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ADVANCED MISSION SYSTEMS CONCEPTS IN SUPPORT OF SPACE EXPLORATION: PHOENIX - A LOW-COST COMMERCIAL APPROACH TO THE CREW EXPLORATION VEHICLE

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ABSTRACT: Since the announcement of President Bush's Vision for Space Exploration (VSE) in early 2004, the architecture of Project Constellation has been selected. The system will be centered around the Orion Crew Exploration Vehicle (CEV), which has been dubbed by NASA administrator Michael Griffin as "Apollo on steroids". The CEV is to be launched on a new launch vehicle, derived from existing shuttle technology. The development of this new spacecraft and launch vehicle is a very costly proposition. An alternate approach is proposed in this study. The Phoenix is a smaller spacecraft designed specifically to be launched on the Falcon 5 vehicle under development by SpaceX. Because the SpaceX vehicle will cost only a fraction of today's launch costs, the Phoenix is estimated to cost 27% of the price of the CEV. This reusable three-person capsule utilizes the innovative ParaShield re-entry concept, which allows for a cylindrical spacecraft with greater interior volume. This extremely cost-effective spacecraft has a Technology Readiness Level (TRL) of 6 and is an attractive option for fulfilling VSE requirements.

1. INTRODUCTION

With the publication of NASA's Exploration Systems Architecture Study report [1], the preferred vehicle systems for the renewed human exploration of the moon and Mars have become clear. The centerpiece of the Constellation transportation architecture will be the Orion Crew Exploration Vehicle (CEV); an Apollo-style capsule which will carry humans into Earth orbit, to the moon, and beyond. Yet, a closer examination of ESAS shows that the CEV concept is a large, complex, and costly spacecraft, and that Vision for Space Exploration (VSE) [2] costs are dominated by the decision to build new launch vehicles based on shuttle components.

As a lower-cost alternative, this study investigates the feasibility of a smaller, more economical spacecraft designed to fulfill the CEV role. To further the low-cost aspect, the focus of this study is to design a human-carrying spacecraft named Phoenix, which can be launched on the Falcon 5 vehicle under development by SpaceX. This launch vehicle is currently being marketed at approximately 25% of the cost of equivalent traditional launch vehicles, and has been designed from the outset to eventually be human-rated for commercial space transportation. By focusing the design process on a smaller spacecraft launched by a low-cost next-generation launch vehicle, this report investigates options for future space exploration at a reduced cost.

Design details for the Phoenix are presented on a system and subsystem level, along with production scheduling and cost estimations. Results of this design effort document the feasibility of this approach to the next generation of human spacecraft, and its applicability to an inexpensive vision for human exploration of the moon and beyond.

2. MISSION OBJECTIVES AND OVERVIEW

The design process started with the establishment of three reference missions for the spacecraft. The first, a "solo" mission, requires the spacecraft to be injected into a circular orbit ranging from 200 to 600 km altitude at 28.5° inclination with full crew. The mission duration is seven days, plus two contingency days. This category of mission was chosen to be representative of critical missions currently performed by the space shuttle, such as Hubble Space Telescope servicing. The second design reference mission is an International Space Station (ISS) crew rotation. For this mission, the spacecraft must achieve a 400 km circular orbit with an inclination of 51.2°. The transfer time to and from the station is 5 days plus 2 contingency days. In addition to carrying crew and high-priority cargo to the ISS, the spacecraft has to be designed to interface with ISS utilities and remain docked for up to a year before returning to Earth. In this mode, the Phoenix would be a direct replacement for the current use of Soyuz vehicles, except with a longer stay time. The third mission is a lunar flight, in direct support of the VSE. After docking with the transfer and landing systems, launched on an unmanned heavy-lift vehicle, the Phoenix will support the crew to and from the moon. The total transfer time is 10 days plus 2 contingency days. The spacecraft must survive unmanned for 14 days in lunar orbit awaiting the return of the crew from the surface.

Additionally, design requirements call for a 20% mass margin. A 15% power margin and a 30% cost margin are also included. The vehicle must be able to carry at least three astronauts. It must be capable of docking with another space habitat and crew transfer in a shirt sleeve environment. The vehicle must also be able to support extra-vehicular activity (EVA) in an undocked configuration.

3. LAUNCH VEHICLE

The Space Exploration Company, SpaceX, is a privately funded company currently developing the Falcon family of launch vehicles in southern California. The planned Falcon family includes the Falcon 1, which can carry 570 kg to low earth orbit (LEO) for $6.7M, the Falcon 5, which can carry 4,100 kg to LEO for $18M, and the Falcon 9, which has variants that can carry 9,300 kg to 24,750 kg to LEO for $27M to $78M. These per-kilogram costs are significantly lower than those of the existing Evolved Expendable Launch Vehicles (EELV) and SpaceX hopes to eventually offer launches for about 10% of today's EELV launch costs. Additionally, the Falcon 5 and 9 launch vehicles are currently the only vehicles offering engine out first stage reliability.

The Falcon 5 launch vehicle was selected for the Phoenix because of its intermediate size. The Falcon 1 is too small for any manned mission to be carried out. Alternatively, the Falcon 9's greater payload capacity was not necessary to fulfill the VSE goals and the higher launch and development costs for a larger manned spacecraft would have made the Phoenix less economical.

The Falcon 5 is a two stage launch vehicle. The main engine is the Merlin engine, developed internally at SpaceX. It is fueled by RP-1 and liquid oxygen. This engine is capable of producing 342,500 N of thrust at sea level and 409,200 N thrust in a vacuum. The specific impulse is 255 seconds at sea level and 304 seconds in a vacuum. The Merlin's thrust to weight ratio is 96. The first stage contains five Merlin engines and the second stage contains one.

A Payload Users Guide is currently only available for the Falcon 1 [3]. However, because of the commonality across the Falcon rockets, the payload interfaces are assumed to be similar. The Falcon uses the Lightband® system as an interface between the rocket and the payload. The separation of the payload from the rocket on orbit is initiated non-explosively and springs impart separation velocity. SpaceX provides the payload side of the interface. This is not counted against the payload mass. By extrapolation from Falcon

1 data, the center of gravity of the Falcon 5 payload is calculated to be no more than 6 cm from the centerline and no more than 2.8 m above the separation plane.

In order to increase the payload mass of Falcon 5 and to remove some restrictions on the size of the Phoenix, the payload fairing will not be used. The hull of the Phoenix is designed to withstand the launch environment without the additional protection of the fairing. The Falcon 5 rocket is not currently certified for transporting humans. Man-rating will occur before the first launch.

4. VEHICLE OVERVIEW

The Phoenix is a cylindrical spacecraft illustrated in Figure 4-1. It is 3.1 m tall and 3.6 m in diameter. It is designed to carry three people to low earth orbit, the ISS, or the moon. There are two hatches, one on the side for crew ingress and EVA activity and one on the aft end for ISS docking and crew egress. This spacecraft, propelled by two hydrazine/nitrogen tetroxide service engines, utilizes deployable solar arrays and does not recycle water or air. The Phoenix employs a unique ParaShield concept for re-entry. The large heat shield, constructed of ceramic fiber stretched over a truss structure, doubles in function as a parachute. The shield fabric is held up by 10 radial trusses composed of 12 segments each. It is stowed like an umbrella on the forward end of the spacecraft and opened right before re-entry. The primary launch site is Cape Canaveral, Florida and the primary splashdown site is off the coast of Florida. The Phoenix can be reused up to 10 times.

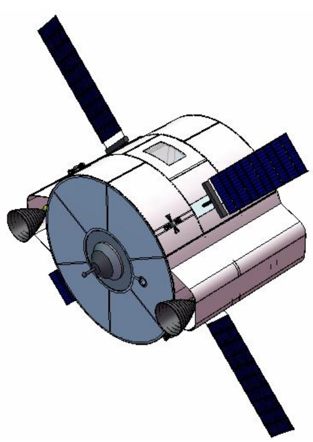


Figure 4-1: Phoenix spacecraft

General re-entry guidelines warrant that the heat shield be fully deployed prior to the de-orbit burn. To protect the heat shield, and prevent it from reversing the thrust, the nozzles need to be pointed away from the ParaShield. Furthermore, the service propulsion system needs to be mounted so that it assists the launch escape tower during an abort and the ISS interface needs to be located on one of the flat surfaces of the capsule. Therefore, the heat shield is placed on the front end of the Phoenix, opposite the ISS interface and the nozzles of the service propulsion system. During launch the crew seating is oriented facing up to sustain launch accelerations in an eyeballs-in orientation. Because the heat shield is located on the forward end of the spacecraft, the crew seating must be aft-facing during re-entry and splashdown. During flight the crew seats are reoriented for the appropriate re-entry position. Figure 4-2 illustrates the crew seating orientation for lift off and re-entry.

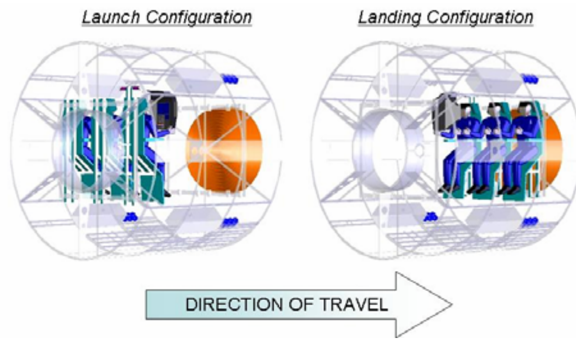


Figure 4-2: Crew seat orientation for lift off and re-entry

The Phoenix is designed to fit the payload capacity of the launch vehicle, which is 4,100 kg to a 200 km circular orbit. This is the reason for the maximum crew of three, despite the six-person LEO crew carrying capacity and four-person lunar crew carrying capacity of the slightly larger CEV. The additional payload capacity gained by removing the payload fairing is offset by the launch escape system. Table 4-1 shows the mass distribution of the various systems on board the Phoenix, including the service propulsion system (SPS), reaction control system (RCS), power system, life support, ISS interface, thermal protection system, EVA equipment, and re-entry system.

Table 4-1: Phoenix mass distribution

Service Propulsion System	
Oxidizer (kg)	337.4
Fuel (kg)	337.4
Pressurant (kg)	3.9
Inert Mass (kg)	428.2

Reaction Control System	
Oxidizer (kg)	33.2
Fuel (kg)	33.2
Pressurant (kg)	0.4
Inert Mass (kg)	75.8
Power	
Batteries (kg)	34.3
Solar Arrays (kg)	28.3
Life Support	
Crew/Gear (kg)	300
Oxygen/Tank (kg)	122.4
Nitrogen/Tank (kg)	238.7
Water/Tank (kg)	201.6
LiOH System (kg)	94.9
Food (kg)	40.3
ISS Interface	
Probe/Hatch (kg)	235
Avionics	
Guidance/Navigation (kg)	10.8
Communications (kg)	6.4
Computers (kg)	10
Thermal	
Active Thermal Protection (kg)	164
Extravehicular Activities	
Extravehicular Mobility Units (kg)	150
EMU Life Support Systems (kg)	45
Re-entry	
ParaShield (kg)	237.4
Truss Structure/Drive System (kg)	100
Total Mass (kg)	3268.6
Falcon 5 Launch Load (kg)	4100
Mass Margin	20.3%

A launch escape system is provided to pull the spacecraft away from the vehicle in the event of a launch catastrophe. Upon a launch failure, a hypergolic fuel naturally burns, whereas non-hypergolic fuel, like that of the Falcon 5, could explosively combust. The launch escape system, illustrated in Figure 4-3, is a 1,250 kg solid rocket motor. It has a thrust to weight ratio of about 10. The fuel is TP-H-1202, which is 21% aluminum, 57% ammonium perchlorate, 10% hydroxyl-terminated poly-butadiene (HTPB) binder, and 12% HMX (octogen). In the event of a launch failure, the launch escape system fires simultaneously with the service propulsion system to move the Phoenix away from the failing launch vehicle.

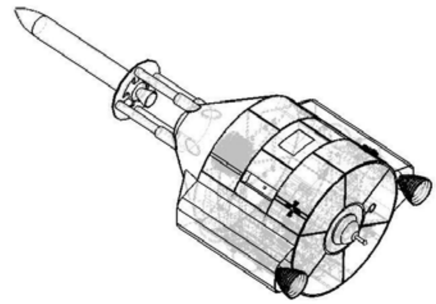


Figure 4-3: Phoenix launch escape system

To reduce costs and development time and to improve reliability, the Phoenix makes maximum use of proven technologies. The Technology Readiness Level (TRL) is a scale of 1 through 9, which NASA uses to specify the maturity of a particular technology and its readiness to be implemented into an operational system [4]. On this scale, 1 signifies a developing, untested concept and 9 represents systems which are proven through successful mission operations. Throughout this report, all vehicle subsystems have a TRL of 9 unless otherwise specified.

5. VEHICLE SUBSYSTEMS

The vehicle subsystems, including propulsion, structures, power, life support, ISS interfaces, avionics, thermal control, EVA capability, and re-entry and landing systems are described in detail in the following sections.

5.1. Propulsion

As mentioned previously, the Falcon 5 is capable of delivering the Phoenix into a circular orbit with an altitude of 200 km at 28.5° inclination. Therefore, a service propulsion system (SPS) is required to reach additional orbits. Per the mission objectives, a change in velocity of 550 m/s is needed to propel the Phoenix for rendezvous with the ISS. This constitutes the largest change in velocity for the various missions.

An additional requirement for the SPS is the ability to stop and restart the engines during the mission. Following the other design mandates for the Phoenix, the SPS has to also to be as compact and light as possible.

To that end, various liquid and hybrid chemical engines were evaluated using a numerical burn simulation [5]. The hybrid combinations included HTPB with hydrogen peroxide, HTPB with liquid oxygen, and HTPB with nitrogen tetroxide. Liquid combinations included liquid oxygen with liquid hydrogen, liquid oxygen with liquid fluorine, RP-1 with hydrogen peroxide, RP-1 with nitrogen tetroxide, and three hydrazine compounds (hydrazine, monomethyl-hydrazine, and unsymmetrical dimethyl-hydrazine) with nitrogen tetroxide [6]. The system masses and volumes of the ten candidate systems are plotted in Figure 5.1-1.

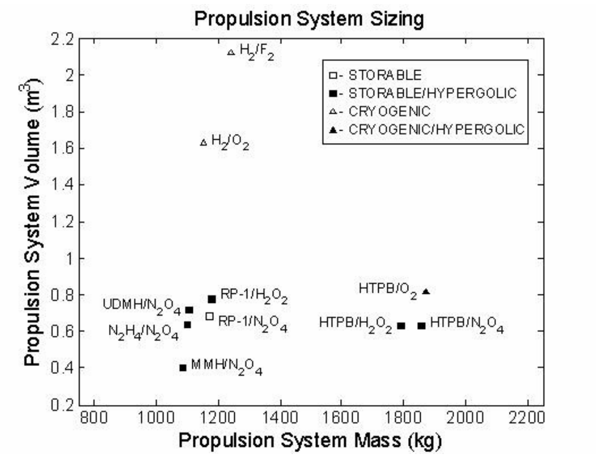


Figure 5.1-1: SPS sizes for various propellant combinations

From the aforementioned analysis, the optimum propellant combination proved to be monomethyl-hydrazine (MMH) and nitrogen tetroxide. Additional advantages of this propellant combination include that the liquids are not cryogenic and their combustion is hypergolic, negating the need for an igniter and promoting increased reliability for multiple engine burns [7].

The SPS onboard the Phoenix is pressure fed by a Helium tank with an initial pressure of 21 MPa. The pressurant is regulated to maintain a fuel tank pressure of 1.77 MPa and an oxidizer tank pressure of 1.80 MPa. Both propellants are introduced in the combustion chamber, which is maintained at 1.4 MPa, before passing through a convergent-divergent bell nozzle designed around an expansion ratio of 60 and a 45 kPa exit pressure. Directional control of the engine is attained from a two-axis gimbal assembly fixed to the combustion chamber. Two SPS engines are located on opposite sides of the Phoenix hull. The SPS placement allows for ParaShield storage on the forward end of the spacecraft and the ISS docking collar on the aft end. Overall, each SPS, shown in Figure 5.1-2, is capable of delivering 15,000 N of thrust for a total of 73 seconds.

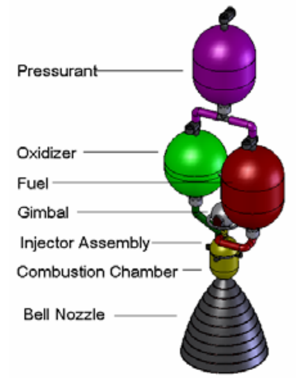


Figure 5.1-2: Service propulsion system

The reaction control system (RCS) is sized using an identical method to that used for the SPS. Again, as illustrated in Figure 5.1-3, the hypergolic combination of MMH and nitrogen tetroxide proved to be the best combination to minimize mass and volume for the system.

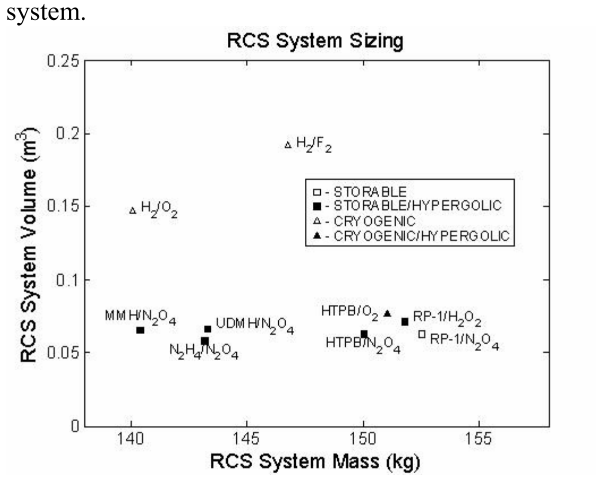
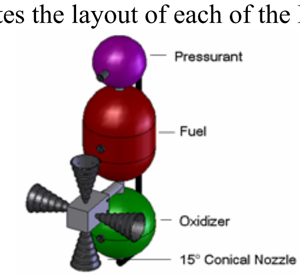


Figure 5.1-3: RCS sizes for various propellant combinations

As with the SPS, the RCS is a pressure fed system which uses high pressure helium at identical tank pressures. However, the RCS is not gimbal controlled and relies on four two-axis thrusters mounted symmetrically on the exterior of the Phoenix.

Each RCS cluster is capable of delivering a maximum of 530 N of thrust for 101 seconds. Figure 5.1-4 illustrates the layout of each of the RCS engines.



5.1-4: Reaction control system

The complete propulsion system, depicted in Figure 5.1-5, represents 1,250 kg of the gross mass for the Phoenix.

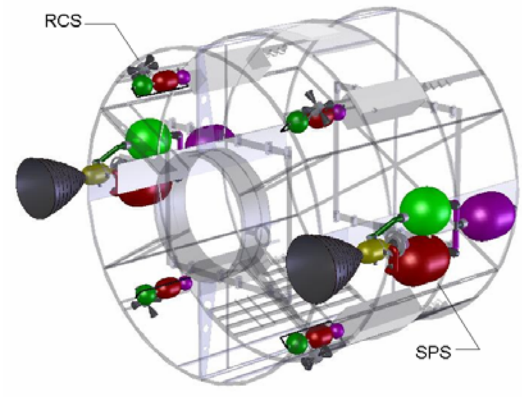


Figure 5.1-5: Propulsion system

5.2. Structures

Critical elements of the Phoenix spacecraft were analyzed to verify their respective structural integrity. These elements include the pressure vessels for the propulsion system, crew cabin, and heat shield truss assembly.

With the exception of the oxidizer and pressurant tanks for the RCS, all pressure vessels onboard the Phoenix are cylindrical tanks. All oxidizer tanks, fuel tanks, and hull walls are made of aluminum 7075; pressurant tanks are made of titanium Ti-6Al-4V; and combustion chambers are made of columbium. Table 5.2-1 lists the resulting safety margins for the applied loads. Note that burst pressure for the tanks is assumed twice the design load and an additional factor of safety of 1.5 has been applied.

Table 5.2-1: Safety margin of pressure vessels

Part	P_{burst} (MPa)	Radius (m)	t_{wall} (mm)	σ_{hoop} (MPa)	Safety Margin
Service Propulsion System Tanks					
Combustion	2.80	0.103	5	87	2.58
Oxidizer	3.60	0.241	10	130	0.21
Fuel	3.55	0.241	10	128	0.23
Pressurant	42.00	0.241	20	759	0.32
Reaction Control System Tanks					
Combustion	2.80	0.020	2	41	6.57
Oxidizer	3.60	0.111	5	60	1.63
Fuel	3.55	0.111	5	118	0.34
Pressurant	42.00	0.082	5	517	0.94
Crew Cabin					
Hull	0.20	3.600	5	109	0.45

The truss assembly for the heat shield is made of an array of lattice structures with slight length discrepancies that generate an arch. Phoenix uses ten of these trusses to support the heat shield fabric during the re-entry phase of the mission. A diagram of the truss design is provided in Figure 5.2-2.

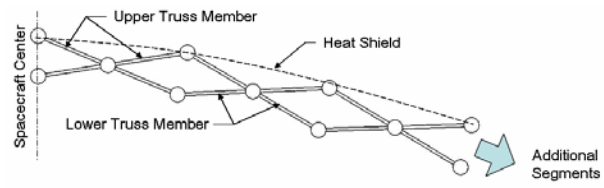


Figure 5.2-2: Truss diagram

For the analysis of the heat shield truss assembly, the matrix displacement method for trusses is applied using the maximum dynamic load experienced during re-entry [8]. Each segment of the truss is made of aluminum 7075 and has a cross sectional area of one square centimeter. Applying a margin of safety of 1.5 to the dynamic loads yields a maximum axial load of 40.1 MPa, corresponding to a safety margin of 1.56.

5.3. Power

The Phoenix power system uses a combination of photovoltaic cells and batteries to power the craft, including a 15% power margin. The photovoltaic cell power configuration was chosen as the lowest mass option after an analysis comparing the photovoltaic and battery combination to fuel cell power systems.

The power breakdown for the Phoenix is as follows: 750 W for computers, 250 W for remaining avionics, 150 W for trace contaminant removal, 411 W for recharging EMU batteries after EVA, and 50 W for actuators (including the actuators that will unfold the ParaShield). In order to provide this much power while the craft is in shadow, the solar cells must supply an additional 1.8 kW of power to recharge the batteries. Therefore, the solar cells must provide a total of 3.96 kW.

The trade study comparing power options included the mass of stored water in the weight of a photovoltaic (PV) cell system, since any water would have to be carried on board. A fuel cell system, on the other hand, would produce sufficient potable water to sustain the crew. In Figure 5.3-1, the weight of each power system is plotted versus the power consumption and the two lines intersect at approximately 2 kW. The plot is in terms of power consumed by the Phoenix subsystems, but includes additional solar cell capacity to recharge the batteries. For instance, the Phoenix subsystems consume approximately 2 kW of power, but the craft also needs the infrastructure to supply 1.8 additional kW to recharge batteries. This infrastructure mass, as well as the mass of stored water, is included in the total power system mass. The study suggests that a PV cell system would be more mass efficient for power requirements above 2 kW, but the fuel cell would be more mass efficient for lower power requirements. The Phoenix power consumption of 2 kW is at the intersection point of the two lines. A photovoltaic cell system was chosen with the assumption that a more advanced Phoenix mission could easily require more than 2 kW of power, and a photovoltaic cell and battery system could be scaled up accordingly, while a fuel cell system would no longer be practical.

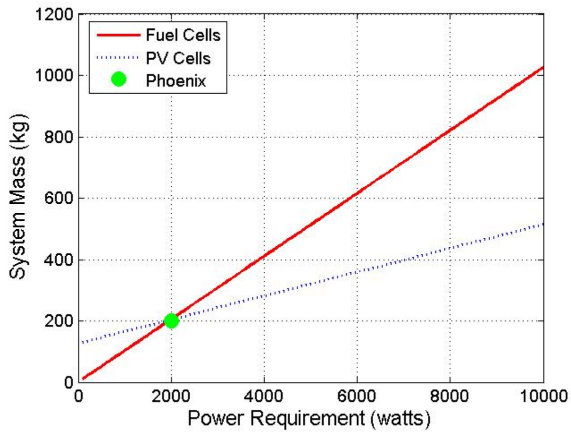


Figure 5.3-1: Mass of power systems vs. Phoenix power requirements

In addition to being low in mass, photovoltaic cells, shown deployed in Figure 5.3-2, are less expensive than fuel cells [9] and they are more suited to a short duration mission because less potable water needs to be carried. The fuel cells would produce more water than is necessary. Finally, a long duration stay at the ISS would be problematic with a fuel cell system because of complications in restarting fuel cells while in orbit. This is not an issue with the photovoltaic cell system.

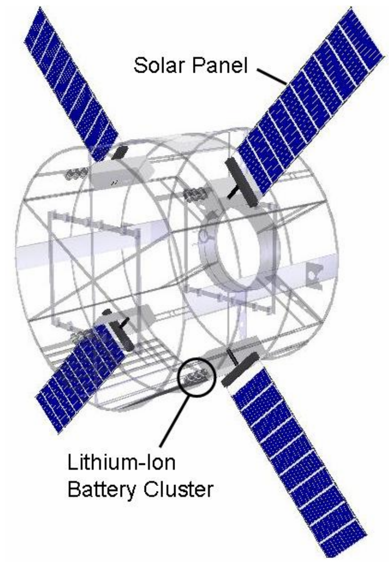


Figure 5.3-2: Phoenix power system

The chosen solar cells are ultra lightweight gallium arsenide Ultraflex® cells, with an efficiency of 23% and power density of 140 W/kg at I AU [9]. Lithium ion batteries, at 80 Wh/kg, were chosen to store power when the craft is in shadow. Sodium sulfide batteries (90 Wh/kg) were also considered, but were ruled out because of their high operating temperature (300°C - 400°C). Table 5.3-1 lists the mass and volume of the power system.

Table 5.3-1: Mass and volume of power system

Phoenix Power Consumption (W)	1853		BATTERY (Lithium Ion)	
Total Power Required (W)	3957		Li Ion Battery Power Density (Wh/kg)	80
			Li Ion Battery Power Density (Wh/L)	160
PHOTOVOLTAIC ARRAY (GaAs)			Li Ion Battery Mass (kg)	17
Efficiency	0.23		Li Ion Battery Volume (L)	8.6
Power Density at 1AU (W/kg)	140		Li Ion Battery Volume (m3)	0.01
Power Production (W/m2)	321			
Array Area (m^2)	12.3		INCLUDING BACKUP BATTERY:	
Array Mass (kg)	28.3		Total Battery Mass (kg):	34.3
Nightime Energy Storage (kW-hr)	1.37		Total Battery Volume (m^3)	0.02
Recharge Power (kW)	1.83		total power system mass (kg)	62.6

5.4. Life Support

The Phoenix must be capable of supporting three passengers for a maximum of 12 days. These passengers will require food, potable water, oxygen, and the cabin air must be clear of trace contaminants and carbon dioxide.

One of the main drivers of the life support system weight is water because each crew member will require at least 3.5 kg of water per day [9]. This value assumes that clothes wash water and hygiene water will be minimal for a mission lasting less than 2 weeks. There are three options for providing water: stored water,

water recycling using vapor compression distillation, and water production using fuel cells.

Water recycling was quickly dismissed as an option for such a short mission, since the mass of the equipment would cancel out any weight savings. The remaining options, therefore, were water production and water storage. An analysis was done comparing a system with photovoltaic cells, batteries, and stored water, to a fuel cell system that would produce water. This analysis, which is described in detail in Section 5.3, suggests that a fuel cell system would be too heavy on a mission of this length, even considering the weight savings from water production.

In addition to water, Phoenix has the capacity to store 61.2 kg of oxygen at 300 atmospheres in two separate, redundant air supply loops. Because the Phoenix will be docked at the ISS, it must have the capability of sustaining an atmosphere at 14.7 psi with 21% oxygen and 79% nitrogen. Therefore, the Phoenix will also carry 119 kg of Nitrogen, enough to counteract a leakage rate of up to 1% of Nitrogen per day and to allow three complete repressurizations of the cabin after EVA. Cabin air will be circulated through Lithium Hydroxide canisters for carbon dioxide scrubbing, and filters will be used to remove trace contaminants. The Lithium Hydroxide canisters weigh about 60 kg, including packaging [10]. Phoenix will also carry 22 kg of food. Table 5.4-1 lists the mass and volume of consumables.

Table 5.4-1: Mass and volume of consumables

Consumable	Safety Margin	Pressure Atm.	Mass with Safety Margin kg	Density kg/m^3	Volume m^3	Container Mass kg	Total Mass kg
Water	1	1	126	1000	0.126	75.6	202
O_2	2	300	61.2	400	0.153	61.2	122
Nitrogen	2	325	119	378	0.316	119	239
Food	1	n/a	22.3	1000	0.022	18.0	40.3
LiOH	1.5	n/a	58.9	611	0.132	36.0	94.9

5.5. ISS Interfaces

Many of the Phoenix design constraints are imposed by the requirement to dock to the International Space Station. These include constraints on maximum external temperature, data bus compatibility, and consumables selection. In addition, the docking procedure requires a physical interface, to allow a shirt-sleeve crew transfer and consumables hookup. There are three options for docking to the ISS: a (passive) berthing assisted by the Space Station Remote Manipulator System, active docking at the U.S. segment, and active docking at the Russian segment.

Phoenix will dock with the Russian segment. The probe-and-drogue mechanism used to dock to the Russian segment is much simpler (and consequently lighter) than the Androgynous Peripheral Docking System (ADPS) required to dock with the U.S. segment. Another alternative, berthing at the U.S. segment, was considered impractical due to the large radius of the Common Berthing Module. The Russian module is also equipped with the KURS automatic rendezvous system, which could be employed by an autonomous unmanned version of the Phoenix.

The Phoenix will use a probe and drogue docking mechanism similar to the system designed by Energia for the European Space Agency's Automated Transfer Vehicle (ATV). The mechanism is relatively compact, and similar to systems that have been used extensively in the past. The docking collar, shown in Figure 5.5-1, also allows direct connection to ISS consumables and power, eliminating the need for umbilicals. The system weighs approximately 235 kg, and the tunnel diameter is 80 cm. [11].

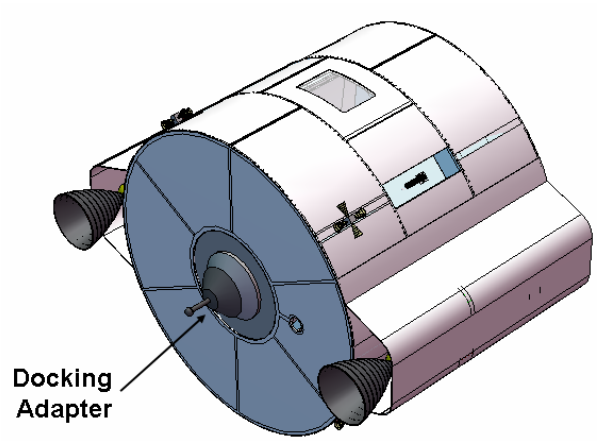


Figure 5.5-1: Phoenix docking adapter

5.6. Avionics

The Phoenix Avionics subsystem takes advantage of off-the-shelf hardware that has already been proven in the field. The cost of integrating these purchased components into the craft is significantly lower than the cost of designing new components. Crucial systems have several layers of redundancy, and all components use radiation hardened electronics to reduce the likelihood of single event upsets. All systems are also compatible with the MIL-STD-1553 data bus, which is standard on the ISS.

The navigation and guidance systems, crucial for the crew's safe passage and return to Earth, include three separate types of devices. The primary navigation system is a pair of Honeywell Miniature Inertial Measurement Units (MIMU). More than 40 of these units have been launched successfully on satellites, including deep space applications [12]. The Phoenix is also equipped with two Ball Aerospace CT-602 Star Trackers [13], which are used to calibrate the MIMUs and correct for drift. A Crewman Optical Alignment Sight (COAS) is also provided. The COAS is used to calibrate the other navigation devices, and to serve as an emergency backup in case the other systems fail.

The Phoenix must be capable of communicating with the ground, with the ISS, and with the astronauts during EVA. Cincinnati Electronics TDRSS S-Band transmitter and receiver [14, 15] are used to handle ground and ISS communications. A UHF transceiver from the same manufacturer is used to communicate with the astronauts during EVA [16].

Data processing is conducted by three single-board 3U CompactPCI computers built for satellites by BAE Systems. Two of the computers are running in a master-slave configuration. The third computer is physically separated from the others, and is running on different software, to allow for additional redundancy in case of a software fault. Each computer is housed in an enclosure for shielding and thermal control. This configuration is assumed to use 250 W of power. Table 5.6-1 lists the mass, volume, and power of the avionics components.

Table 5.6-1 Avionics mass, volume, and power

Component	Qty.	Power W	Mass kg	Volume m^3
MIMU	2	32	4.7	3.1
Star Tracker	2	8	5	3.6
COAS	1	~0	1.1	4.0
TDRSS Receiver	1	37	2.28	3.7
TDRSS Transmitter	1	<8	2.05	2.5
UHF Transceiver	1	60	2.07	2.0
3UCompactPCI	3	250	10	7.1

5.7. Thermal

Thermal control of the Phoenix includes both active and passive control schemes. Passive control is accomplished by the selection of thermal coatings on the craft exterior, the use of multi-layer insulation (MLI), and heaters around temperature-sensitive areas. Active thermal control has been required on every manned craft, and Phoenix is no exception, since it must be capable of reacting to unexpected thermal loads. The active thermal control system includes both external and internal cooling loops to draw metabolic and equipment heat loads from the cabin and reject them to space. The mass of the thermal control system is estimated as 4% of the spacecraft gross mass [17].

Because Phoenix will dock with the ISS, its external temperature must be maintained below 113°C [18]. This requirement is meant to protect astronauts during EVA and to prevent thermal damage to their extravehicular mobility units (EMU). As a result, the thermal coating chosen to maintain the baseline exterior temperature below 113°C at the worst thermal case is aluminized Kapton® manufactured by DuPont.

The Phoenix can be exposed to six distinct thermal environments: Earth orbit, cis-lunar orbit, and lunar orbit; all with and without incidental sunlight. For purposes of this analysis, the Phoenix is approximated as a cylinder 3.6 m in diameter and 3.1 m long. Solar flux is approximated as 1353 W/m^2, Earth albedo as 0.3, moon albedo as 0.07, Earth apparent temperature as 280°K, and apparent temperatures of 100°K and 340°K on the dark and light side of the moon, respectively. It is assumed that 90% of the spacecraft internal power is dissipated as heat radiated into space.

A heat balance of the spacecraft in each of the eight environments, shown in Table 5.7-1, indicates that an absorptivity to emissivity ratio of 0.5 is adequate to maintain the spacecraft baseline temperature below 113°C.

Table 5.7-1 Heat balance on the Phoenix

Environment	External Temp K	Power Dissipated W	Environ. Temp K	Solar Flux W/m^2	Albedo W/m^2	α/ε Ratio
Near Earth Sun	352	3407	280	1353	406	0.5
Near Earth No Sun	320	3407	280	0	0	0.5
Near Moon Sun	386	3407	343	1353	94.7	0.5
Near Moon No Sun	257	3407	100	0	0	0.5
Deep Space Sun	300	3407	4	1353	0	0.5
Deep Space No Sun	256	3407	4	0	0	0.5

5.8. EVA

Because of the small size and weight of the Phoenix, it does not contain an airlock. However, the spacecraft is required to have EVA capability during all stages of orbital flight. To satisfy this requirement, three shuttle extravehicular mobility units (EMU) are stowed in on-board lockers. If a planned or contingency spacewalk is required, each of the three crew members dons an EMU, the capsule is depressurized, and the main hatch is opened to the vacuum of space as in the Gemini program. If EVA is required while the Phoenix is docked to the ISS, the suited astronauts egress and ingress via the US airlock.

The mass of each EMU is 50 kg and the accompanying portable life support system (PLSS) is 15 kg [19]. The power consumption is 148 W. The EMU's are launched fully charged to minimize on-orbit charging. If multiple EVAs are required during a single mission, the EMU's must trickle charge at 137 W for at least 12 hours before they can be used again [20]. The maximum EVA duration is 9 hours. The requirements for recharging the EMUs are listed in Table 5.8-1.

Table 5.8-1: Estimated requirements for recharging shuttle EMUs after 9 hour EVA

EMU Power Requirements		
Voltage provided	18.5	VDC
Current Provided	8	A
Peak Power Consumption	148	W
Max duration of EVA	9	hrs
Energy required for 1 EVA:	1.3	kWhrs
Battery Efficiency	90%	
Required Energy Storage per Battery	1.5	kWhrs
Energy Required to Recharge	1.6	kWhrs
Time to Recharge	12	hrs
Power Required to Recharge	137	W
Total Power Required to recharge 3 EMUs	411	W

5.9. Re-entry and Landing

The re-entry and landing device selected for the Phoenix is the ParaShield. This unique concept involves using a heat shield with a low ballistic coefficient, which deploys like an umbrella prior to re-entry. The heat shield also doubles as a landing parachute. Phoenix's heat shield is made of high temperature fabric stretched over 10 aluminum spars with 12 folds when stowed. Figure 5.9-1 illustrates the heat shield in stowed and deployed configurations. This lightweight design has many benefits. The umbrella configuration has great stowage flexibility. Unlike the ablative heat shields of Mercury, Gemini, and Apollo, this design does not dictate the shape of the capsule. Instead of a blunt-end cone capsule, the vehicle can be any shape that fits inside the wake of the shield.

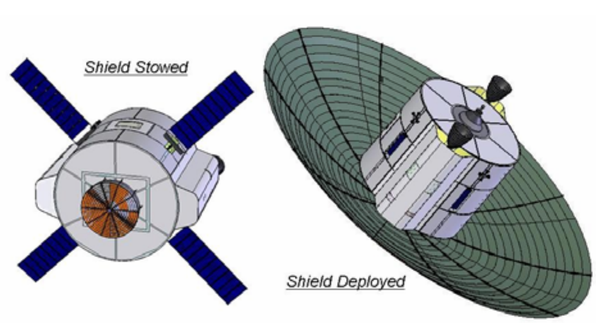


Figure 5.9-1: Phoenix ParaShield in stowed and deployed configurations

A fourth order Runge-Kutta integration was performed to determine the flight characteristics of the Phoenix with the ParaShield during re-entry from a low earth orbit (LEO) mission and a direct return from a lunar mission. The re-entry profile after a LEO mission is initiated in a Hohmann transfer orbit from mission altitude to an altitude of 165 km with a shallow initial flight path angle of 2°. This is above the majority of the dense earth atmosphere and allows for a complete analysis of re-entry conditions. The initial velocity ranges from 7.9 km/s for a mission altitude of 600 km to 7.8 km/s for a mission altitude of 200 km. The driving design variable is the ballistic coefficient. A desirable value for a ParaShield is on the order of 200 Pa [21] to maintain low heat shield temperatures. Using the inert mass of the Phoenix vehicle and a coefficient of drag of 0.157, the ParaShield is to have a 6 m radius. The corresponding ballistic coefficient is 225.27 kg/m^2. The selected lift to drag ratio is 0.23 for spacecraft controllability during re-entry. Though the Falcon 5 launch places strict center of gravity restrictions on the Phoenix, the center of gravity is shifted as consumables are expended and allows for a non-trivial lift to drag ratio.

The duration of the LEO re-entry from the de-orbit burn to touchdown is 19 minutes. The altitude profile and sensed acceleration over time are illustrated in Figure 5.9-2. The maximum acceleration is 3.358 g; a safe value for astronauts and sensitive payloads. For comparison, the Apollo Command Module (CM) reached a maximum sensed acceleration of 6.5 g [22]. The altitude profile includes a slight phugoid oscillation near an altitude of 74 km. This is due to the coupling of altitude and air density; a rapid increase in air density temporarily produces additional lift. This phugoid is also reflected in the sensed acceleration profile as a temporary reduction in re-entry acceleration.

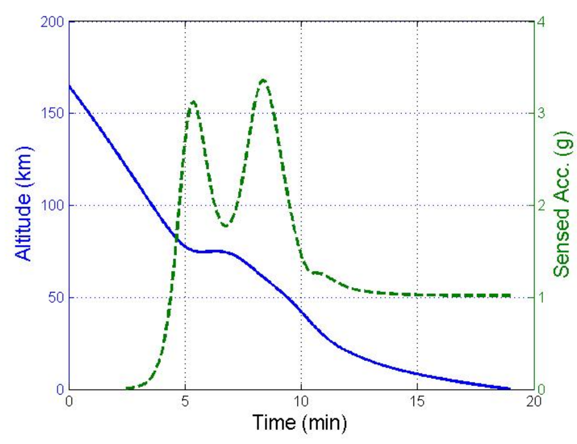


Figure 5.9-2: Re-Entry altitude on the left ordinate and sensed acceleration on the right ordinate as a function of time

Due to the low ballistic coefficient, the re-entry heat dissipation is distributed over the larger area of the heat shield and reduces the peak ParaShield temperature. Figure 5.9-3 shows the shield temperature profile during LEO re-entry. The majority of the deceleration, and therefore heat dissipation, during re-entry occurs during the first 10 minutes. After this time the vehicle is no longer in a hypersonic flight regime and the heat dissipation drops. This is also evident in the sensed acceleration plot in Figure 5.9-2. After the first half of the LEO re-entry, the sensed acceleration settles to the acceleration due to earth's gravity. The maximum temperature the heat shield reaches is 1836.8 °F. As a relative comparison, ablative heat shield of the three-person Apollo CM reached a temperature of 5000 °F [22].

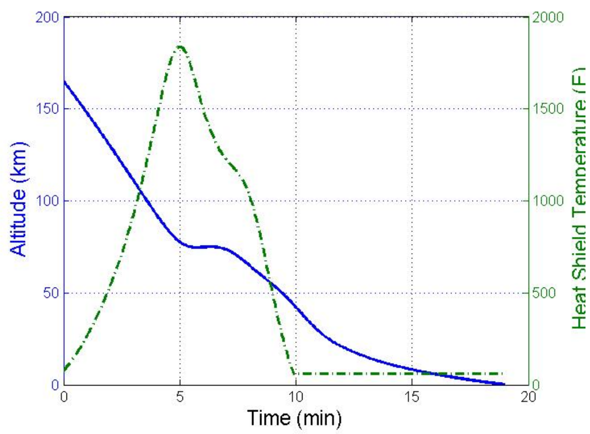


Figure 5.9-3: Re-Entry altitude on the left ordinate and heat shield temperature on the right ordinate as a function of time

The shape of the heat shield is the section of a sphere which would enclose the end of a 90° cone of equal radius. The volume inside this cone is considered to be the wake of the shield and represents the volume of space protected from the heat of re-entry. The shield is made of 3M™ Nextel™ 312 ceramic fiber. This material was selected because its density, 2.70 g/cc, is lowest among competitors and the rated temperature, 2200°F, is sufficient for the re-entry trajectory [23] with a 19.8% margin.

The overarching assumption in the aerodynamic analysis of the ParaShield is the applicability of the solid boundary condition, despite the fact that the material is a permeable fabric. If a substantial amount of the hot flow gets through the shield, it will impinge on the unprotected spacecraft and cause damage. When the ParaShield is modeled as a filter governed by Darcy's law of filtration, it is found that 26 layers of 0.25 mm thick Nextel™ 312 fabric are necessary to reduce flow seepage to 1% and an additional 4 layers decreases the flow seepage to 0.25% [21]. The 30 layers of fabric have a thickness of 7.5 mm and a mass of 861 kg. To reduce the mass and thickness of the fabric, the Phoenix ParaShield has only 4 layers of 0.25 mm thick Nextel™ 312 fabric for thermal protection and limits seepage with a 2 mm non-permeable silicone coating with a density of 1.47 g/cc. The mass of the heat shield is 237.4 kg. This is without struts and motors.

The ParaShield is completely deployed prior to the de-orbit burn by activating the redundant motor, which turns the lead screw and opens the ParaShield. The center of mass of the spacecraft is located low inside the ParaShield to maintain stability and prevent the spacecraft from pitching over. The orientation of the crew inside the spacecraft is now opposite of that during launch. During final descent, the ParaShield acts as a parachute. The ParaShield is jettisoned immediately before splashdown. Upon impact, floatation devices around the spacecraft inflate to maintain positive buoyancy and upright orientation until the sea support ship arrives. The vehicle landing configuration is illustrated in Figure 5.9-4.

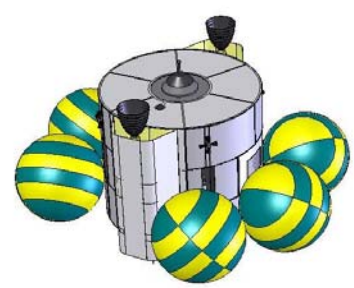


Figure 5.9-4: Phoenix landing configuration

The discussion of the Phoenix LEO re-entry has thus far focused on the case where the roll angle of the spacecraft is held steady at 0° during the entire re-entry. Increasing the roll angle increases the crossrange distance and decreases the downrange distance, thus changing the landing position. Varying the roll angle from 0° to 180° results in an elliptical area of possible landing positions called a landing footprint. It is desirable for this landing footprint to be large for flexibility of possible de-orbit times from various orbital passes for a landing in the designated area. The primary landing site for the Phoenix is in the Atlantic Ocean off the coast of Cape Canaveral for close proximity to launch and ground operations. The landing footprint of the Phoenix is illustrated in Figure 5.9-5. The outline encloses the theoretically achievable landing positions. However, at high roll angles the sensed acceleration and heat shield temperature increase beyond allowable limits. When the roll angle is increased past 29°, the sensed acceleration is above 3.5 g and determines the roll angle limit.

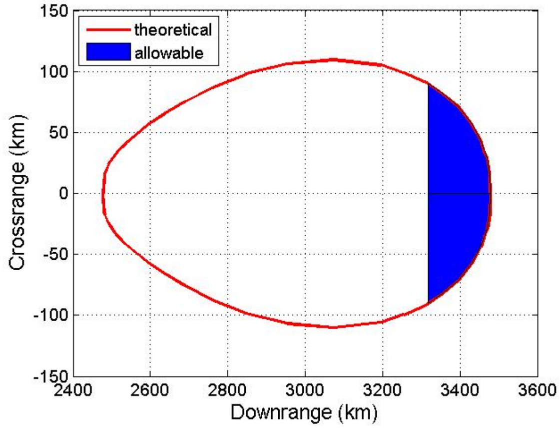


Figure 5.9-5: Landing footprint of Phoenix, only roll angles up to 29° result in allowable sensed acceleration and heat shield temperatures

The re-entry from a lunar mission is different than a LEO re-entry because of the higher initial velocity. The direct re-entry from the moon is initiated at an altitude of 180 km, with a flight path angle of 6° and velocity of 10.9 km/s. Aerobraking is used to reduce the velocity rather than onboard thrust in an effort to conserve fuel and reduce spacecraft mass. Figure 5.9-6 shows the trajectory of the lunar re-entry overlaid with sensed acceleration. The Phoenix performs three phugoid oscillation maneuvers and orbits the earth 2.15 times before touching down 3 hours and 51 seconds after initiating re-entry. The maximum sensed acceleration reached is 3.074 g. There is sufficient fuel on board during re-entry that trajectory corrections can be made after each oscillation to correct for accumulated errors and assure an accurate re-entry profile.

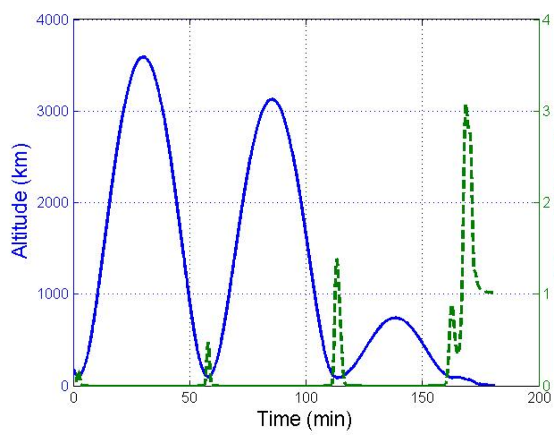


Figure 5.9-6: Re-Entry altitude on the left ordinate and sensed acceleration on the right ordinate as a function of time during a direct re-entry from a lunar mission

Figure 5.9-7 illustrates the heat shield temperature profile during the lunar re-entry. High temperature spikes are experienced at the bottom of each phugoid oscillation. The maximum temperature is 2072.3 °F.

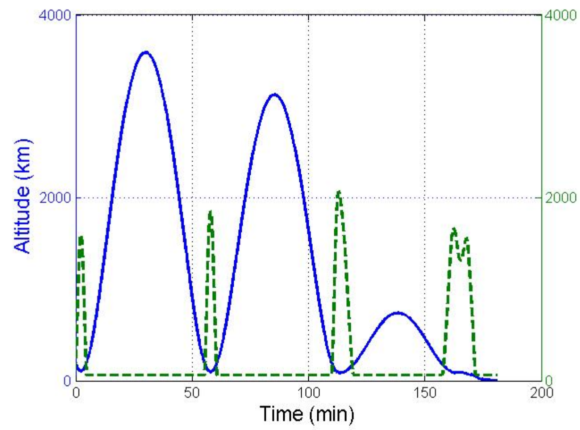


Figure 5.9-7: Re-Entry altitude on the left ordinate and heat shield temperature on the right ordinate as a function of time during a direct re-entry from a lunar mission

Because of the aerobraking, the final portion of the direct re-entry from a lunar mission is more benign than the LEO re-entry. The maximum sensed acceleration is 8.5% less. The two trajectories, pictured from the time in each when the spacecraft is at 165 km altitude, are illustrated in Figure 5.9-8. No landing footprint is calculated for the lunar re-entry mission because lunar departure is timed appropriately for correct splashdown location.

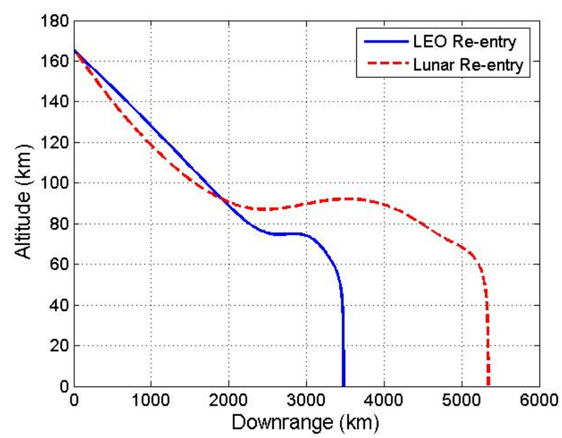


Figure 5.9-8: Re-entry trajectories from a LEO mission and a lunar mission

The TRL of the ParaShield system is 6. This concept was implemented on a small capsule designed and built at MIT in 1989 called Skidbladnir [24]. It was set to launch on the maiden voyage of the American Rocket Company's hybrid rocket. Unfortunately, a launch failure occurred and Skidbladnir never got off the ground. A TRL level of 6 is a system that was demonstrated in a relevant environment. MIT's capsule was thoroughly tested in free fall and high temperature environments. Had the rocket not failed, this technology could have been demonstrated in space and held a TRL level of 7.

6. COSTS AND SCHEDULE

The cost analysis comparing the Phoenix to the proposed CEV was performed using the NASA cost estimating relations [25]. The most recent version of these equations calculated the costs in $M2005. All of the costs provided in this report are converted into $M2006 using a 3.2% inflation rate in the net future value formula. The Phoenix nonrecurring and 1st unit production costs were found to be $1645.7M and $125.5M, respectively. These are calculated using the vehicle level cost estimating relations for a manned spacecraft as a function of inert mass, which is 2708 kg. Because the Falcon 5 is not man-rated, a conservative estimate of $1 billion is added to the calculated nonrecurring cost for man-rating the launch vehicle, bringing these costs to $2645.7M. The nonrecurring costs are assumed to be paid for during the first five years of the program. A beta function is used to spread these costs with values of 0.5 used for both cost fraction and peak width. The complete launch price of the Falcon 5 is listed at $18M. This cost is added to each Phoenix vehicle flown. It is assumed that a refurbishment of a vehicle for reusable flight is 20% of the original price. This is a very conservative estimate; the shuttle refurbishment fraction is 6-20% [9]. A learning curve of 80% is assumed for both the production and refurbishment costs and the cost discounting rate is 10%. A 30% cost margin is included in all estimates excluding the launch cost of the Falcon 5.

The CEV costs are calculated using the same estimating relations. This cost analysis includes CEV manned vehicle, and the first and second stage of the shuttle derived Ares Crew Launch Vehicle (CLV). NASA's Project Constellation will also include the Ares Cargo Launch Vehicle (CaLV), the Earth Departure Stage, and the Lunar Surface Access Module. Similar components will have to be developed for the Phoenix lunar mission and are not reflected in either cost analysis. Because the Phoenix is designed for three crew members, as opposed to the four-person CEV, the lunar mission components for Phoenix will be relatively less expensive. The CEV and first stage of the CLV are reusable up to ten times. The second stage of the CLV is not reusable.

The CEV nonrecurring and 1st unit production costs are listed in Table 6-1. These are calculated as a function of inert mass [26] using the cost estimating relations for a manned spacecraft and launch vehicle stage where appropriate. The nonrecurring cost of the first stage of the CLV is assumed to be half of that calculated because this stage is derived from the shuttle solid rocket booster technology. The second stage is a completely new design. Because the CLV is designed specifically for the CEV, the man-rating expenses are already included.

Table 6-1: Nonrecurring and 1st unit production costs for CEV components

Vehicle Component	Nonrecurring Cost	1st Unit Production Cost
CEV	$4044.0M	$370.3M
CLV 1st Stage	$2116.6M	$356.8M
CLV 2nd Stage	$1812.7 M	$128.6 M

The CEV cost analysis is performed using the same assumptions as the Phoenix. The nonrecurring costs are spread over the first five years using a beta function with values of 0.5 used for both cost fraction and peak width. Refurbishment of each vehicle component is assumed to cost 20% of the original cost, the discounting rate is 10%, and an 80% learning curve is expected. A 30% cost margin is included.

The cost per flight depends on the flight schedule. To achieve all of the VSE goals, it is expected that this program will last 25 years. Table 6-2 details the suggested 100-mission flight schedule in a condensed format. The ultimate flight frequency of five per year is on par with past NASA manned space exploration programs. To achieve this flight frequency, 12 vehicles are constructed such that each vehicle completes at most 10 missions. The net present value of the entire Phoenix program with this flight schedule is $3,898M or $39.0M per flight. The CEV program with the same flight schedule will cost $14,306M or $143.1M per flight. This analysis shows that the Phoenix will cost 27% of the CEV price.

Table 6-2: Condensed flight schedule, multiple year entries list flights and vehicles per year

Program Year	Calendar Year	Flights	New Vehicles	Refurb. Vehicles
1-4	2007-2010	0	0	0
5	2011	2	1	1
6	2012	3	1	2
7-8	2013-2014	5	2	3
9-14	2015-2020	5	1	4
15-25	2021-2031	5	0	5
Total	**25 years**	**100**	**12**	**88**

The cost per flight varies greatly depending on the number of flights on the manifest. With a greater flight frequency, the nonrecurring costs are more evenly distributed among the flights and the learning curve further reduces the anticipated expenses. Figure 6-1 illustrates the correlation between the number of flights throughout the program to the net present value cost per mission. Similar flight schedules were developed for manifests ranging from 25 to 200 flights in increments of 25. The flight schedules all include paying off the nonrecurring costs during the first five years, with flights starting in 2011, and reaching full flight frequency by 2013. The price gap between the Phoenix and the CEV decreases as the number of flights increases because nonrecurring costs play a smaller role in the pricing.

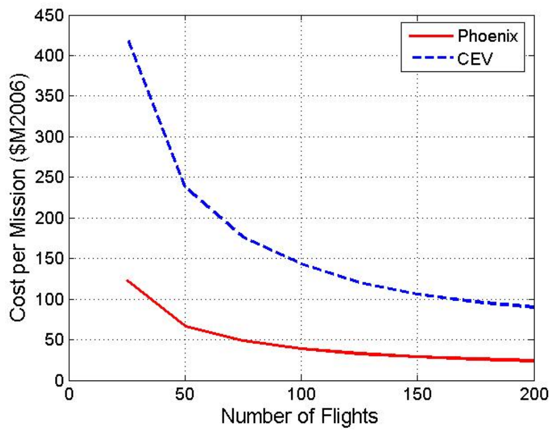


Figure 6-1: Cost per mission for the Phoenix and CEV as a function of the number of flights

7. CONCLUSION

The Phoenix, shown in Figure 7-1, is a viable alternative to NASA's proposed CEV for a fraction of the cost. The lightweight ParaShield concept allows for a greater internal volume than the alternative cone shape, which would be dictated by a more traditional heat shield design. This ParaShield system holds the lowest TRL level of the entire vehicle and dictates that the TRL level of the Phoenix is 6. Because the selected launch vehicle is the relatively inexpensive Falcon 5, the cost per mission of the Phoenix is only 27% of the CEV. Despite the fact that the crew size is only three, as opposed to the four-to-six-person CEV, the spacecraft is versatile and capable of performing three reference missions comparable to those slated to be performed by the CEV. Though a reduced crew will only be able to perform a diminished workload of tasks and experiments, the per-mission cost of the Phoenix spacecraft allows for almost four times as many missions to be flown on a CEV budget.

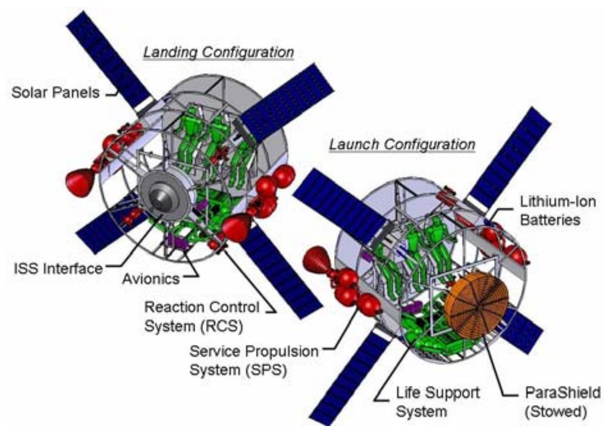


Figure 7-1: Internal layout of the Phoenix

8. ACKNOWLEDGEMENTS

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