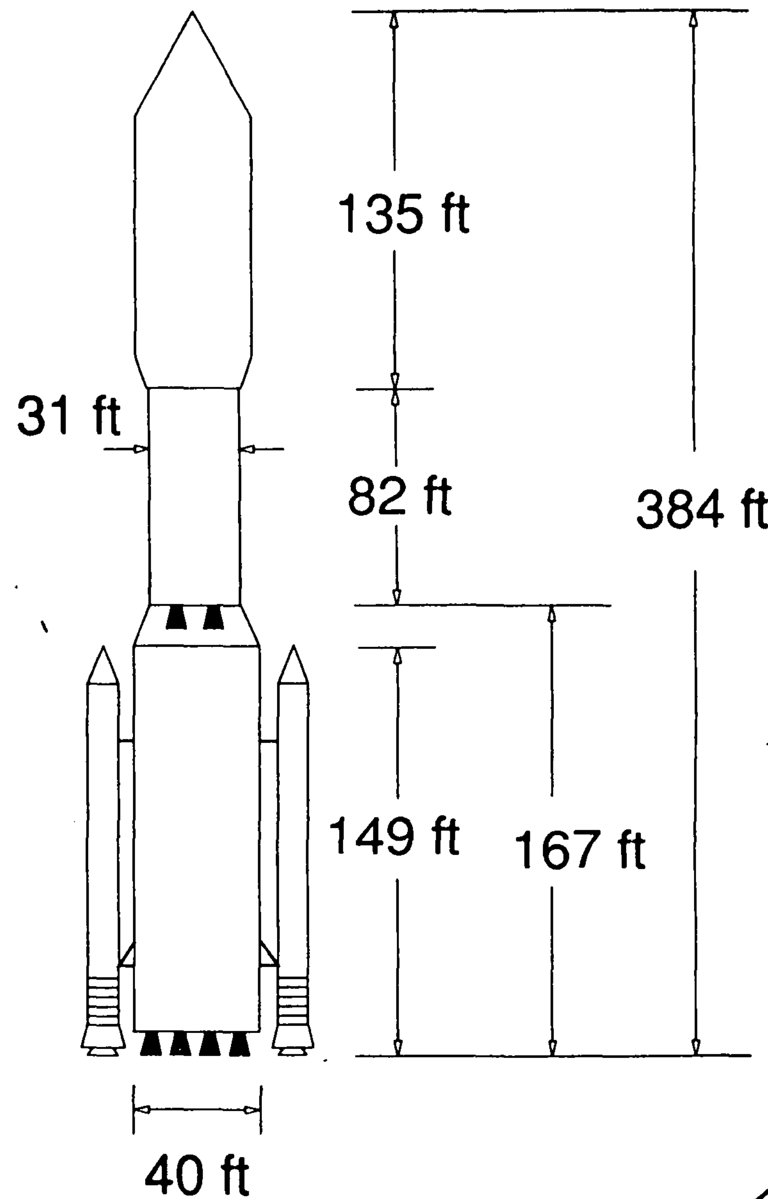


Figure 3.4 SR-2 Launch Vehicle

Spacely's Rockets

200 MT Capacity

GLOW	12,399,460 lb
Payload Shroud	11,023
2nd Stage	
Structural Mass	125,662
Propellant Mass	817,702
3 SSME's	
1st Stage	
Structural Mass	326,281
Propellant Mass	3,141,555
8 SSME-35's	
6 SRB's	

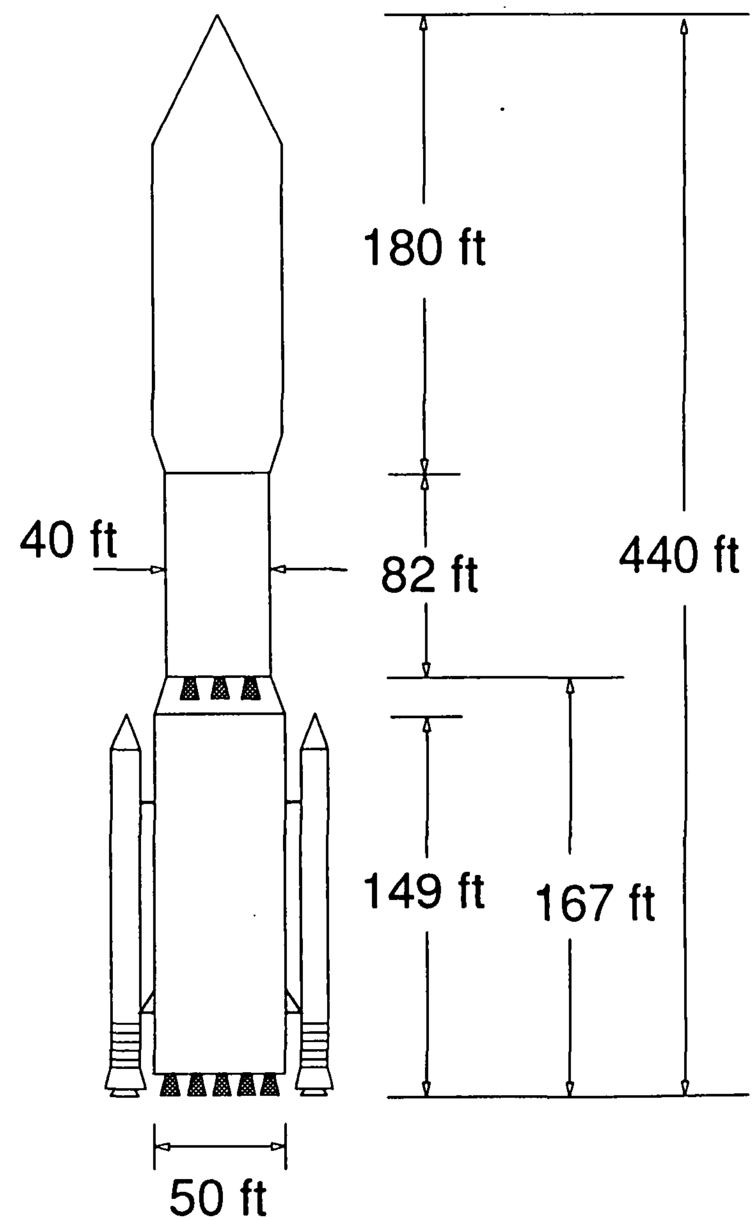


Figure 3.5 SR-3 Launch Vehicle

3.7 Technical Research Areas

3.7.1 Propulsion

The propulsion system for the FHLLV must provide high thrust and high specific impulse in order to carry a payload up to 200 metric tons to GEO. Currently, chemical propellants are the only ones to meet these requirements with available technology on a cost-effective, reliable, and environmentally safe basis. The three classes of chemical propellants are solids, liquids, and hybrids. Each of these classes has desirable and undesirable characteristics of a propulsion system for an Earth launched rocket. Selection of one or a combination of these for the FHLLV is based on the propellant which most closely meets the previous requirements.

3.7.1.1 Solid Motors

Solid rocket motors consist of a hollow cylinder of a rubbery propellant which contains a mixture of both the fuel and the oxidizer. Once ignited, solid motors burn until their fuel is exhausted; the rate of burn and the amount of thrust produced is controlled only by the shape of the propellant grain and the use of inhibitors which restrict burning on some surfaces of the propellant. Specifically, the thrust of the solid motor is controlled by the exhaust velocity and the burning rate. Large thrust requires either a large burning surface or a fast burning rate, which in turn requires a high grain temperature and chamber pressure. Using a high percentage of oxidizer allows for higher specific impulses as does the addition of light metals such as aluminum. The fuel binder gives the solid motor its mechanical and structural strength, thus allowing for a thin and light outer metal or composite casing.

Solid motors generally provide a propulsion system with high thrust, excellent reliability (98%), easy storability, simplicity, instant readiness, potential for high acceleration, high density, and inexpensive production because of their established use. However, they usually have low specific impulse, no throttling capability, and are heavy polluters.

3.7.1.2 Hybrid Engines

Hybrid engines combine some of the advantages of both solid and liquid engines. Typically, they have a rubbery fuel similar to the solid motor grains, but use a liquid oxidizer that is sprayed from a nozzle down the central bore of the fuel cylinder. Like a solid motor, the engine burns from the inside outward.

Hybrid engines offer the advantages of storing the fuel and oxidizer in separate tanks, thus improving safety and the structural properties of the fuel grain (because of the removal of the oxidizer) and also eliminating the dependence of burning time on grain temperature. In addition, hybrid engines have start-stop and throttling capabilities, high density, instant readiness, potentially high acceleration, easy storage, and simplicity. The main drawback of hybrid engines is the unproven and undeveloped technology. Also, hybrid engines currently yield only moderate specific impulses and thrust values while exhausting heavy pollutants. The lack of development also makes costs unknown for designing a large hybrid engine.

3.7.1.3 Liquid Engines

Liquid engines have a fuel and an oxidizer which can be combined in one substance (monopropellant) or separated into two substances (bipropellant) carried in separate tanks. The fuel and oxidizer must be pumped or pressure fed rapidly into the combustion chamber where they ignite. Fuels and oxidizers may be liquids at room temperature, storable propellants, or cryogenic propellants. Typically, cryogenics have much higher specific impulses than storable propellants and are considered high energy propellants. Chamber pressure, altitude, and nozzle expansion determine the performance of the propellants.

Overall, liquid engines give high thrust and specific impulses, start-stop and throttling capability, reliability (96%), and non-polluting exhaust gases. Complexity and high cost due to the preparation and storage of cryogenics are the main drawbacks. Also, for large engines to be efficient and effective, the turbopumps (in the case of a pump fed system) must provide high propellant pressure to give high specific impulse values and be reliable to ensure that fuel and oxidizer do not mix in the pumps and explode.

3.7.1.4 Propulsion System Selection

A decision matrix was used to weigh the advantages and disadvantages of each propulsion system and to determine the best one for the FHLLV. A priority ranking scale was used to rate the importance of the criteria to meet the design goals of the FHLLV using 4 as the highest rating and 1 the lowest. The matrix is shown in Figure 3.6 and is based on the following criteria listed in decreasing order of importance:

Performance	A rating of the thrust, specific impulse, efficiency, and controllability of the system.
Estimated Mission Reliability	A rating of the probability that the system will perform as designed without failure, losses in performance, or defects.

Environmental Impact	A rating of the negative impact the propellants have on the ground and in the upper atmosphere.
Availability	A rating of the cost and degree of difficulty to which propellants can be acquired.
Storability	A rating of the complexity, difficulty, safety, and cost of storing the various propellants.

The decision matrix shows that liquid engines are the best choice for the FHLLV propulsion system because of the need for high thrust and high specific impulse. However, solid motors have been chosen for boosters because liquid boosters are much too large to be implemented in a flexible system of launch vehicles. SRB's are targeted to be used in the FHLLV system for the first five years of it development. Later, they will be replaced by liquid boosters or ASRM's when the technology provides more efficient thrust. This decision analysis shows a strong need for further advances in high thrust, non-polluting solid propulsion

Criteria	Priority	Solid	Hybrid	Liquid
Performance	4	Fair 8	Fair 8	Good 12
Estimated Mission Reliability	3	Good 9	Fair 6	Fair 6
Environmental Impact	3	Poor 3	Fair 6	Good 9
Availability	2	Good 6	Poor 2	Good 6
Storability	1	Good 3	Fair 2	Fair 2
	Totals	29	24	35

Ranking: Poor - 1
Fair - 2
Good - 3

Figure 3.6 Propulsion Systems Decision Matrix

3.7.1.5 Reasons for Using SRB's

Solid rocket motors are effective for providing a large amount of additional thrust when size and weight constraints for boosters are very restrictive. For the FHLLV, solid boosters provide the extra thrust necessary for the vehicles to have an initial thrust to weight ratio greater than one. They can be sized shorter than the first stage and have a smaller diameter than the core stage. On the other hand, liquid boosters providing equivalent thrust with the same diameter would be two to three times taller than solids because of large propellant volume requirements. Because the core stages use liquids and are throttlable, there is no real need for the engines providing "boost" thrust for short durations to have throttling capability. Liquid boosters also add additional structural weight because pumps and electronic control systems are needed to monitor and maintain proper tank pressures to control the rate of flow of the propellants. All of these additional requirements increase cost, complexity and preparation time. At the present time, Spacely's rockets has chosen to trade off the negative environmental impacts of solid boosters for their positive aspects of small size, high thrust, and simplicity.

3.7.2 Liquid Propellant Selection

In order to achieve a high lift capacity, the FHLLV must maximize engine performance. Therefore, the selection of the of both oxidizer and fuel is critical to the effectiveness of launch system. The performance of various liquid propellants was compared on the basis of specific impulse, exhaust velocity and specific propellant consumption.

For high performance, the propellants must have a high content of chemical energy to produce high combustion chamber temperatures. A low molecular mass of the products of combustion is also highly desirable. Therefore, only fuels rich in free or combined hydrogen were considered for the FHLLV project.

The following is a brief description of the physical hazards which should be avoided when selecting rocket engine propellants:

1. <u>Corrosion Hazards</u>
   Those propellants that require special material containers for storage, or react with products of corrosion should be avoided. (examples: Nitrogen Tetroxide, Hydrogen Peroxide)

2. <u>Explosion Hazards</u>
   Those propellants that are highly unstable, or tend to detonate under temperature or shock should not be used. (examples: Hydrogen Peroxide, Nitromethane)

3. Fire Hazards
   Those oxidizers that ignite spontaneously upon contact with organic materials are expensive to handle, and should be avoided. (examples: Nitric Acid, Nitrogen Tetroxide, Hydrogen Peroxide)

4. Toxicity
   Those propellants which are poisonous to the environment or to the personnel at the launch site should not be used. (example: Fluorine)

The above hazards limit the type of materials, storage conditions and design of the overall rocket system. Since structural parts, electrical subsystems and launch personnel may be exposed to these hazards, the decision on the type of propellants used must include safety of personnel and launch vehicle in addition to high performance.

3.7.2.1 Oxidizer Selection

The following five oxidizers were considered for use by the FHLLV project:

1. Liquid Oxygen (O_2)
2. Liquid Fluorine (F)
3. Nitrogen Tetroxide (N_2O_4)
4. Nitric Acid (HNO_3)
5. Hydrogen Peroxide (H_20_2)

Although it has the highest performance potential, liquid fluorine was rejected due to its high toxicity and corrosiveness. Likewise, nitric acid is poisonous in most forms, and has fairly restrictive storage options. Both hydrogen peroxide and nitrogen tetroxide may ignite spontaneously on contact with wood, oils and many other organic compounds. From the oxidizers mentioned above, only liquid oxygen combines a minimum of the aforementioned hazards with high performance.

Because of its relative ease in handling, liquid oxygen is the oxidizer of choice for the FHLLV project. Liquid oxygen is noncorrosive and nontoxic. It will not deteriorate the container walls of its storage vessel, however; it cannot be stored for prolonged periods due to rapid evaporation. Because of this tendency to evaporate, all lines, tanks and valves that contain liquid oxygen must be insulated well. The storage tanks of this oxidizer must be insulated against heat absorption from the surroundings. Since water always condenses on the walls of vessels that contain the supercooled liquid oxygen, external drainage systems must be provided to eliminate the water buildup within the rocket propulsion system.

3.7.2.2 Fuel Selection

The following five fuels were considered for use by the FHLLV project:

1. Hydrocarbon Fuels
2. Liquid Hydrogen (LH_2)
3. Hydrazine (N_2H_4)
4. Unsymmetrical Dimethylhydrazine [$(CH_3)_2NNH_2$]
5. Monomethylhydrazine (CH_3NHNH_2)

From the above list, Hydrazine, Unsymmetrical Dimethylhydrazine (UDMH) and Monomethylhydrazine (MMH) are toxic and ignite easily with exposure to polar organic compounds that include often used lubricants. The hydrazine family is also expensive to produce in the large quantities required by a heavy lift rocket. Therefore, the last three fuels on the list are rejected for use by the FHLLV.

Hydrocarbon fuels (like diesel fuel, gasoline and RP-1) are petroleum derivatives that are usually inexpensive to manufacture. They vary in performance based upon their composition. A high performance hydrocarbon known as RP-1 has been widely used as a rocket propellant and performs well. It is relatively easy to store and handle. However, the burning of hydrocarbon fuels produces some upper atmospheric pollution.

Liquid hydrogen imparts excellent thrust capabilities to liquid engines due to its low molecular weight. It is also a good regenerative coolant, and delivers no pollution when burned with liquid oxygen. However, due to hydrogen's low density, very bulky fuel tanks are required. The fuel must be stored at very low temperatures, and all tanks, fuel lines and valves must be well insulated to prevent evaporation of hydrogen. The same tanks and lines must also be well sealed to prevent the condensation of water or air into ice and solid air which could enter the propulsion system, and interfere with its operation. Liquid hydrogen is explosive when mixed with air.

From the above list, two fuels were considered appropriate for the FHLLV: liquid hydrogen and RP-1. Even though LH_2 is obviously harder to handle, much experience with this fuel exists in its use with the Space Shuttle and Centaur Upper Stage rockets. Liquid hydrogen's superior performance justifies its consideration for the FHLLV engines.

3.7.3 Liquid Propellant Feed Systems

Two types of propellant feed systems were considered for this design: pressure fed and pump fed.

3.7.3.1 Pressure Feed Systems

Pressure fed systems rely on pressurized fuel and oxidizer tanks to fill the combustion chamber of the rocket engines. To accomplish this, a high pressure gas is fed into the propellant tanks at a controlled rate thereby providing a controlled propellant discharge. This is the simplest and most common feed system to date, however it is primarily used on small propulsion systems. It is the system of choice for low-thrust applications and space vehicle attitude control. Large scale versions of this system were considered, but are still unproven and require further development.

3.7.3.2 Pump Feed Systems

Turbopump feed systems utilize turbine driven pumps to pressurize the propellants prior to combustion. The turbines are driven, in turn, by the expansion of the post-combustion gases. Turbopump systems are usually used on high-thrust, long duration missions. Because of their complexity, these pumps require high-tolerance manufacturing to achieve reliable performance, and are relatively expensive. However, the use of turbopumps is an established technology that has the benefit of a very flexible throttling capability.

3.7.3.3 Feed System/Propellant Selection

In order to select the optimal propellant and feed system, a decision matrix was used to rank each of the options. The options were:

1. LO_2/LH_2 & Turbopump Feed System
2. LO_2/RP1 & Turbopump Feed System
3. LO_2/RP1 & Pressure Feed System

Note that a LO_2/LH_2 & Pressure feed system combination was not included in the options. This is because the LO_2/LH_2 combination under constant high pressure at launch is very dangerous. If a high pressure leak developed, the contents of the propellant tanks would, most likely, explosively erupt from their pressure vessel, and devastate the surrounding area.

The decision matrix is shown in Figure 3.7. A priority ranking scale was used to rate the importance of the criteria to meet the design goals of the FHLLV using 3 as the highest rating and 1 the lowest. The matrix considers the following criteria:

Economic Risk	A rating of the projected success of constructing the system. A low risk is desirable and can be achieved by utilizing technology that is currently available.
Evolution and Growth	A rating of the system's usefulness in concurrent or future project that can utilize similar engines. Good means that the system is flexible enough to be used in many applications.
Environmental Impact	A rating of the environmental hazards that the system and its propellants could pose. Good means low environmental impact.
Cost	A rating of the relative cost of the three systems. Good means relatively low cost.

Of the three systems, the LO_2/LH_2 propellants with a turbopump feed system provide the optimal engine performance and economic effectiveness desired for use in the FHLLV.

Criteria	Priority	LO2/LH2 Pump Fed	LO2/RP-1 Pump Fed	LO2/RP-1 Press. Fed
Economic Risk	3	Good 9	Good 9	Poor 3
Evolution and Growth	2	Very Good 8	Good 6	Fair 4
Environmental Impact	2	Good 6	Fair 4	Fair 4
Cost	1	Good 3	Good 3	Fair 2
	Totals	26	22	13

Ranking:
Poor - 1
Fair - 2
Good - 3
Very Good - 4

Figure 3.7 Propellant and Feed System Decision Matrix

3.7.4 Liquid Engine Selection

The following liquid engines were considered for use on the FHLLV:

1. Space Shuttle Main Engines (SSMEs)
2. Modified SSMEs
3. Advanced Launch System (ALS) engines

Each of the above engines were tested in an ascent/propulsion computer model to determine if the engines produce enough thrust with acceptable fuel consumption to accomplish a heavy lift mission.

The SSMEs considered were typical Space Shuttle engines with no modified components. The modified SSMEs share the same combustion chamber, throat area and feed system as regular SSMEs, however, they have different nozzle exit area and expansion ratios. The ALS engines are high thrust engines being developed for a heavy lift system under consideration by NASA and the US Air Force.

Due to the five year development plan, the ALS engine was eliminated for Phase I. To meet the time restrictions, SSME's and modified SSME's are utilized.

3.7.5 Tank Configuration

The LO_2/LH_2 liquid rocket engine system has an oxidizer tank and fuel tank. The tanks can be arranged in a variety of ways and the arrangement can be used for limited control in the change of the location of the center of gravity. Four typical tank configurations are shown in Figure 3.8. Spherical tanks provide the optimum shape to minimize mass but are not very desirable for vehicles flying in the atmosphere. Therefore, cylindrical tanks will be used. A decision matrix was employed to determine the best tank arrangement of the ones shown in Figure 3.8. A priority ranking scale was used to rate the importance of the criteria to meet the design goals of the FHLLV using 3 as the highest rating and 1 the lowest. The decision matrix is shown in Figure 3.9 and is based on the following criteria listed in decreasing order of importance:

Safety	A rating of the ease in providing a safe tank arrangement - such as ease of crack detection. A good rating corresponds to a safe arrangement.
Flexibility	A rating of the adaptation of the arrangement for different propellant volumes. A good rating corresponds to high flexibility.

Structural Mass	A rating of the structural mass required for the arrangement. A good rating corresponds to low structural mass.
Simplicity	A rating of the simplicity of the connections, construction, and filling of the tanks. A good rating corresponds to a simple arrangement.

Of the four arrangements, the tandem tanks with external piping provides the best design.

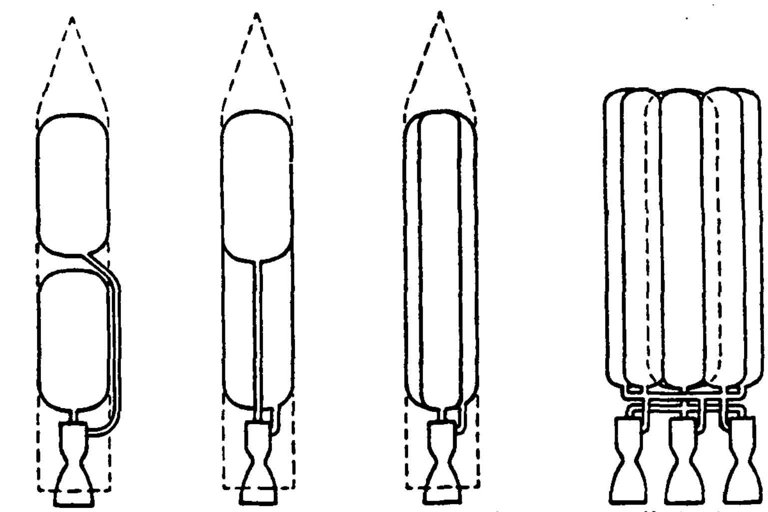


Figure 3.8 Typical Tank Arrangements

Since the propellant will be cryogenic, the tanks will include vents or other pressure relief provisions to prevent self-over-pressurization. Also, insulation of the tanks will be required to lessen the continuous evaporation of the cryogenic fluid. These designs will be included in the tandem tanks with external piping arrangement.

Criteria	Priority	Tandem Tanks	Tandem Tanks With Bulkhead	Concentric Tanks	Multiple Tanks
Safety	3	Good 9	Fair 6	Poor 3	Good 9
Flexibility	2	Good 6	Fair 4	Fair 4	Good 6
Structural Mass	2	Fair 4	Good 6	Fair 4	Poor 2
Simplicity	1	Good 3	Good 3	Good 3	Poor 1
	Totals	22	19	14	18

Ranking: Poor - 1
Fair - 2
Good - 3

Figure 3.9 Tank Design Decision Matrix

3.7.6 Launch Vehicle Sizing Model

A sizing model was supplied by Hugh Davis of Davis Aerospace which provided payload weight to orbit, vehicle size, and GLOW for a single core and liquid boosters. Figure 3.10 describes the model structure and the required input. The model accounts for aerodynamic loses and the changing characteristics of the atmosphere with altitude. An iterative procedure between the vehicle sizing routine and tank sizing routine is employed to provide the GLOW.

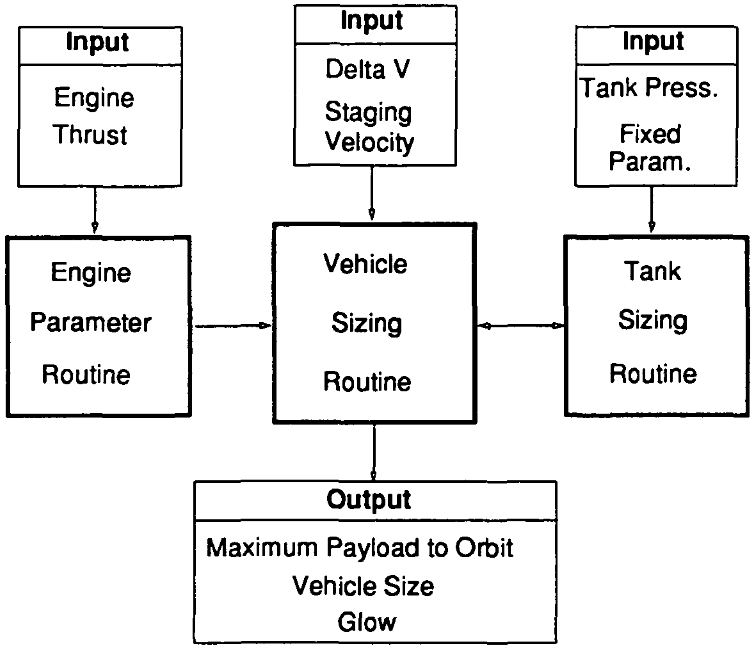


Figure 3.10 Sizing Model Structure

3.7.7 Avionics

The avionics subsystem includes a guidance and navigation system, software, thrust vector control system, telemetry, tracking, and command system, and a separation system. Each of these systems will employ varying degrees of redundancy.

3.7.7.1 Guidance and Navigation System

The guidance and navigation system will consist of an Inertial Navigation System (INS), a Global Positioning System (GPS) radio receiver, and General Purpose Computers (GPC's).

3.7.7.1.1 Inertial Navigation System

An inertial navigation system provides a completely self-contained system that is used to measure the accelerations and rotations of the vehicle with respect to an inertial reference frame. The system will use three two-degree-of-freedom gyroscopes, allowing measurements to be made about the pitch, yaw and roll axes, with one degree of redundancy about each axis. These measurements provide inputs to a navigation computer system that calculates the vehicle's current velocity and position. Three INS's are needed for redundancy, providing protection against a total system failure.

3.7.7.1.2 Global Positioning System

The Global Positioning System will be used in conjunction with the INS to measure the position and velocity of the vehicle. The GPS is a radio navigation system using on-ground computers to calculate the desired vehicle orientation needed to reach its target conditions. A radio transmitter is used to relay the results from the ground computers to the vehicle. A Collins Navstar Receiver with one antenna, antenna electronics unit and two GPS receivers can determine the position of the vehicle to within 55 ft and the velocity to within 0.4 ft/sec.

3.7.7.1.3 General Purpose Computers

Five general purpose computers will be dedicated to the guidance, navigation, and control of the vehicle, as well as monitoring engine performance and calculating the telemetry and tracking commands. They will be located in the second stage which will enable them to remain available through the entire ascent phase of the trajectory.

The GPC's use the outputs from the INS and the vehicles last state to determine the current position and velocity. Guidance algorithms are then used to determine the pitch, yaw and roll angles. After determining these angles, the GPC's send commands to the thrust vector control system to steer the vehicle onto the desired trajectory.

3.7.7.2 Software

The software used for the FHLLV must be able to support multiple vehicle configurations. For example, it must be flexible enough to handle varying numbers of boosters, core stages and payload ranges. The software needed can be separated into two different categories: non-recurring and recurring. Non-recurring software includes guidance and control algorithms, libraries of common routines and the operating system. Recurring software includes all mission specific data such as payload, engine performance data and target conditions (ie. altitude, inclination, etc.).

3.7.7.3 Thrust Vector Control System

A thrust vector control (TVC) system will be required to maintain the vehicle attitude and attitude rates within safe limits during the region of high dynamic pressure and direct the thrust vector to provide velocity corrections commanded by the navigation and guidance system. This system receives turning or guidance steering commands to change the direction of the applied rocket acceleration. Engine swivels or gimbal arrangements will be used on the core engines to deflect these forces from the center of mass to achieve control torques. The SRB's can also be used for thrust vector control by using a flexible joint system which is used to pivot the nozzle. The engines and boosters are arranged so that control torques can be applied about the pitch, yaw, and roll axes. Figure 3.11 shows a typical arrangement for a configuration of eight SSME's and six SRB's.

A control system will also accomplish the following tasks for the propulsion control valve actuators:

1. Start rocket operation.
2. Shut down rocket operation.
3. Restart, if desired.
4. Maintain programmed operation (predetermined constant or varied thrust, preset propellant mixture ratio and flow).
5. Make emergency shutdown when safety devices sense a malfunction or a critical condition of the vehicle or the engine.
6. Fill with propellants.
7. Drain excess propellant after operation.
8. Check-out proper functioning of critical components or a group of components without actual hot operation before flight.

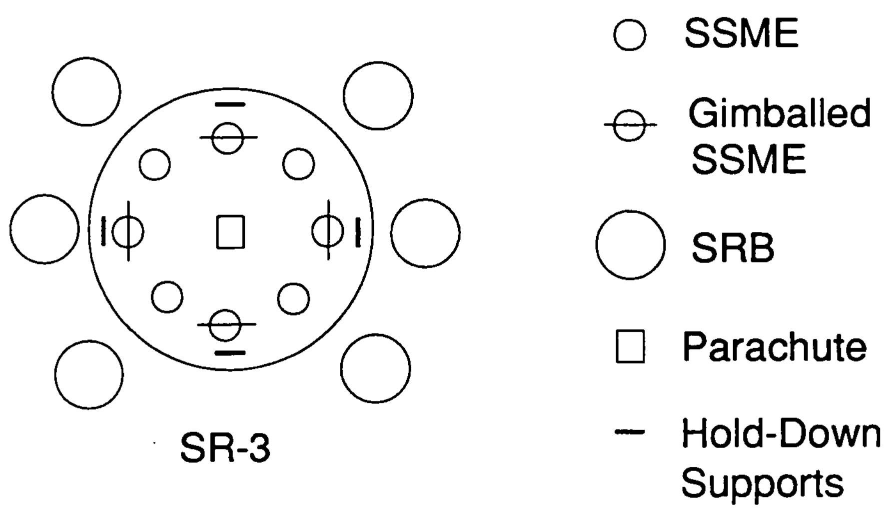


Figure 3.11 Thrust Vector Control

3.7.7.4 Telemetry, Tracking and Command

Radar / radio tracking will be used as a monitor to confirm that the rocket is achieving the desired trajectory. S-band electronics and antennae will be used for uplink and downlink commands during the ascent phase of the trajectory. Range safety electronics and antennae will be used to protect populated land masses and to control any reusable components of the vehicle. The vehicle will rely on the Tracking and Data Relay Satellite System (TDRSS).

TDRSS satellites are currently being used by the Space Shuttle Transportation System. Radio signals from the ground are uplinked to a TDRSS satellite and relayed to the launch vehicle. Signals from the launch vehicle go back to the TDRSS satellite and are downlinked to the ground station. TDRSS satellites use multiple access S-band and single access S and K-band frequencies. Figure 3.12 illustrates the telemetry, tracking and command system for the launch vehicle.

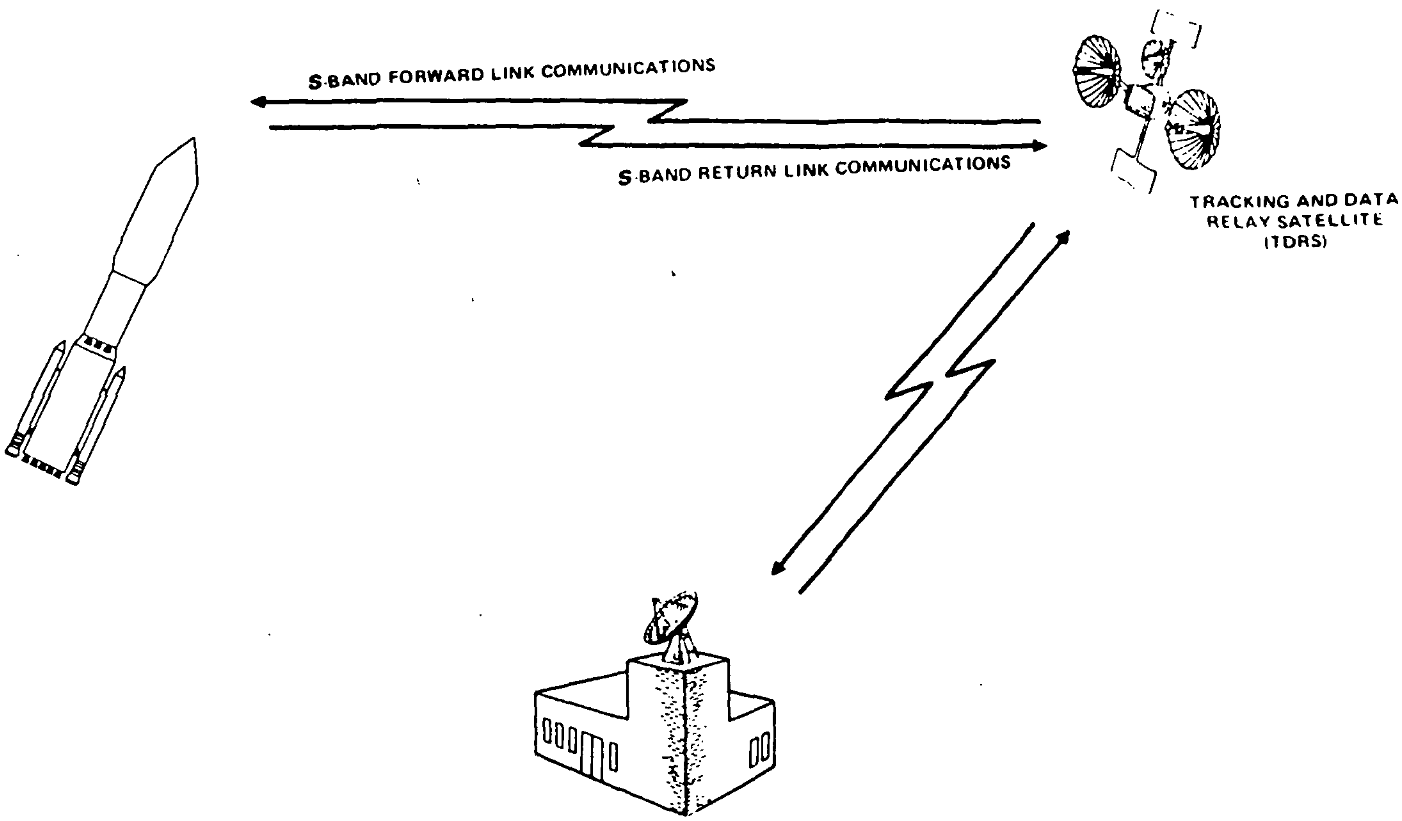


Figure 3.12 Telemetry, Tracking, and Command

3.7.7.5 Separation System

A separation system is needed to detach the SRB's from the lower stage and the lower stage from the upper stage. All trajectory staging will be initiated through the on-board GPC's. Core staging will employ Saturn V type technology with explosive bolts and retro-rockets to separate the stages. The SRB's will use the separation scheme that is currently being used by the Space Shuttle. Explosive bolts are used to separate the SRB from the first core stage and retro-rockets at the top and bottom of the booster are fired to guide it away from the vehicle.

3.7.8 Power

The type and size of the power subsystem for the FHLLV is dependent on the power requirements of all other subsystems. The power system will be limited to providing power to the launch vehicle only since the payload is assumed to be self-sustained. It must also be reliable since all aspects of a successful mission depend on power being supplied to the other systems.

Auxiliary Power Units (APU's) will be used to supply power for the thrust vector control of the core engines. APU's are currently being used to power the SSME's on the Space Shuttle. The weight and power output per engine are listed below.

Power = 100 KWatts

APU = 90 lbs

Fuel = 310 lbs

Total APU Wt / engine = 400 lbs

A 45 horsepower turboelectric generator will be used to power a majority of the core stage subsystems, excluding the thrust vector control. The power output per generator is 33615 Watts with a weight of approximately 90 lbs. Sodium-Sulfur batteries will provide backup power for the core and total power for the boosters. The boosters and the first core stage will need power for approximately six hours. The second core stage will be recovered from orbit and will require power for approximately 48 hours. The energy density of this type of battery is 200 Wh/kg. The power requirement and weight of the battery for each stage is given below.

Booster:
Power = 1460 Watts
Duration = 6 hrs
Weight = 100 lbs

Core:
1st Stage
Power = 1460 Watts
Duration = 6 hrs
Weight = 100 lbs

2nd Stage
Power = 1460 Watts
Duration = 48 hrs
Weight = 775 lbs

Tables 3.1 and 3.2 give a break down of the power requirements of some of the components of the Avionics subsystem.

Table 3.1 Breakdown of Core Avionics.

Hardware	Number	Weight (lbs)	Power (watts)
INS	3	120	300
Flight Computer	5	125	275
Remote Control Unit	3	180	240
Master Data Unit	3	36	240
Remote Data Unit	3	36	150
Electrical Distribution Unit	1	65	0
Launch Processing Interface	-	20	0
C-Band Electronics	-	35	200
C-Band Antenna	1	2	0
Range Safety Electronics	2	50	50
Range Safety Antenna	2	4	0
Pyrotechnic Initiators	40	20	5
Cabling	-	250	0
		943	1460

Table 3.2 Breakdown of Booster Avionics.

Hardware	Number	Weight (lbs)	Power (watts)
Remote Control Unit	2	120	160
Remote Data Unit	1	12	50
C-Band Electronics	-	35	200
C-Band Antenna	1	2	0
Range Safety Electronics	-	50	50
Range Safety Antenna	1	4	0
TVC	-	3200	1000
		3423	1460

3.7.9 Structures

The structures subsystem includes selection of the materials for the launch vehicle structure and propellant tanks, tank sizing to withstand all loadings, and payload shroud development. The structures subsystem contributes a significant part of the weight to the launch vehicle and an accurate assessment is needed for a more robust vehicle sizing analysis.

The structure of the core stage and boosters must support the propellant tanks, payload, and subsystem devices subjected to all loading conditions. The structural requirements for the core stages can vary due to the different number of SRB's and the different payload weights. Three options were considered to account for these varying structural requirements of the core stage:

1. One core structure capable of supporting all combinations of boosters and payload weights.
2. Several core structures for different combinations of boosters and payload weights.
3. One core stage with the option of adding support members as needed for different combinations of booster and payload weights.

The first option has a weight penalty for smaller payloads and booster combinations, but allows for one design and therefore less production cost. The second option reduces the weight penalty but increases the production costs. Finally, option three provides a greater flexibility but also increases the complexity of launch preparation. The first option using one design for all configurations for each of the three sized core stage designs was chosen. The underlying criteria for this selection is simplicity and reduced production costs. This single design methodology is implemented in the design of the booster connections and core structure which are discussed in the following sections.

3.7.9.1 Materials

The materials used in the structure and tanks of the FHLLV must be light weight to minimize the mass. The structural material must also be able to withstand the loadings related with launch and trajectory. Whereas the tank material must withstand these loadings including the temperature effects due to the cryogenic fluids associated with the propellant. Therefore, since the loadings for the tanks include very low temperatures, brittle behavior was anticipated when choosing the material. From these requirements, the main material for support structures and tank construction will be 2219-T87 aluminum. This is same material used for the external tank of the Space Shuttle. The material was chosen for its ability to handle the cryogenic conditions of the propellants, high strength to weight ratio, and welding compatibility with other structural members.

3.7.9.2 Tank Design

Failure of one or both of the propellant tanks would most likely lead to an explosion that could destroy the launch vehicle. The loadings that can occur on cryogenic propellant tanks are due to the pressure, temperature, g-loads, and sloshing of the cryogenic fluids. These loadings were considered when designing the propellant tanks for safety.

The propellant tanks in each of the two core stages is constructed of 2219-T87 aluminum. The locations of the liquid hydrogen and liquid oxygen tanks are shown in Figure 3.13. The tanks provide the structural support of the core in these regions which reduces the mass by eliminating an outer structure of the core to support the tanks as well as the structure above and other loadings. Longerons and rings are needed in the tank construction to prevent the major failure mode of buckling in the core stages.

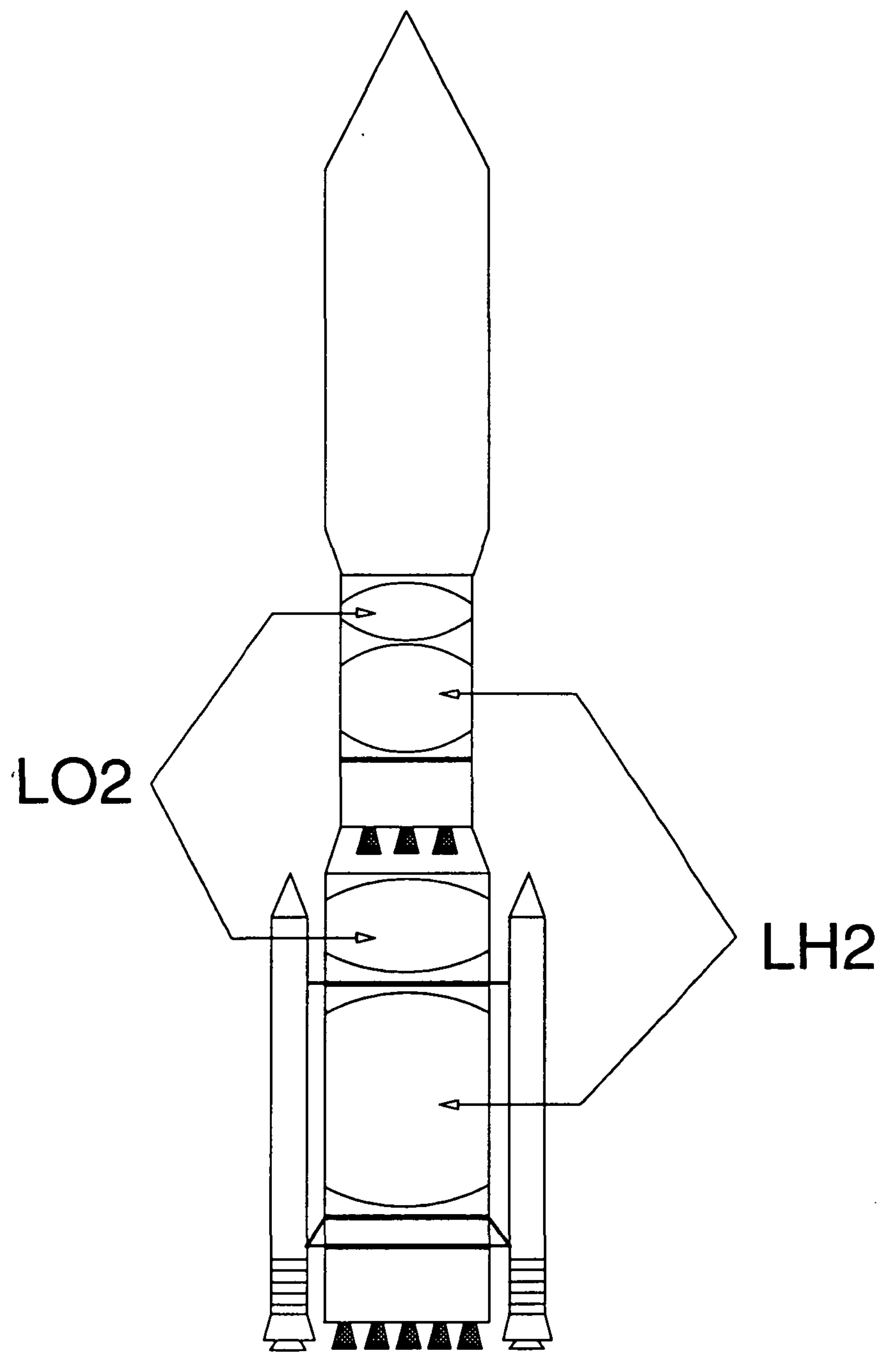


Figure 3.13 Tank Design

Another consideration in the tank design is sloshing of the cryogenic propellants which adds additional loadings to the tanks. The longerons and rings in the tank structure reduce the sloshing. Lightweight slosh baffles are also required to reduce sloshing not stopped by the longerons and rings within the tanks.

Finally, the cryogenic tanks must be insulated to retard boil off. The spray-on insulating foam used on the Space Shuttle external tank was chosen.

Detailed structural analysis will be required to accurately size the tank thickness and supporting longerons and rings.

3.7.9.3 Thrust Transfer Structures

A single design is utilized for the thrust transfer structures to transmit the thrust of the core engines and the SRB's. These structures consist of the thrust support structure that houses the SSME's in the core stages and the booster support rings for connections of the SRB's. Figure 3.14 shows the location of these structures.

The thrust support structure transfers the thrust of each of the SSME's at the base of each core stage to the skin and stiffeners. The structure is designed to transfer the locally applied engine loads to a near uniform compressive load around the circumference of the lower end of the core structure. The thrust structure for both core stages are similar. The major difference is the overall size of the structure for the first and second core stages, which is dependent on the number of SSME's and the loads it must support.

The booster support rings are designed to distribute the thrust of eight SRB's to the core, but also allows for the two, three, four, or six booster configurations. There are two lower connections and one upper connection for each SRB on the core. Figure

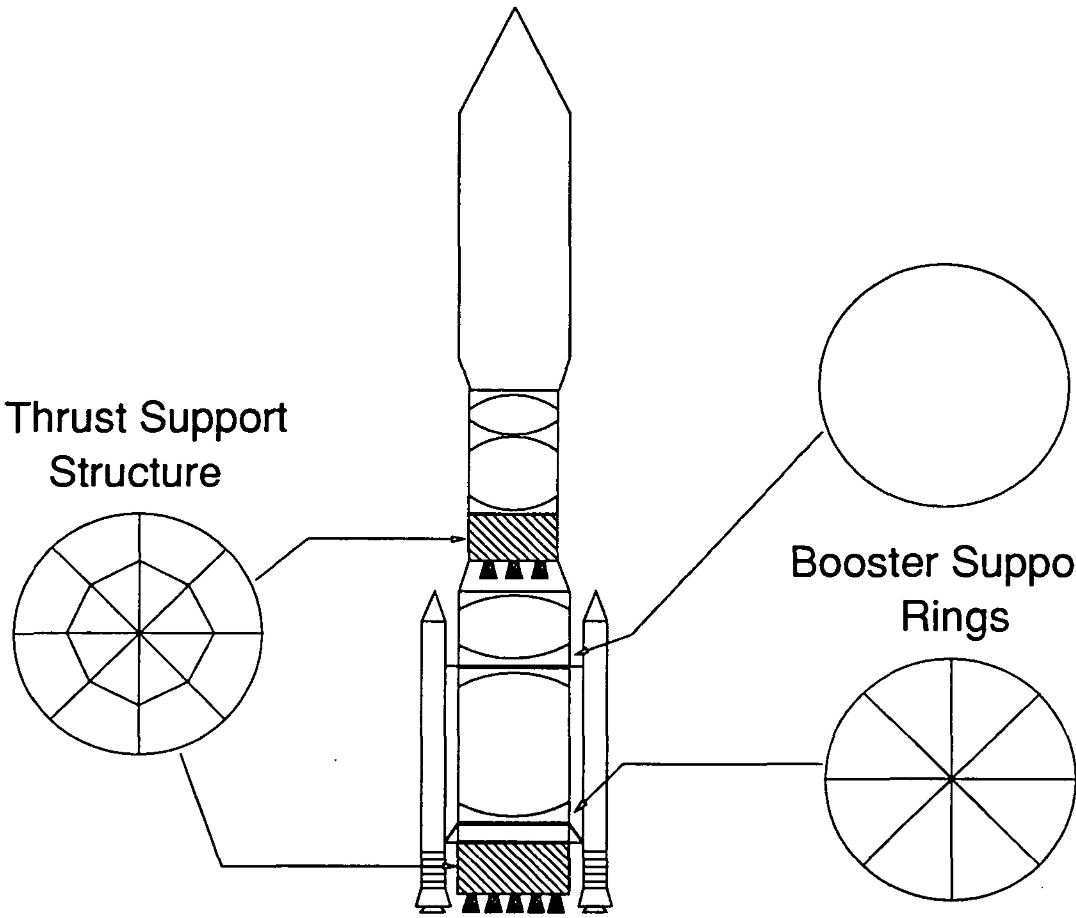


Figure 3.14 Thrust Transfer Structures

3.14 shows the booster support rings. The lower SRB connection consists of attachment to the thrust support structure and the lower booster support ring shown in Figure 3.14. This lower booster support ring is connected to the reinforcement stringers of the core body. These lower attachments are where most of the upward load of the boosters is concentrated due to the angled braces which will be discussed in the next section. The upper connection ring is located between the two propellant tanks and is connected to the reinforcement stringers above and below. This upper connection primarily holds the booster in place rather than transferring the thrust of the SRB's. Therefore, this ring is essentially a reinforcement ring in the core body.

The space between the thrust support structure and the lower booster connection ring houses the subsystem required for the first core stage. A similar space is allowed for in the second core stage even though it does not have a booster connection ring. Separation for recovery purposes occurs above this subsystem package.

There are many approximations made in the thrust support structures. Detailed structural analysis will be required to validate these approximations and to provide a detailed mass estimate.

3.7.9.4 Booster Connections

There are three structures that connect the SRB to the first stage of the core, two at the lower portion of the core and one at the upper portion. The attachment structure is connected at the same rings used in Shuttle flights.

The lower booster attachment consists of two angled brackets as shown in Figure 3.15. The bracket is attached at one point on the SRB and two points on the core. The angled beam of the bracket is connected to the lower booster support ring of the core and the horizontal beam of the bracket is connected to the thrust support structure. This angled bracket transfers most of the thrust load from the boosters to the core. This dual angled bracket assembly for each booster at the lower end is designed to resist torque in the longitudinal axis and the axis through the SRB and the core.

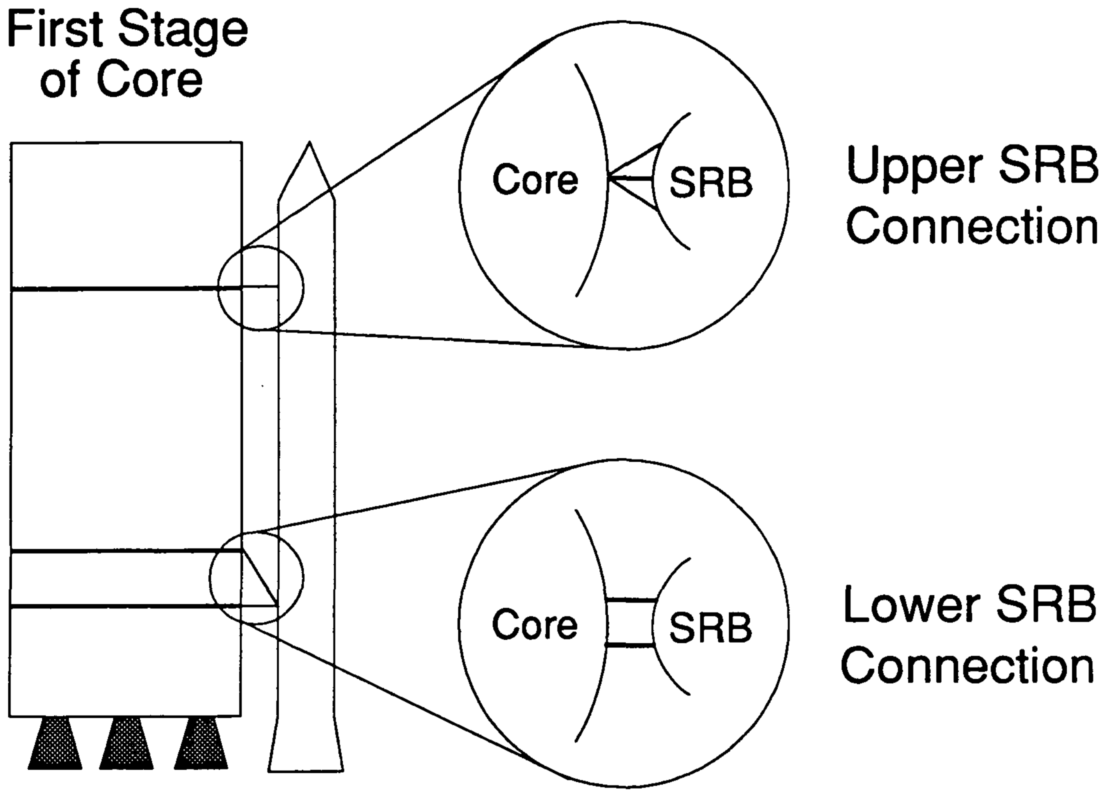


Figure 3.15 Booster Connections

The upper booster attachment is also shown in Figure 3.15. The main purpose of the upper connection is to hold the booster in place. It must also support the aerodynamic loads associated with fight through the atmosphere including wind gusts. This connection resists torque associated with the third axis. The connection structure consists of a beam and two rods on either side to resist torsion.

These booster connection structures describe the means of transferring the thrust from the SRB's to the core. Detailed structural analysis will be required to accurately size the beams and rods for these connections. Also, an aerodynamic study is needed to predict the possible loads on the upper booster connection.

3.7.9.5 Core Structure

This section address the portions of the core not supported by the tank structure. These parts of the core are the areas between the thrust structure and the hydrogen tank, the hydrogen tank and the oxygen tank, and the oxygen tank and the next stage or payload. The main failure mode in these regions is buckling due to the compressive loads. These structures consist of a skin supported by longerons and rings constructed of 2219-T87 aluminum for weldability purposes. The portions of the core where the longerons and rings are needed are shown in Figure 3.16.

To determine skin thickness and the number and size of the longerons and rings, detailed structural analyses will be required for structural integrity of the core.

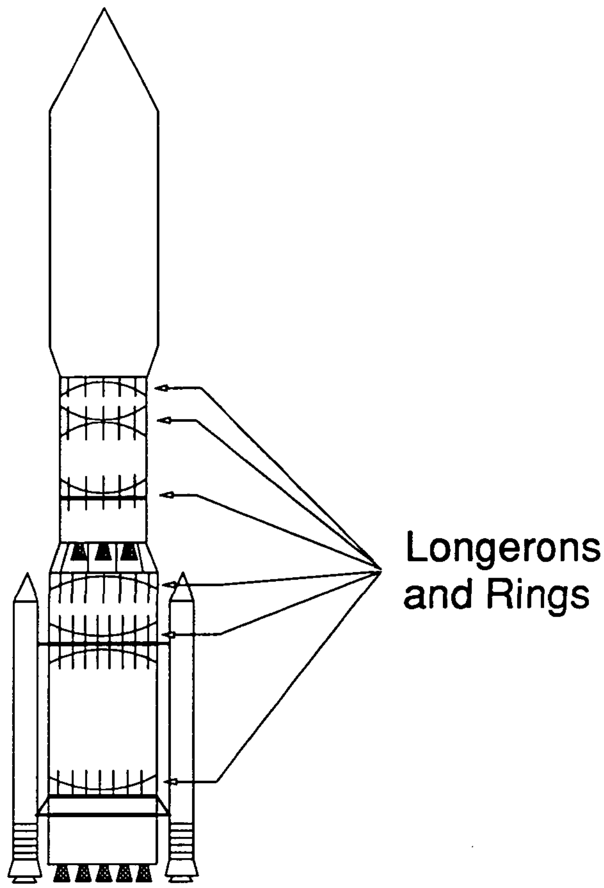


Figure 3.16 Core Structure

3.7.10 Reusable Components

To provide low cost to orbit payloads and to avoid long preparation time for the FHLLV, the design of reusable components was considered. If boosters, core stages, and engines can be returned to earth with minimal damage, and if these parts can be readied for use again in a small time frame, then reusable components may be much cheaper to use for successive missions. The determining factors for deciding this was as follows. Can the old components be cleaned and repaired for less money than the manufacture of new ones? Can they be prepared in sufficient time for use in the next mission? Will they be as reliable as new parts after their initial use? Will reusable components have a long mission lifespan? These questions lead to the following recovery options.

1. No Recovery
2. Booster Recovery - Expendable Core
3. Booster Recovery - Full Core Recovery
4. Booster Recovery - Core Engine Recovery

An elementary cost analysis showed that reusable vehicles were more cost effective to achieve low cost to orbit. The main reason is that six to eleven SSME's, very expensive items, are used on the three configurations. The disposal of these engines with each mission would create prohibitive recurring costs.

The reliability and mission lifespan of the Space Shuttle main engines are well documented from Space Shuttle missions. One engine has flown at least 14 flights which logged 11,274 seconds of hot-fire time. The main consideration is the recovery techniques to prevent damage to the engines.

The ascent trajectory takes the vehicle over the Atlantic Ocean to take advantage of the Earths rotation. This requires that the first core stage and boosters be recovered in the ocean. The SRB recovery will be completed in the same manner as they are recovered after each Space Shuttle mission. The wet recovery can be very destructive to the engines because of the possibility of saltwater in the complicated machinery and high impact speeds. Therefore the recovery must ensure that the engines do not become submerged and that it occurs at low impact speeds if they are to be easily reused. Full core recovery was eliminated because of the difficulty in ensuring low impact speeds and flotation. The engine package can be recovered by deploying parachutes and impact bags to reduce the impact velocity. Flotation bags will be used to keep the engines from becoming submerged.

Dry recovery at lower impact speeds would reduce damage and refurbishment time. Several options were studied but required further development and would not meet the five year development plan. These options are discussed in section 3.7.11.

Since the avionics package is located adjacent to the engine module, these components will also be recovered. This package is referred to as the wet engine and avionics recovery module or WEARM. Figure 3.17 shows the WEARM and the associated flotation and impact bags.

Figure 3.18 describes the overall mission scenario and the recovery methods for the boosters and both core stage WEARM's. The following sections describe the recovery methods for the SRB's, and first and second stage WEARM's.

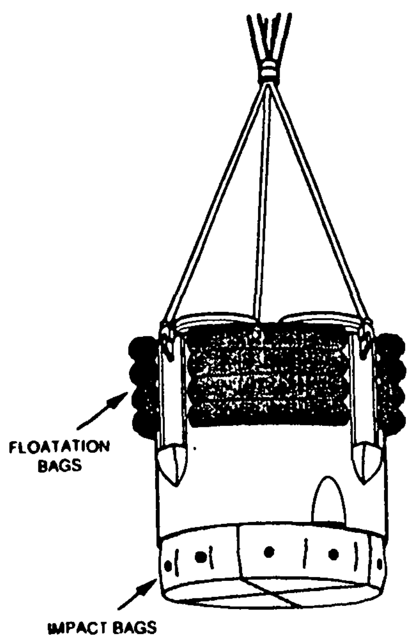


Figure 3.17 WEARM Recovery Detail

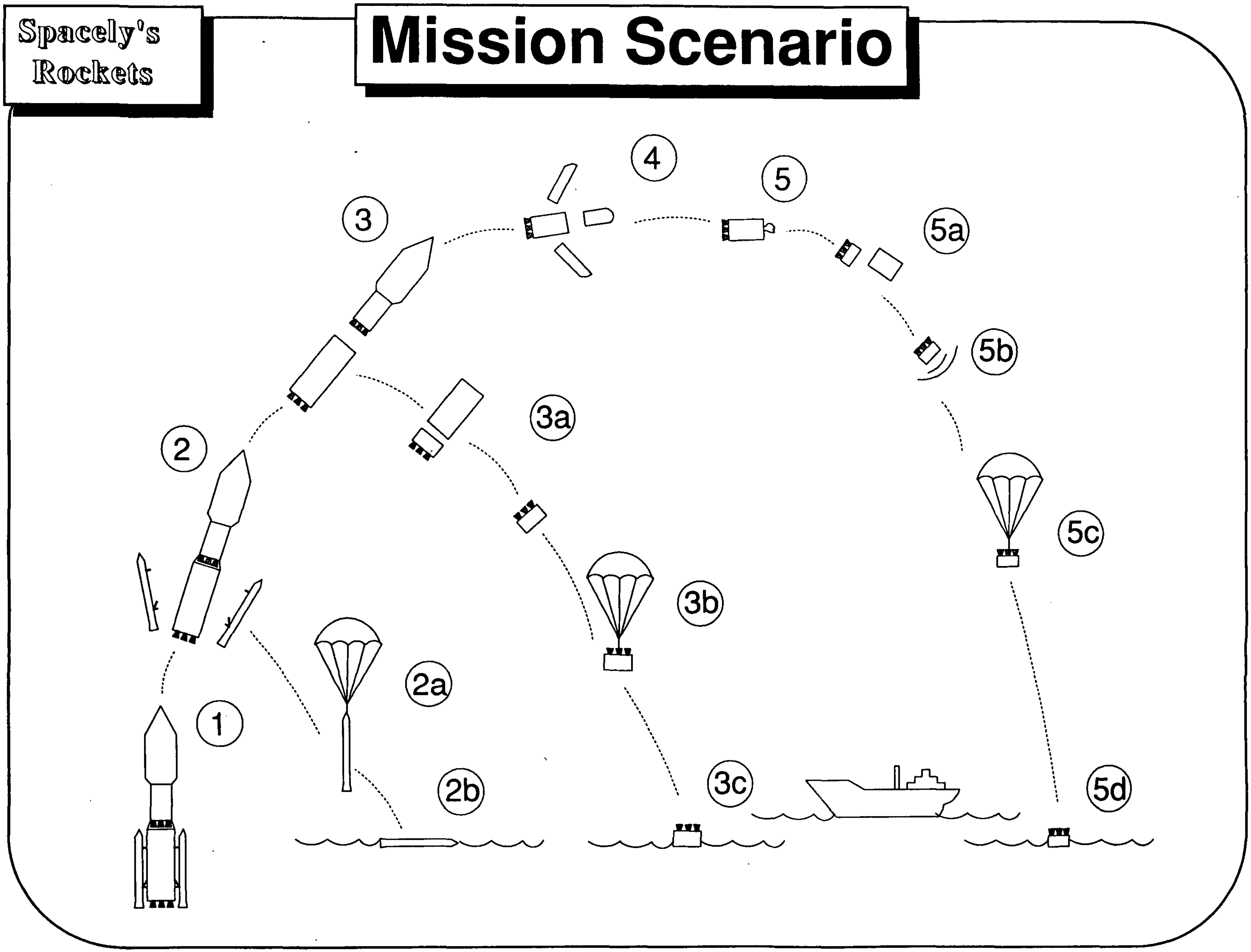


Figure 3.18 Mission Scenario

3.7.10.1 SRB Recovery

The methods for recovering the SRB's after each Shuttle mission has proven to be efficient and economical. Therefore, these methods will be utilized for the family of heavy lift launch vehicle. Points 2, 2a, and 2b in Figure 3.18 show the recovery method for the boosters. Figure 3.19 describes the SRB recovery in further detail.

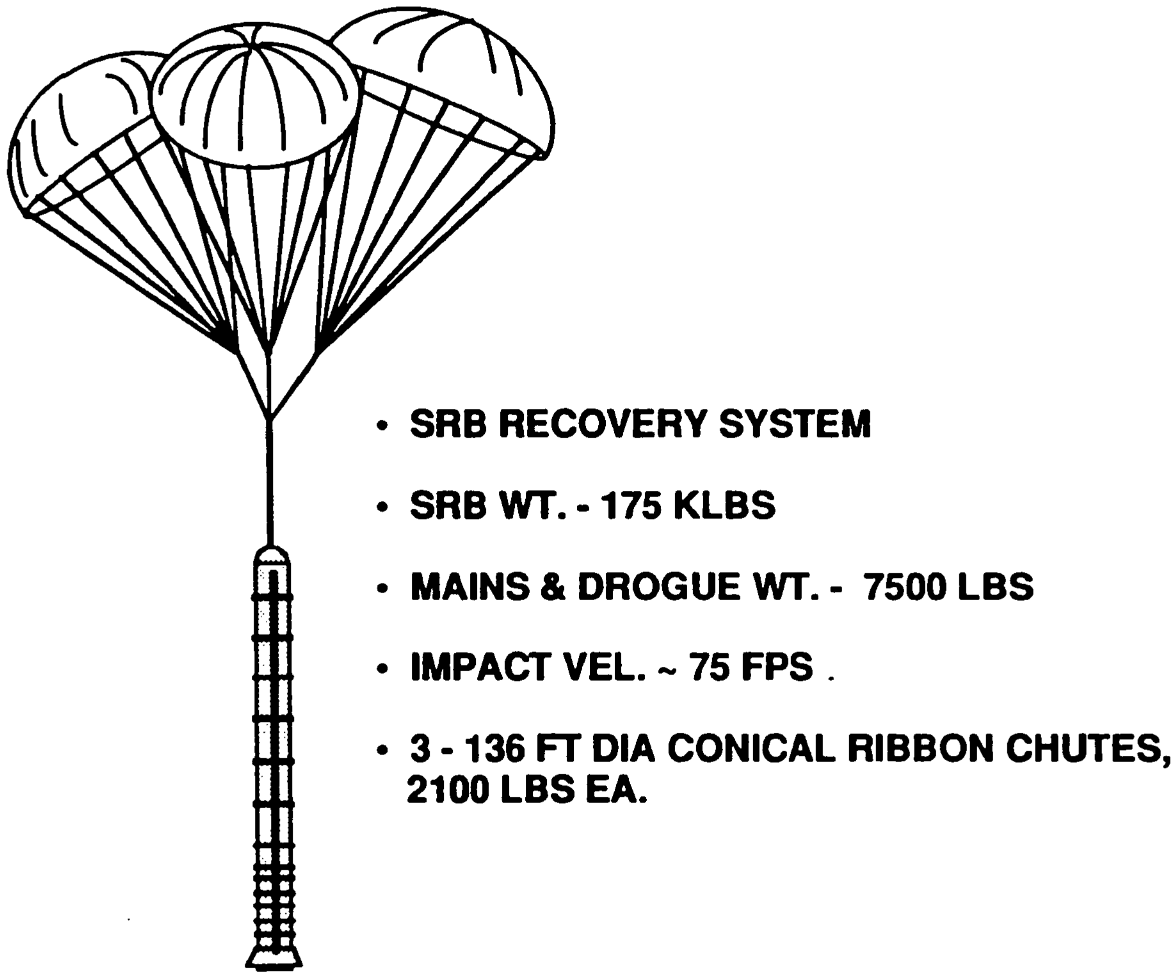


Figure 3.19 SRB Recovery

3.7.10.2 First Stage Recovery

The first stage WEARM recovery is depicted by points 3, and 3a - 3c in the mission scenario shown in Figure 3.18. Figure 3.20 shows the steps in the recovery of the first stage WEARM just prior to splash down. The components of the first stage WEARM recovery are listed below.

1. Separation of the first core stage
2. Separation of the WEARM
3. Attitude adjustment
4. Velocity reduction due to drag
5. Parachute deployed for deceleration
6. Splash down
7. Refurbishment

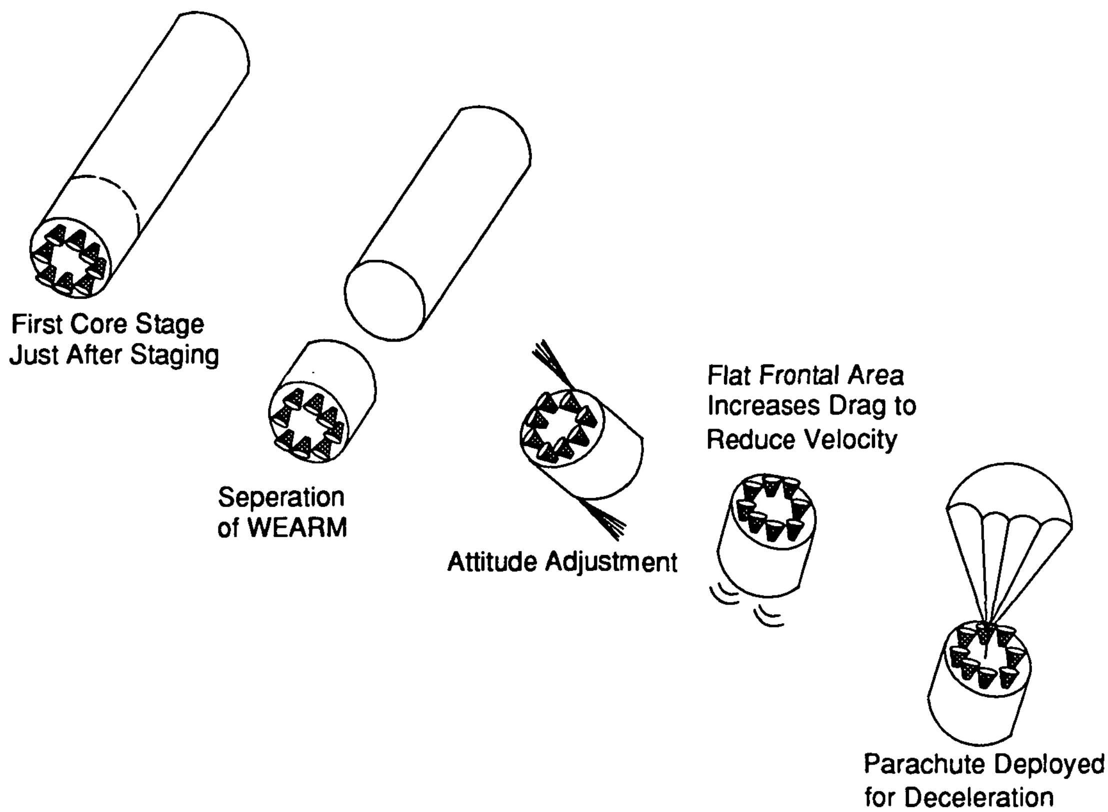


Figure 3.20 Detailed First Stage WEARM Recovery

3.7.10.3 Second Stage Recovery

The second stage WEARM recovery is depicted by points 4, 5 and 5a-d in the mission scenario shown in Figure 3.18. The components of the second stage WEARM recovery are listed below and shown in detail in Figure 3.21

1. Payload deployment
2. OMS firing to deorbit and reorient stage
3. Stage decelerates in atmosphere
4. WEARM separation

5. Parachute deployed for deceleration
6. Splash down
7. Refurbishment

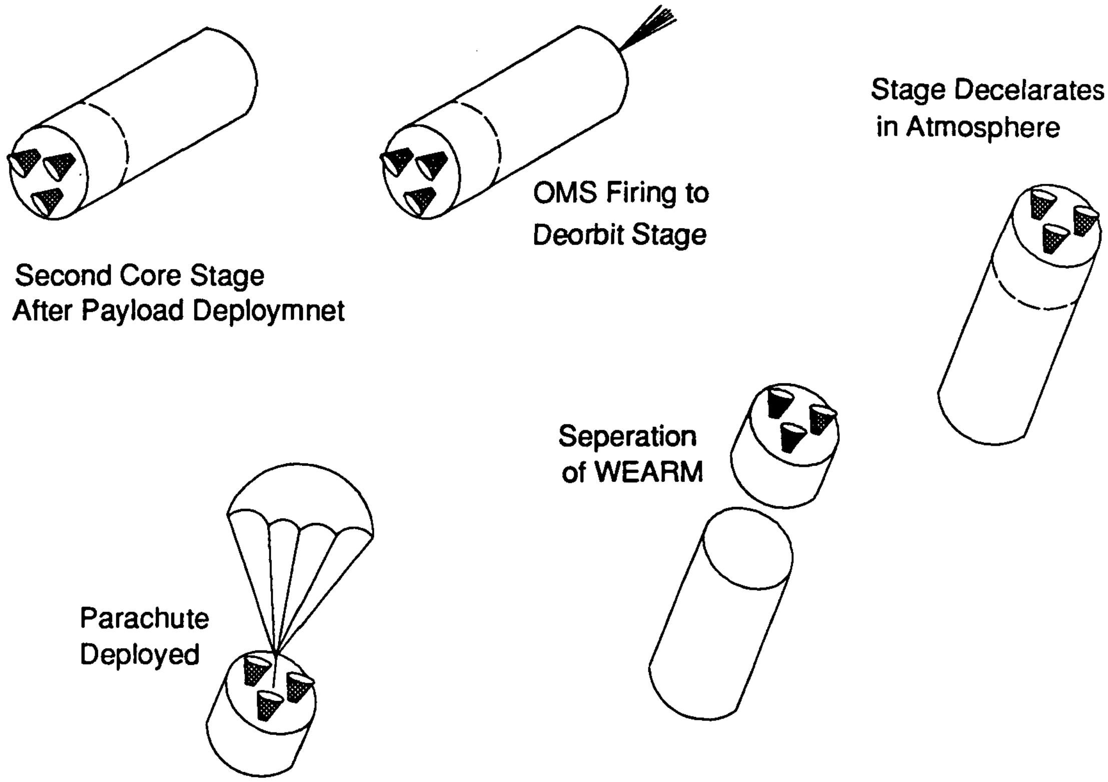


Figure 3.21 Detailed Second Stage WEARM Recovery

3.7.11 Suggested Future Studies

Research needs to be completed by engineering and chemical companies to develop non-polluting, high thrust solids, hybrids, and liquid engines which utilize a more efficient combustion process. If great advances are made in the latter, then liquid boosters could be sized much more closely to solid boosters, while having better safety and controllability.

On the other hand, if solid propellants are developed which have little or no environmental impact and improved safety conditions, then the desire to use all liquids for large vehicles like the FHLLV may disappear.

The design requires a detailed structural analysis to verify the assumptions made in the structural components. These analyses can also be used to reduce weight. Further weight reduction can be obtained through the use of advanced materials.

Several options were considered for dry recovery of the WEARM to prevent water damage and reduce impact damage. One option proposed by Hugh Davis of Davis Aerospace would be to catch the engine and avionics module aboard a large barge. Figure 3.22 shows this approach. Another approach would be to utilize flyback modules that could be guided to a landing point. However, current technology for these recovery techniques is not available to meet the five development plan for the FHLLV Phase I design.

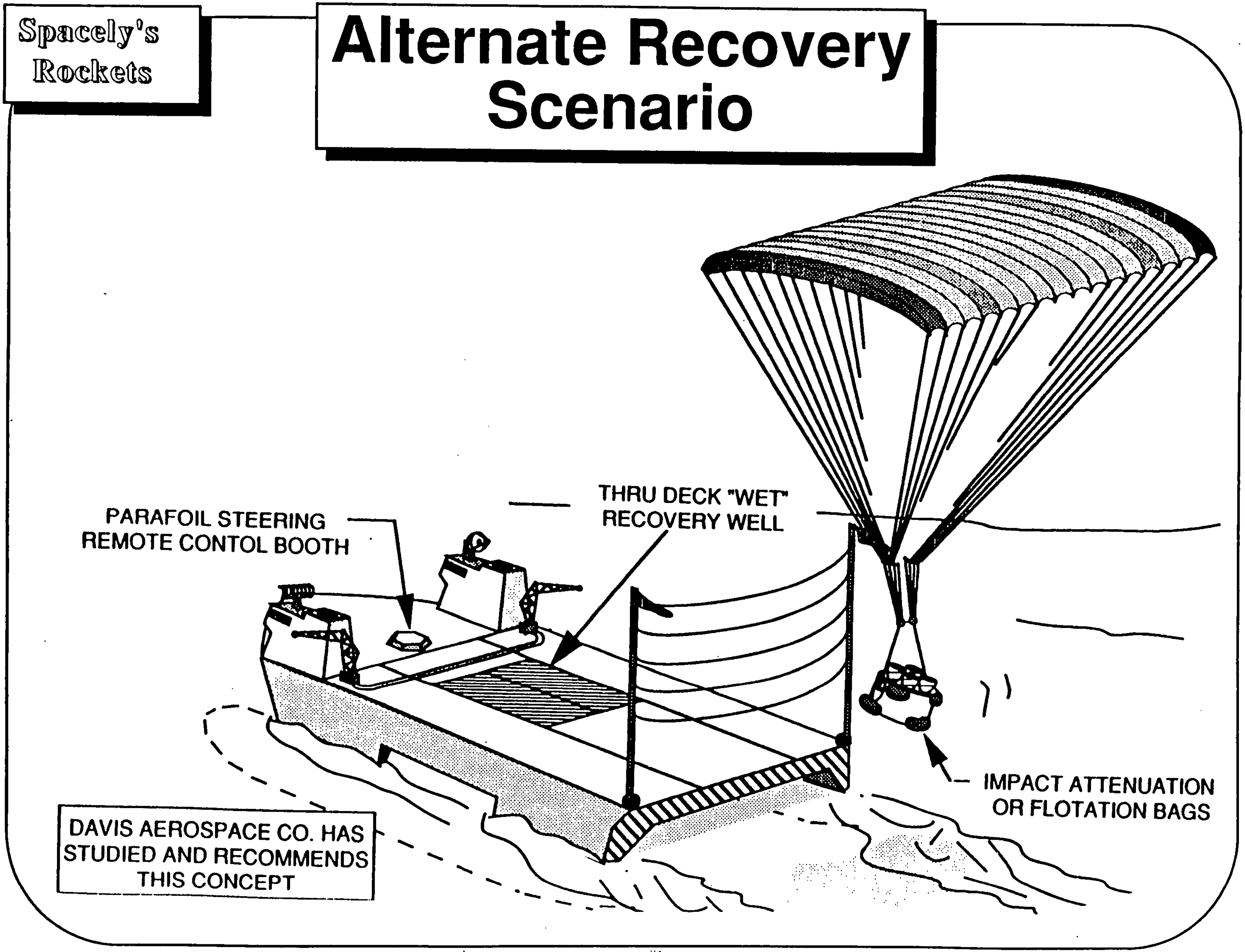


Figure 3.22 Alternate Recovery Method

4 Management

The design team of Spacely's Rockets is headed by a Project Manager whose primary responsibilities are to tend to administrative needs, and to ensure the timely completion of the project. Furthermore, the Project Manager is an interface between the design team and the contractor. Technical assistance to the manager is provided by a Chief Engineer. The Chief Engineer is also responsible for coordinating the efforts of the two company divisions, and maintaining the technical integrity of the project.

For purposes of this project, Spacely's Rockets was organized into two divisions; the PLS Division and the Launch Vehicle Division. Each of these was lead by a Senior Engineer. For a graphical description of the company organization, please refer to Figure 4.1.

4.1 The PLS Division

The PLS Division was directly responsible for the conceptual design of the Personnel Launch System and its requisite subsystems. This division was further subdivided into three departments, each with specific task assignments.

The Trajectory Group studied the possible trajectories for the PLS vehicle, based upon different mission scenarios. These trajectories define the performance that the vehicle must achieve to be a viable transportation system.

The Body Group was required to design the PLS vehicle with the exception of its subsystems. The structure, shape and space-worthiness of the PLS system were the primary concerns of this group.

The Subsystems Group determined the necessary subsystems required by the PLS vehicle, and their integration into the overall design.

Each of the above groups was headed by a designated staff engineer that performs the tasks of a Group Manager. Each of the Group Managers, their technicians were responsible for completing their group's assigned tasks. The Group Managers were supervised by the PLS Division's Senior Engineer.

4.2 The Launch Vehicle Division

The Launch Vehicle Division had the responsibility of designing a family of unmanned launch vehicles which include a heavy-lift capacity. In order to manage the design, the Launch Vehicle Division was further divided into three departments: Trajectory Design, Propulsion and Vehicle Design.

The Trajectory Design Group defined the critical parameters in the ascent trajectories from launch to orbit. This group also tested the final design, to ensure that it satisfies the mission objectives.

The Propulsion Group had the sole task of researching and designing the propulsion systems for the launch vehicles. This group relied on input from the Trajectory Design Group to derive the most reliable and efficient propulsion system.

The Vehicle Design Group integrated the propulsion system, payload and required subsystems into an overall vehicle design. The group also performed a weight and sizing analysis on prospective designs.

Each of the above groups was headed by a designated staff engineer that performs the tasks of a Group Manager. Each of the Group Managers, their staff of engineers and technicians were responsible for completing their group's assigned tasks. The Group Managers were supervised by the Launch Vehicle Division's Senior Engineer.

4.3 Project Integration

To provide an avenue of information exchange between the PLS and Launch Vehicle Divisions, a special Systems Integration Team was created. This team consisted of two staff engineers, one from each division. Leadership of the team was provided by the Chief Engineer.

4.4 Design Strategy

Figures 4.2 and 4.3 are flowcharts of the activities which led to the final conceptual designs of the PLS and Launch Vehicle systems respectively.

4.5 Task Schedules

Figure 4.4 was the task schedule used for achieving the final design of the PLS system. Similarly, Figure 4.5 was the task schedule for the family of launch vehicles.

4.6 Completed Tasks

As of the time of this report, all tasks indicated on the task schedules for both the PLS and FHLLV projects have been completed. This report reflects the final design achieved after accomplishing the studies shown in the flowcharts.

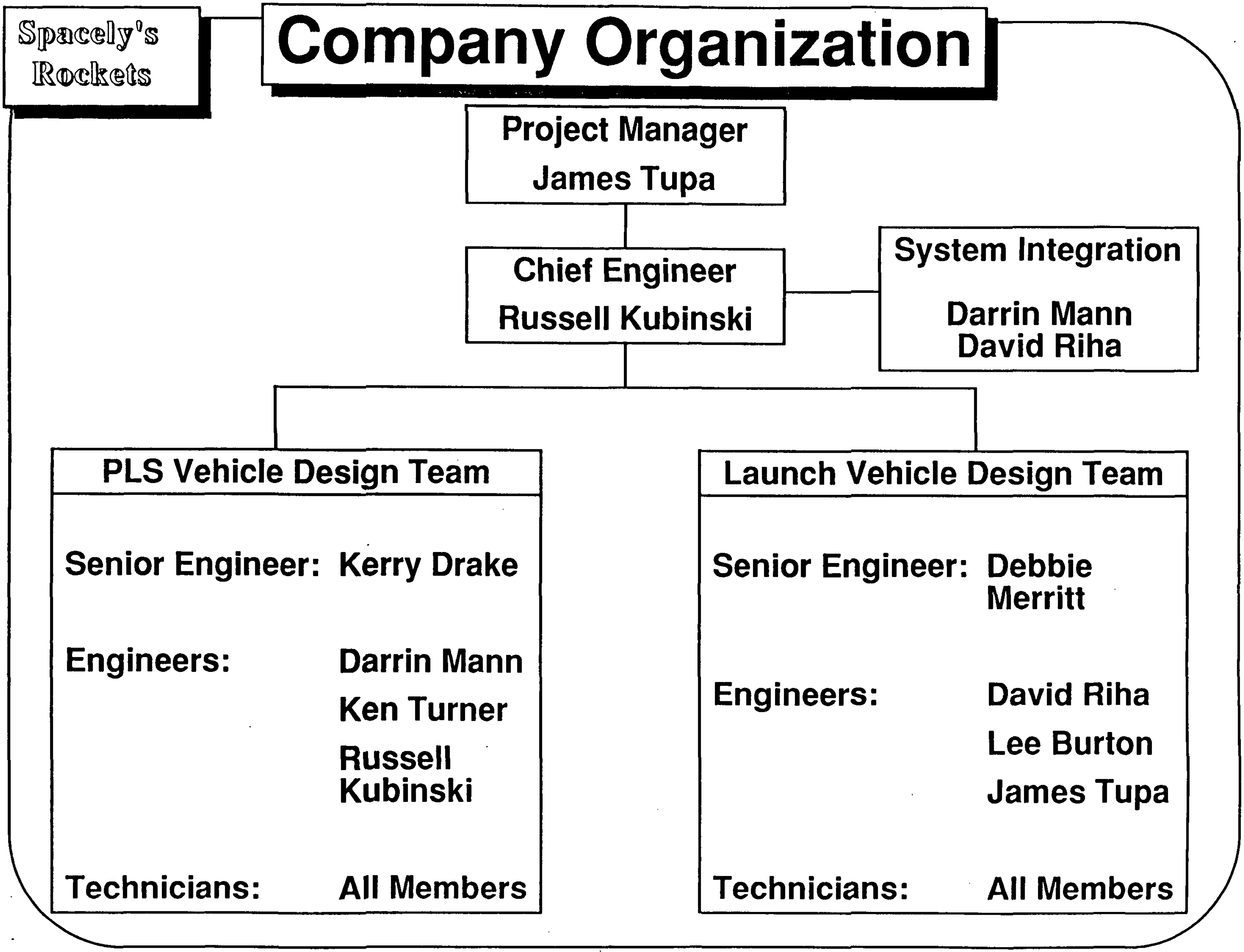


Figure 4.1 Company Organization

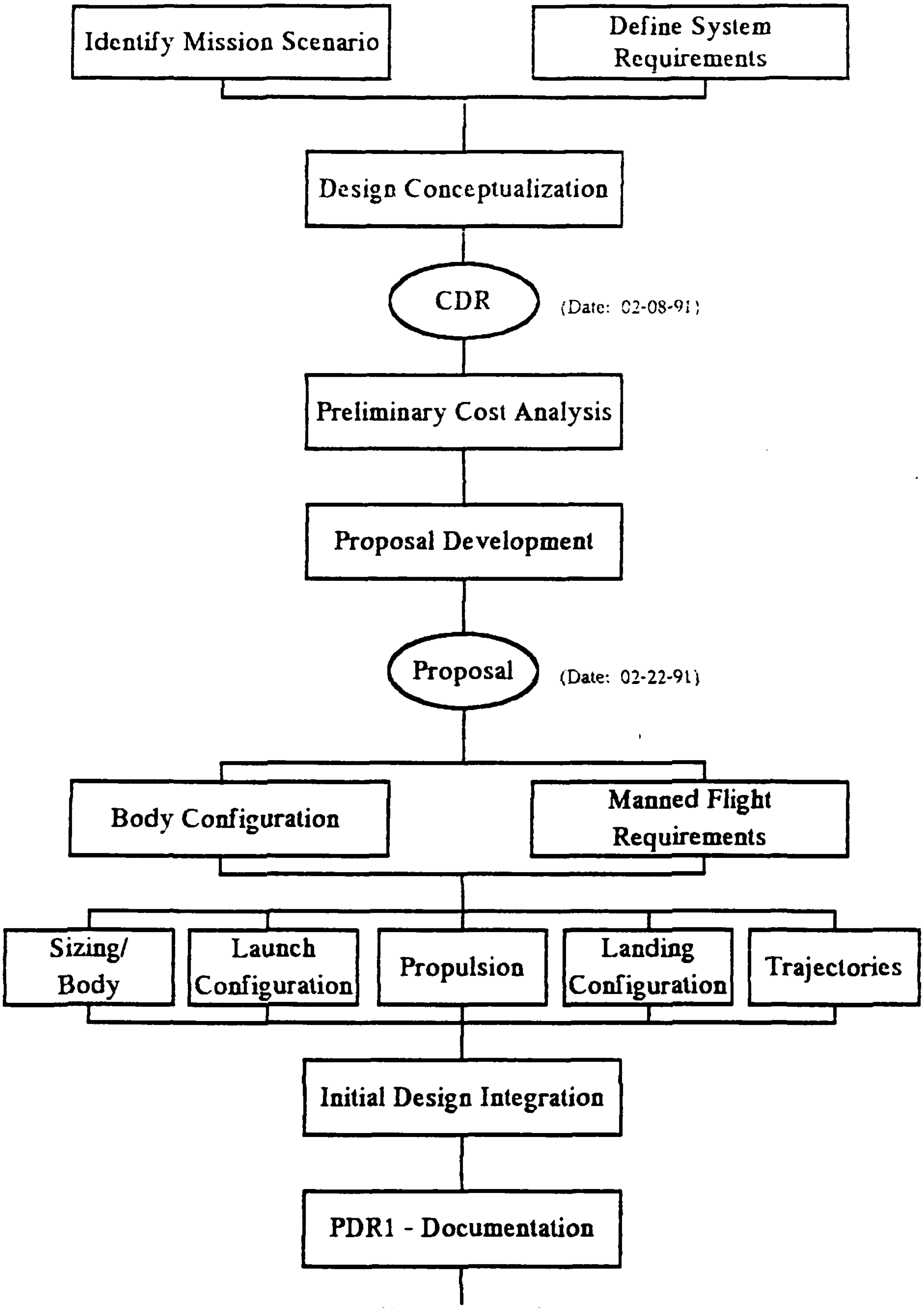


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Figure 4.2. Design Strategy for the PLS Division.

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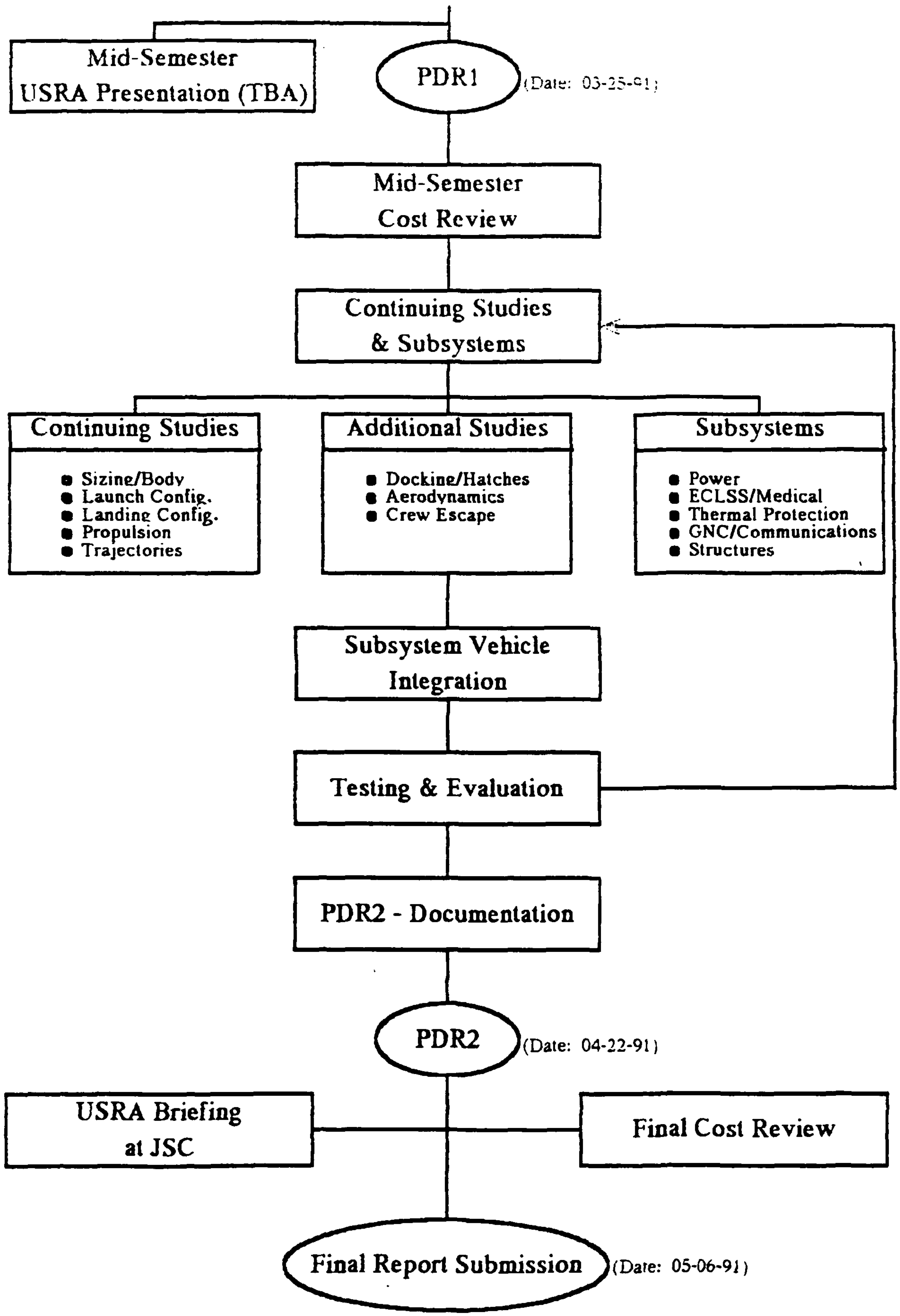


Figure 4.2 (cont.). Design Strategy for the PLS Division.

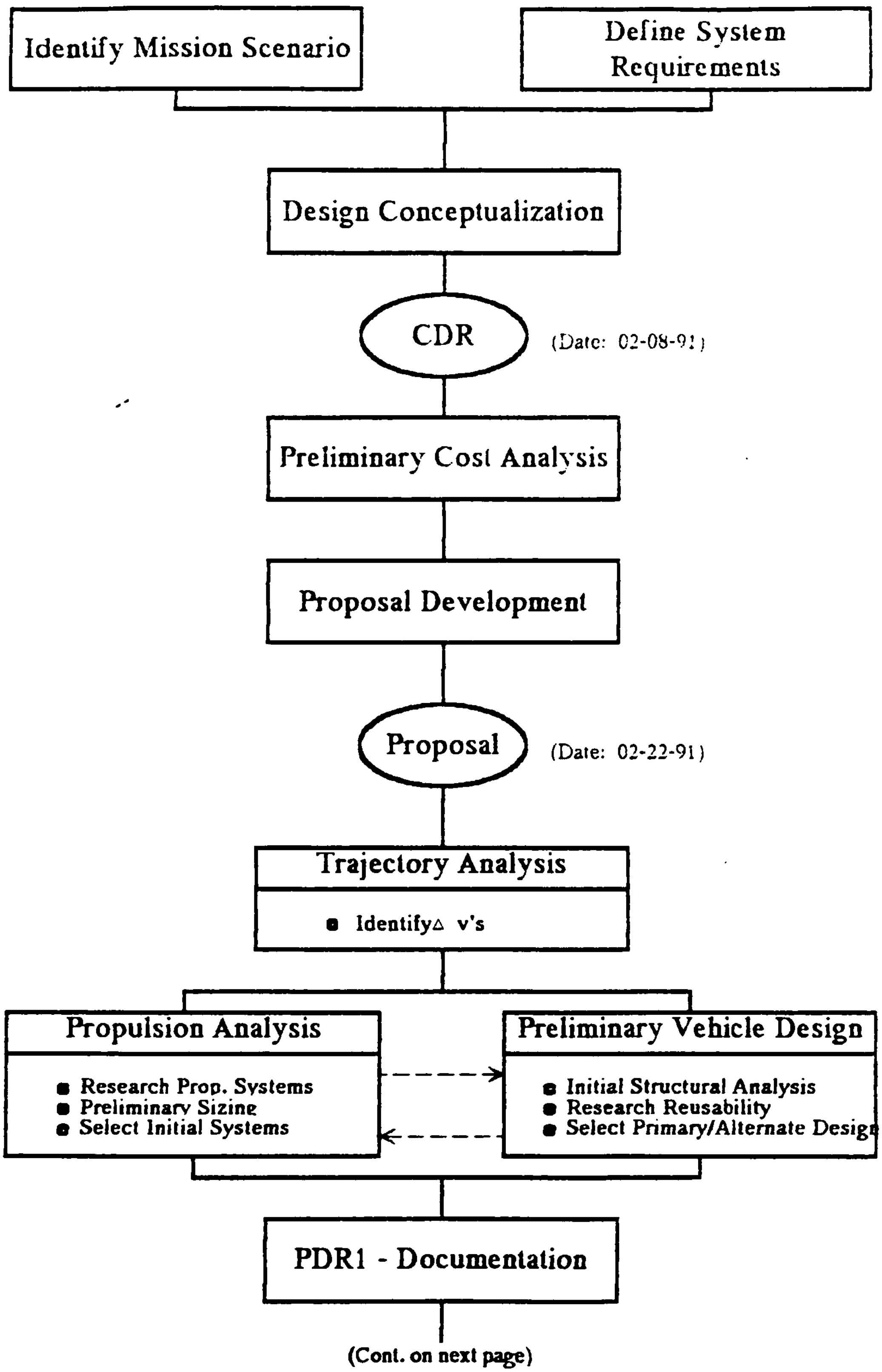


Figure 4.3. Design Strategy for the Launch Vehicle Division.

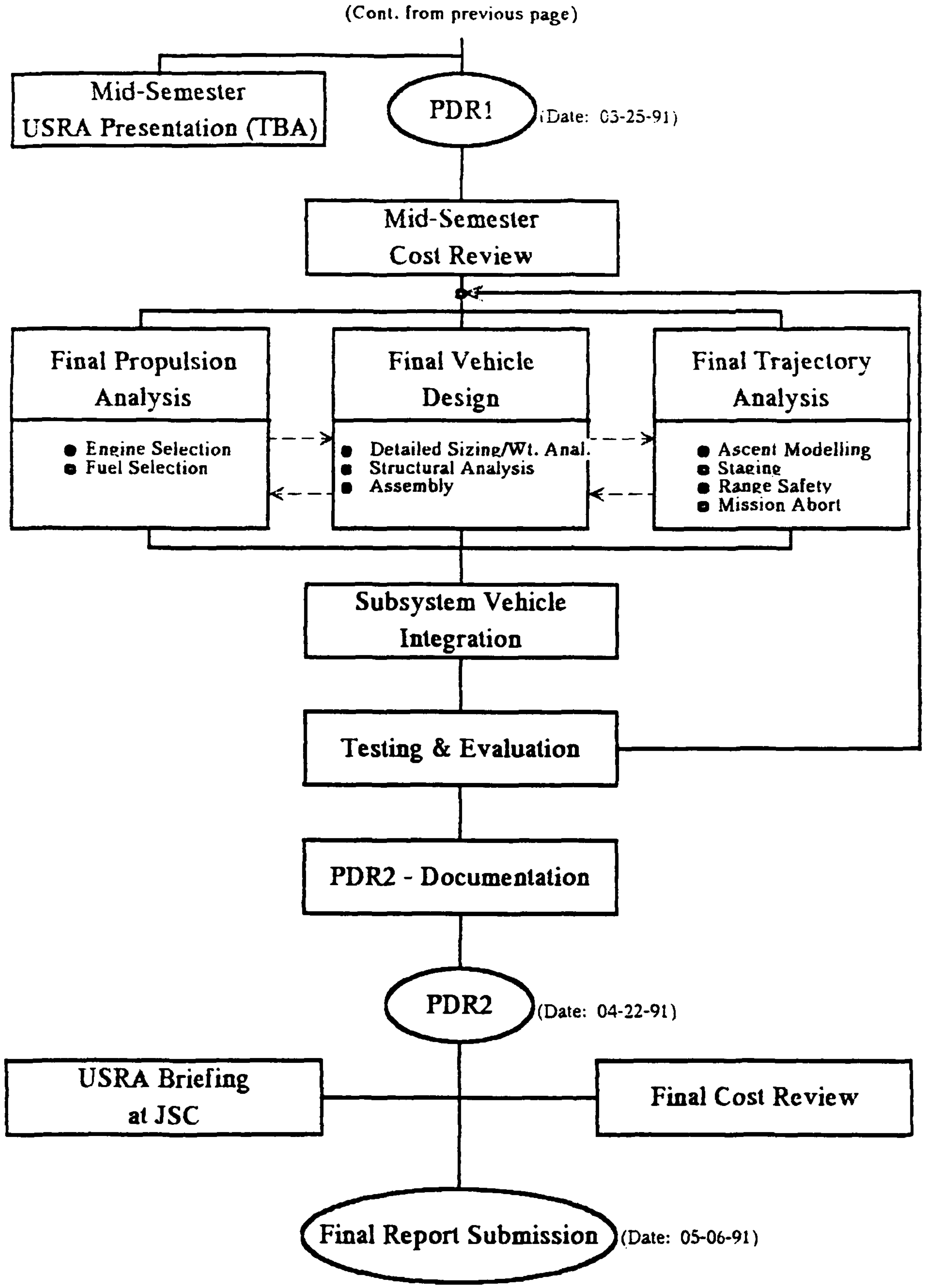


Figure 4.3 (cont.). Design Strategy for the Launch Vehicle Division.

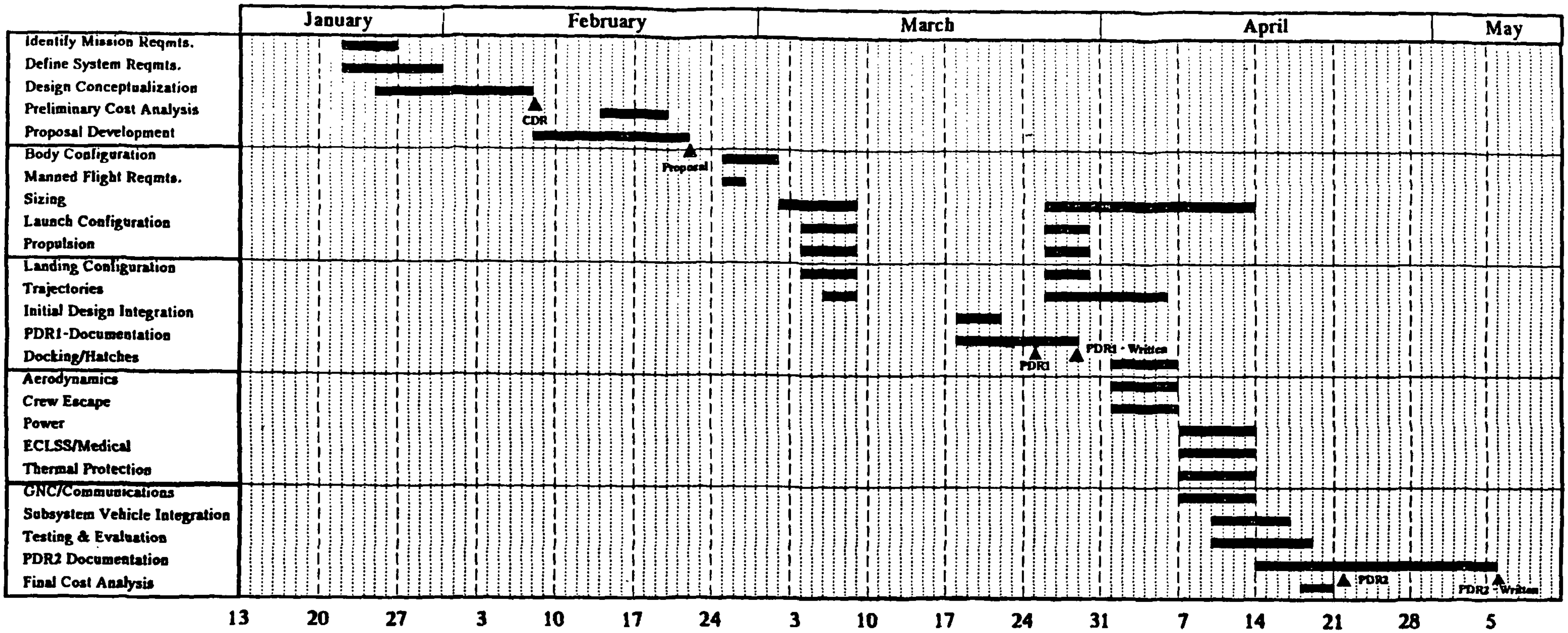
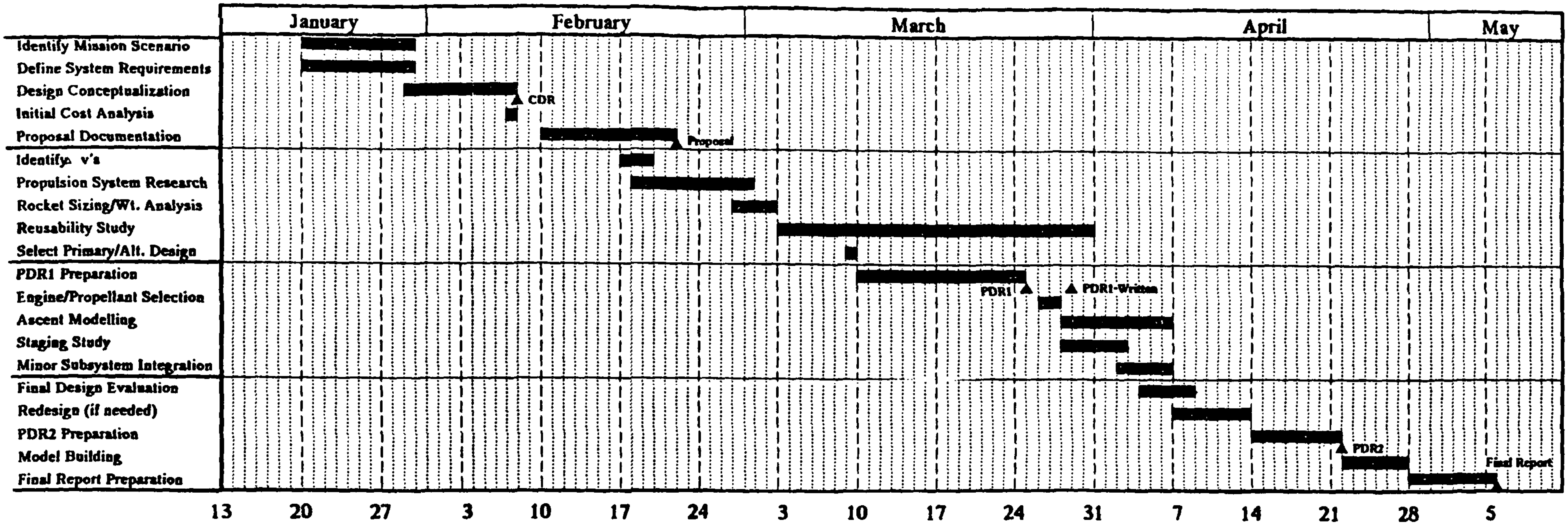


Figure 4.4. Task Schedule for the PLS Division.



ure 4.5. Task Schedule for the Launch Vehicle Division.

5 Cost Summary

5.1 Personnel Cost Estimate

Based on the pay scales listed in the Request for Proposal the following personnel cost estimate was prepared.

Pay Scales:

Project Manager	$ 25/hr
Chief Engineer	$ 22/hr
Senior Engineer	$ 20/hr
Staff Engineer	$ 15/hr
Technician	$ 10/hr
Consultant	$ 75/hr

5.2 Estimated Economic Performance

The following charts are the estimated cost analysis that was presented in the proposal. These results will be compared to the actual cost schedule to evaluate the true economic performance of the team.

Estimated Weekly Personnel Costs:

1	Project Manager	x 8hrs	200.00
1	Chief Engineer	x 8hrs	176.00
2	Senior Engineers	x 8hrs	320.00
8	Staff Engineers	x 12hrs	1440.00
	Consultants	x 8hrs	600.00
		Total Weekly Costs	$ 2736.00

Additional Costs for Peak Weeks:

1 Project Manager	x 4hrs	100.00
1 Chief Engineer	x 4hrs	88.00
2 Senior Engineers	x 4hrs	160.00
4 Staff Engineers	x 4hrs	240.00
8 Technicians	x 4hrs	320.00
	Total Peak Week Costs	$ 908.00

Total Estimated Personnel Costs:

Total Weekly Costs x 10	27360.00
Total Peak Week Costs x 2	1816.00
Subtotal	29176.00
plus 10% error est.	2917.60
Total Estimated Personnel Cost	$ 32093.60

5.3 Estimated Material and Hardware Cost Estimate

The material and hardware cost estimates were based on the expenses of previous design groups.

Material and Hardware Costs:

4 month rental of IBM PCs	4000.00
VAX/VMS mainframe time	200.00
photocopies @ $.10/each	60.00
transparencies @ $.60/each	120.00
models	240.00
miscellaneous supplies	40.00
Subtotal	4660.00
plus 10% error est.	466.00

Total Proposed Material and Hardware Costs	$ 5126.00

Total Cost Estimates:

Total Proposed Material and Hardware Costs	$ 5126.00
Total Proposed Personnel Cost	32093.60
Total Proposed Material and Hardware Cost	5126.00
Estimated Total Cost of Design	$ 37219.60

5.4 Actual Economic Performance

The following is a summary of the actual costs to complete the project.

Estimated Total Cost of Design	$ 37219.60
Total Personnel Cost	36793.75
Total Material Cost	6541.65
Total Project Cost	43335.50

Figure 5.1 is a graph that tracks the actual personnel cumulative cost per week versus the estimated weekly cost. This is an approximate reflection of the weekly productivity of the design team at Spacely's Rockets.

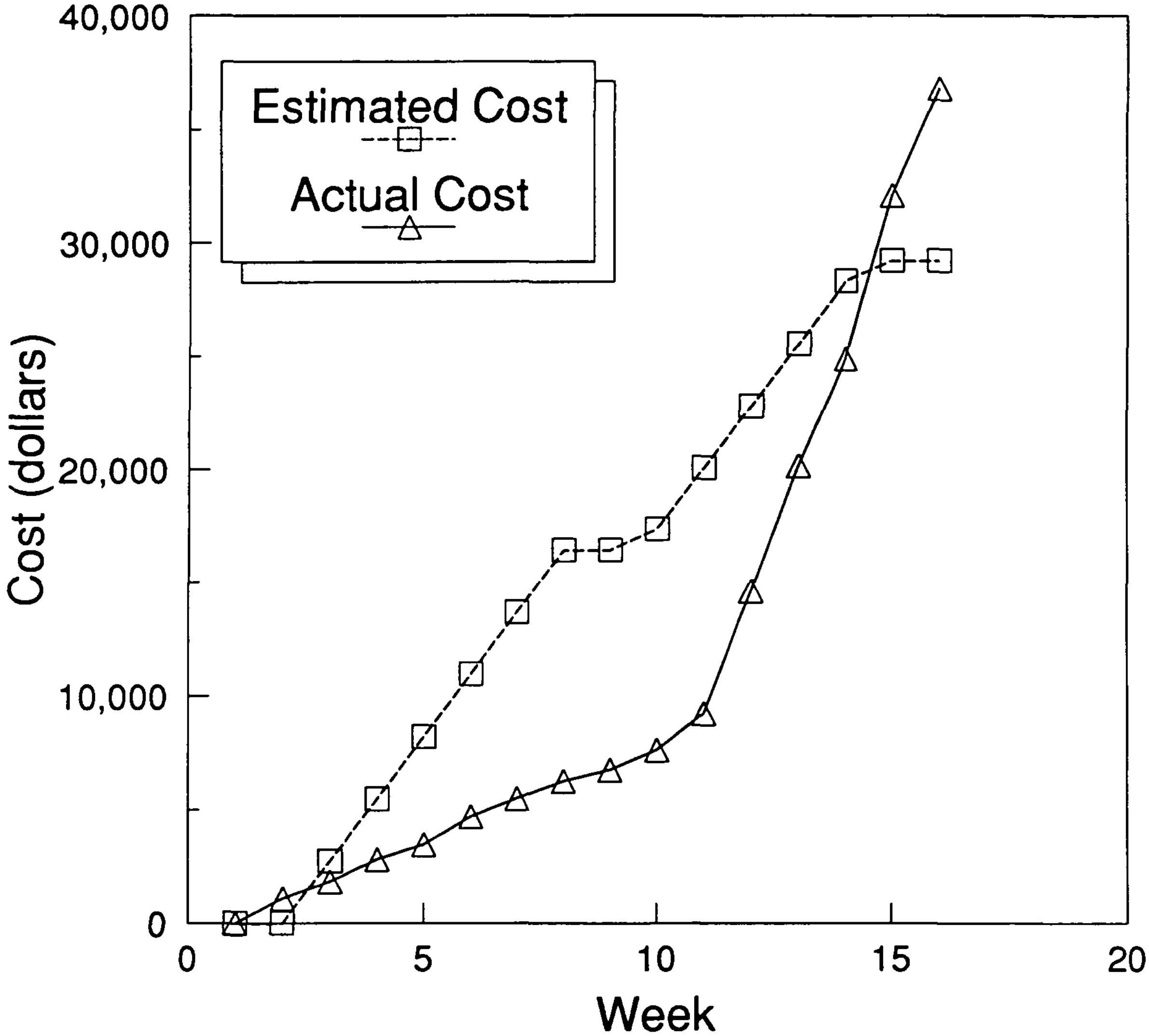


Figure 5.1 Cumulative Cost Comparison

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