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PROJECT FREEBIRD

An Orbital Transfer Vehicle

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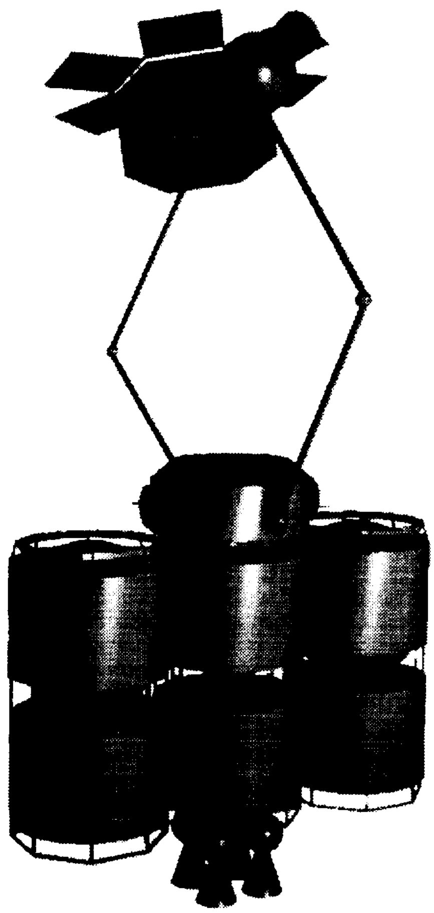

16.83 Space Systems Engineering
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Spring 1994

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PROJECT FREEBIRD

An Orbital Transfer Vehicle

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Abstract

Freebird is a space-based orbital transfer vehicle designed to repair and deorbit orbital assets. Freebird is based at International Space Station Alpha (ISSA) at an inclination of 51.6° and is capable of three types of missions: crewed and teleoperated LEO missions, and extended robotic missions. In a crewed local configuration, the vehicle can visit inclinations between 30.8° and 72.4° at altitudes close to 390km. Adding extra fuel tanks extends this range of inclinations up to 84.9° and down to 18.3°. Furthermore, removing the crew module, using the vehicle in a teleoperated manner, and operating with extra fuel tanks allows missions to polar and geosynchronous orbits.

To allow for mission flexibility, the vehicle was designed in a semi-modular configuration. The major system components include a crew module, a "smartbox" (which contains command, communications, guidance and navigation equipment), a propulsion pack, extra fuel tanks, and a Vehicle Storage Facility (VSF) for storage purposes.

To minimize risk as well as development time and cost, the vehicle was designed using only proven technology or technology which is expected to be flight-qualified in time for the intended launch date of 2002. And, because Freebird carries crew and operates near the Space Station, it must meet or exceed the NASA reliability standard of 0.994, as well as other standard requirements for such vehicles. The Freebird program was conceived and designed as a way to provide important and currently unavailable satellite repair and replacement services of a value equal to or exceeding operational costs.

PROJECT FREEBIRD

An Orbital Transfer Vehicle

16.83 Space Systems Engineering
Massachusetts Institute of Technology
Spring 1994

Final Report

Part I: Mission Definition
Part II: Vehicle Design
Part III: Operations

NASA/Universities Space Research Association
Advanced Design Program

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No animals were harmed in the production of this document.

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Introduction

Freebird was conceived as a space-based orbital transfer vehicle whose primary function is the repair and/or deorbiting of orbital assets and whose missions are all performed with the International Space Station Alpha, ISSA, as a base at an inclination of 51.6°.

On-Orbit Needs

A vehicle based at International Space Station Alpha (ISSA) could serve a variety of mission scenarios. The nature of the mission scenarios and the mission objectives that were established for Freebird depended heavily upon the apparent needs of the current United States and world space programs. Three needs of today's space programs were determined to be: the need to tend existing assets in space, the need for exploration, and the need to retrieve and rescue stranded astronauts from other space platforms.

The need to tend existing assets is clear because of the large number of assets in orbit which are either obsolete or unserviceable by existing space vehicles. In particular, there are assets in LEO (Low Earth Orbit) between the altitudes of 200 km and 1000 km and assets as far away as GEO (geosynchronous orbit) at 36 000 km in altitude. Several of the assets in orbit--including communications, observation, and tracking satellites launched by various nations--contribute to the growing dilemma of orbital debris. Once a satellite becomes obsolete, it most likely remains in orbit, interfering with other orbital operations and occupying space where new satellites could be deployed and used effectively. Freebird, if designed to be capable of tending orbital assets (that is, repairing existing assets and reboosting or deorbiting obsolete equipment), would extend the lifetime of equipment in orbit and provide a cost-effective method of extending the current world space program.

There are currently hundreds of satellites in orbit which could potentially be serviced to extend their lifetimes or to enable them to recover from mission failures. The Space Shuttle, however, the only vehicle currently capable of such service missions, is limited to low earth orbits and inclinations below 40°, (although it can reach slightly higher inclinations with a sacrifice of payload mass). This leaves a large number of satellites that cannot be reached by today's capabilities. A vehicle such as Freebird, which can not only reach more of these assets, but also perform tending missions more cheaply, would help ensure that this vital task could be continued in the future.

In addition to the primary need for tending assets, the need for exploration can be fulfilled by Freebird as well. Based at ISSA, Freebird has the potential to reach Lagrangian points or to orbit the moon.

Certainly, the rescue of stranded astronauts from other space platforms is a valuable capability that would augment the public's faith in the world space program. In general, rescue capability would add an additional level of safety for crewed missions in space.

Mission Objectives

The mission objectives of the Freebird system were derived from the on-orbit needs discussed above. The two primary objectives for Freebird are:

1. The vehicle shall be capable of tending assets in LEO, between the inclinations of 13.2° and 90°.
2. The vehicle shall be capable of tending assets in GEO, at equatorial inclinations.

These primary objectives establish what the system capabilities must be. Moreover, the vehicle was designed for tending assets at various altitudes and inclinations, where tending refers to the repairing, reboosting, and deorbiting of assets, and not the deploying of assets--an activity which is already adequately performed by existing space vehicles and which is unnecessary in the light of the fact that Freebird is based at the ISSA (and boosting satellites into orbit after they have been launched to the inclination of 51.6° is not a particularly advantageous or fuel-efficient maneuver).

Because of the primary mission objectives, Freebird will have both teleoperated and crewed capabilities. In some cases, astronauts may prove to be more effective than robots in conducting repairs. Non-crewed capabilities are reserved for missions on which the potential for radiation damage to an astronaut performing an EVA (Extra Vehicular Activity) would be greater than safety standards permit.

In addition to the two primary objectives, two secondary objectives were designed for:

1. The vehicle shall be capable of rescuing stranded crews from other crewed orbital missions.
2. The vehicle shall be capable of ferrying cargoes between orbits.

Despite the added complexity of incorporating rescue capability into the Freebird design, the benefits of having such a capability are worthwhile.

These four mission objectives were significant design drivers of the Freebird system.

Mission Scenarios

Three basic mission scenarios were developed on the basis of the established objectives: 1. LEO missions, 2. GEO missions, and 3. Rescue operations.

A mission in LEO is the repair of a recently deployed satellite at a 28.5° inclination. This mission allows astronauts to work for 3 days on repairs using remote manipulator arms for grappling with the satellite. In addition, the astronauts will perform EVA with the satellite.

A simpler LEO mission is to perform repair as in the above scenario without astronauts. Instead, in a teleoperated scenario, repairs will be performed by the remote manipulator arms which will be controlled from ISSA or the ground.

A GEO mission would be teleoperated since it would not be practical to expose astronauts to the harsh radiation environment of GEO. A typical GEO mission is the repair or removal of a non-operational satellite.

A rescue mission could be something as simple as getting a Soyuz capsule that missed its target orbit into its desired orbit, or as complicated as evacuating a crew of seven from a vehicle similar to the Space Shuttle.

System Level Requirements

The primary and secondary mission objectives established not only the types of missions that Freebird would be likely to accomplish, but also the system level requirements that would be necessary for the operation of Freebird. The system level requirements were set to provide a framework for the design of the Freebird system. The requirements are subdivided into four top level system requirements. The system level requirements state that the vehicle must:

1. Support the required number of humans and any cargo for the duration of the mission.
    - Be capable of supporting a crew of 3 for a 4 day mission.
    - Be capable of supporting EVA's.
    - Be capable of supporting a crew of 8 for a 24 hour mission.
2. Be capable of reaching its destination and returning safely.
    - Be capable of carrying 500 kg of cargo to a polar inclination at an altitude of 210 km.
    - Be capable of carrying 500 kg of cargo to an equatorial inclination at an altitude of 36 000 km.
    - Be capable of carrying the crew module within LEO between inclinations of 28.5° and 74.7°.
    - Be controllable remotely.
    - Be controllable by on board astronauts when the crew module is attached.
    - Be capable of retrieving astronauts from another platform in LEO within 24 hours.
3. Safely interact with its environment.
    - Safely maneuver around, rendezvous, and dock with ISSA, Space Shuttle, Soyuz and satellites.
    - Survive radiation, debris, atmospheric, and thermal environments between altitudes of 210 km and 36 000 km.
    - Be capable of communicating with ISSA, Space Shuttle, Soyuz, and ground stations
4. Be reusable.
    - Have a vehicle lifetime equal to that of ISSA.
    - Be ready for launch by the time ISSA is operational.

Vehicle Concept

After assessing various design concept options, trades were made. The system level trades of modularity, expandablilty, propulsion system, and launch vehicle were considered in the process of concept selection. A semimodular vehicle which consists of a crew module, smartbox, propulsion pack, extra fuel tanks, and the VSF (Vehicle Storage Facility) for storage purposes was finally designed (see Figure 1).

The crew module is a shielded pressure vessel that contains all necessary life support equipment and human interfaces. It has sufficient space for 3 people to move comfortably about, as shown on Figure 2. The "smartbox" is a mechanical link between the crew module and the propulsion pack and contains the power subsystems. It

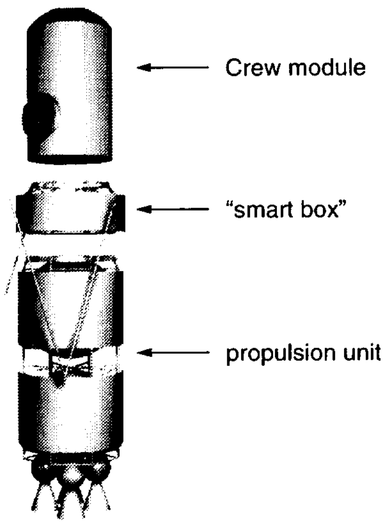

FIGURE 1. Freebird concept.

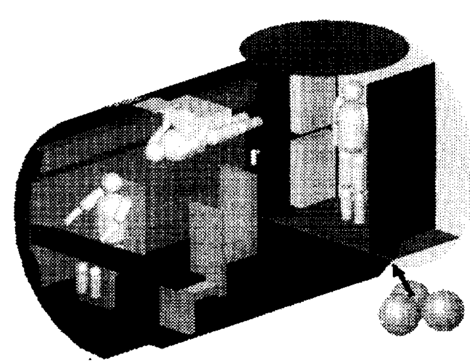

FIGURE 2. Cabin layout of Freebird.

also provides a mounting point for the antennas and two remote manipulators. The propulsion pack consists of cryogenic hydrogen and oxygen fuel tanks and four 92.5 kN vectored thrust engines. Extra fuel tanks can be attached to the propulsion pack to provide the vehicle with a greater range. The VSF is a simple structure near the space station where the vehicle (or sections of the vehicle) will be stored when not in use. The purpose of the VSF is to extend the life of the vehicle by reducing micrometeorite and radiation damage to the primary structure while reducing the shielding weight on the vehicle itself. Titan IV was chosen as the launch vehicle for Freebird after considering cost, mass, and geometrical constraints, with later fuel launches to be conducted with the less expensive Model C Proton.

The vehicle was designed to be assembled for each mission according to the specific scenario to be accomplished. For example, for a crewed mission to LEO, a crew module, which includes all the necessary life support equipment and human interfaces, will be mated to the base unit. If extra fuel tanks are required for extended missions, they are mated to the propulsion pack.

Operation

Launch

The vehicle will be launched in two separate sections: the crew module in a Proton C and the base in a Titan IV. The base will not be fully fueled since the total wet mass of the fully fueled vehicle exceeds the maximum lifting weight of the Titan IV. The fuel required for the Freebird will be launched in the Proton C. The Proton C launch vehicle was chosen primarily because of its large lift capacity and relatively low cost per launch.

Ground Support

All missions will be controlled and monitored from the Johnson Space Center (JSC), where the combined Space Shuttle and ISSA control center is located. Although this is the primary control center, there might be teleoperated missions where direct control by JSC would not be desirable due to the relatively long lag time that is required for signals to travel from Freebird to JSC and back. In this case, a temporary control console would be established under the geostationary asset.

Maintenance

Preventive maintenance for each module will be performed on orbit at regularly scheduled intervals. The most consuming and most costly maintenance activity will be the replacement of the propulsion pack after every 100 engine starts. With a nominal number of 8 missions per year, this activity will be required approximately once every three years. Other maintenance includes the visual inspection of the structural interfaces, diagnostics of the onboard computer and replacement of the EMU batteries. The crew module also has its own tool box in which are carried spare parts such as spare batteries, extra replenishables for the crew module, and an interface repair kit.

Storage

While the Freebird is not in use it will be stored in the VSF with an adequate amount of fuel to perform a rescue. The VSF will provide radiation and micrometeorite shielding to extend the lifetime of the vehicle to 10 years. It will also provide power necessary for the operation of system such as the thermal venting system, the computer (which monitors all vehicle systems), and the thermal unit (which will maintain critical instruments at an acceptable temperature).

Cost and Producibility

The design of Freebird has implemented cost minimization practices while maximizing efficiency and reliability. The total cost of Freebird delivered to the ISSA is estimated to be $1.5 billion. This figure reflects a 36 to 40 month period of design finalization, testing, and manufacturing. A combination of top down and bottom up estimation was used. The primary cost model used is an Orbital Transfer Vehicle (OTV) cost model provided by Lockheed. The total yearly prorated operating cost of the vehicle is estimated at $834 million.

Limitations

At this step in the design process, Freebird has inherent limitations to vehicle and mission performance. Safety considerations, the available technology, the space environment, and orbital dynamics are among the issues that impose restrictions on the design. Vehicle restrictions and a range of missions which is more narrow than that implied by the primary mission objectives are the result of these issues.

Clearly, various missions cannot be accomplished within the operating envelope established for the primary mission objectives. The destination location, rescue mission considerations, the life support system design, and the potential for radiation damage act to restrict the range of missions that Freebird is anticipated to accomplish. The destination location affects the Δv required, which in turn affects the fuel required and the types of missions that can be completed. Particularly, extended missions, missions to retrograde orbits, and equatorial missions in low earth orbit are generally not feasible. Rescue mission considerations, such as mass ratio considerations and availability of the vehicle, restrict the spatial and temporal range that rescue missions may cover. Missions which are crewed have additional implications. The life support system design limit of sustaining 3 astronauts for 4 days in a non-rescue situation simply restricts the possible duration of crewed missions. The potential for radiation damage to an astronaut performing an EVA on repair missions is another issue which prohibits Freebird from EVA operations in geosynchronous orbit and at polar inclinations in low earth orbit.

Those restrictions which apply to the vehicle itself include the amount of payload that Freebird can carry, the protection that Freebird requires from radiation and micrometeorite debris when not on a mission, and the system lifetime. The amount of payload that Freebird can carry is obviously limited by the altitude and inclination of the desired orbit. The protection of the vehicle, the VSF (Vehicle Storage Facility) is a significant element of the system which requires further design detail and development. The system was designed for a 10-year survival time--a length of time which would drop to 2 years without the protection of the VSF.

The next level of design iterations should be performed to minimize or overcome limitations. For many of the limitations discussed above, alternative options exist but have other implications which complicate the design. The future growth of Freebird, however, depends upon modifying or extending the current design to preclude these limitations without sacrificing efficiency and effectiveness in performance.

Part I: Pages 9-22
Mission Definition

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Chapter 1

Mission Prioritization

Freebird is a space based, reusable orbital transfer vehicle based at International Space Station Alpha (ISSA). It can be crewed, but can also be teleoperable and thus capable of completing missions autonomously. A vehicle like Freebird greatly increases the capability of the space program for both the United States and the world. There are a great many missions that such a vehicle could fulfill. In order to properly determine what specific capabilities Freebird should be designed for, the various needs for a space based orbital transfer vehicle were examined. Once the particular needs were evaluated, the mission objectives for Freebird were established and the vehicle was designed to fulfill them.

1.1 Need Identification

A vehicle based at ISSA could be designed for a variety of different missions. The following missions were considered for Freebird.

1.1.1 Tending Existing Assets

Currently, there are assets in Low Earth Orbit (LEO) at altitudes between 200 km and 1000 km. There are also assets as far away as geosynchronous orbit (GEO) at an altitude of 36 000 km. These assets include many satellites launched by various nations to perform a variety of roles, such as communications, observation, and tracking. Figure 1.1 shows a graphic representation of all operational satellites that have been launched since 1984 by all nations. The radius of the orbits is represented in the radial dimension. The units are Earth radii. The inclination of the orbits is represented in the angular direction and is measured in degrees. Each satellite's orbit is represented by a square for its perigee and a circle for its apogee. There is a need to tend these assets.

Tending includes: repair of existing assets, reboosting assets, deorbiting obsolete equipment, and conceivably deploying new assets from ISSA. Many satellites require an upper stage to be carried from their launch vehicle to their target orbit. In some cases, a reusable, space based upper stage is more efficient for this operation than an expended upper stage that must be designed for the constraints of the launch vehicle [1]. A variety of failures can occur throughout the lifetime of a satellite on orbit. A vehicle that can travel to these satellites and

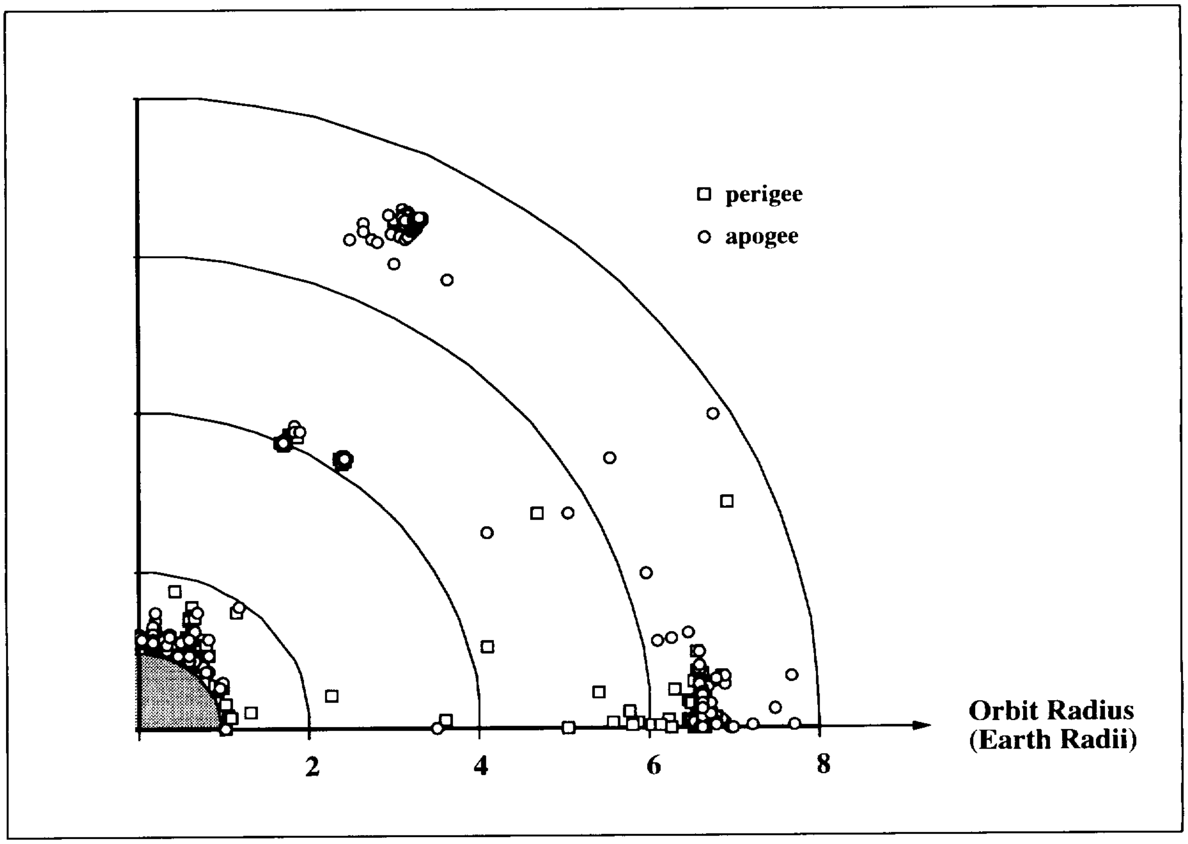

FIGURE 1.1 *Altitude and Inclination of Satellites Launched Since 1984.*

perform repairs could save the space program money by extending the lifetime of equipment on orbit, rather than requiring an entire new system to be built and launched. Once a satellite has become obsolete, it remains in orbit, contributing to the debris that is already causing a problem for orbital operations. Having a vehicle that could deorbit this "junk" would not only help cut down on this problem, but would also free up valuable space in popular orbits so that new satellites could be deployed.

Currently, the space shuttle can tend some of the existing satellites. However, its envelope of operation is confined to LEO and it cannot reach higher inclinations (above approximately 40°) without the mission becoming prohibitively expensive. That leaves a large number of satellites that cannot be reached by today's methods. In particular, there are large groupings of satellites in GEO equitorial and LEO polar orbits that cannot be tended. In addition, the space shuttle is aging and expensive to launch. It is possible that it will be phased out in the near future. Having a newer vehicle that can perform tending missions more cheaply, will help ensure that this vital task can be continued in the future.

1.1.2 Exploration

The vehicle could be designed with the capability to reach Lagrangian points or to orbit the moon. While there are currently no assets in either of these locations, having a vehicle that can reach them is the logical first step to making use of them.

1.1.3 Rescue and Retrieval

Rescue of stranded astronauts from other space platforms is another possible mission. The existence of a space station and a vehicle that can reach the orbits of other crewed spacecraft from it provides an additional level of safety for crewed missions in space. Should the Shuttle or Soyuz fail for some reason, the crew could be transferred to the Space Station to be returned to Earth later. In addition to this, Freebird could conceivably assist in the proximity of ISSA by retrieving errant objects and equipment. It could also assist with modifications and repair of ISSA.

1.2 Mission Prioritization

The difficulty and expected interest in the possible missions allows them to be prioritized. Some of these missions could be performed by an uncrewed vehicle in either a remote or autonomous configuration. Others, such as complex repair missions would be best performed with astronauts on site.

1.2.1 Mission Complexity

Sending a vehicle on a round trip to GEO can be a very difficult task. There are a variety of reasons for this. First of all, the environment at GEO is more detrimental than at LEO (see Section 1.4.2 on page 29). In addition, communication, navigation, and tracking are more difficult in GEO than at LEO.

LEO missions are comparatively easy to design for, as they are where experience is greatest. In addition, they tend to use less fuel than missions to GEO. Polar orbits however, require the most fuel of any orbital mission considered.

Building a vehicle that could reach Lagrangian points or the moon would be at least as difficult as GEO. The missions would require more fuel, higher vehicle complexity, and infrastructure for navigation, communication, and tracking.

Providing rescue for astronauts who become stranded does provide some complications to the design. Construction of a system that is capable of rescuing a crew of astronauts from every crewed platform, for any conceivable failure, at all times, would put serious limitations on its capabilities. It requires the vehicle to be able to perform a variety of missions with little or no advance warning. It requires the vehicle to fulfill these missions with little or no preparation time. Finally, and most important, a dedicated rescue vehicle must be available for rescue at all times. This means that the system would not be able to perform any other mission during the times that it was standing by for a rescue.

1.2.2 Cost of Missions

Examination of the total cost of satellites launched between 1986 and 1990 reveals that a significant asset investment exists in both LEO and GEO orbits. A breakdown of satellite system costs by location in FY90 dollars is shown in Figure 1.2 (LEO) and Figure 1.3 (GEO). The horizontal axis on these graphs is the cost in millions of dollars. The vertical axes represent the number satellites of that value in each location. The mean value for LEO assets is approximately \$100 million and for GEO, approximately \$300 million. These values include the satellite construction cost, launch cost, and insurance costs. It is perceived that satellite owners would be willing to spend at least 50%, and perhaps as much as 70%, of these costs in the repair of these assets. The risks of launching a new system coupled with the loss in revenue to the satellite owners is a powerful incentive to repair existing assets. Based on this operational income to the Freebird project, various mission scenarios and constraints can be calculated.

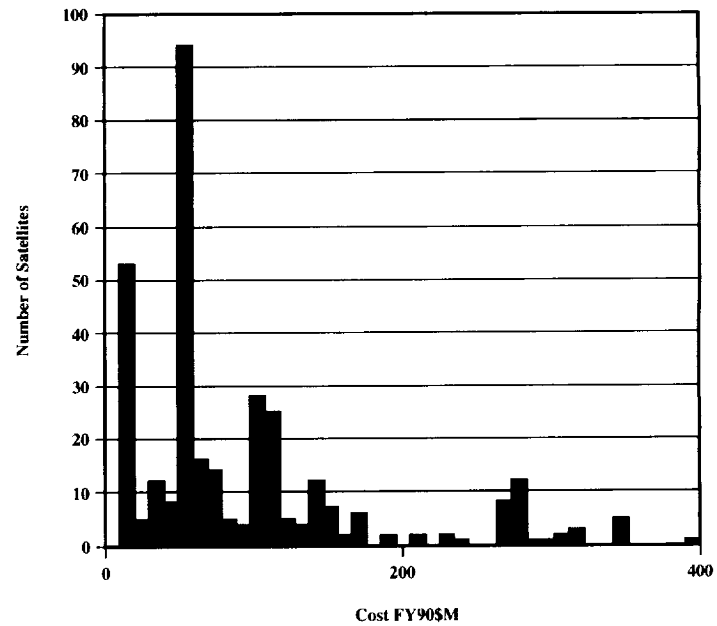

FIGURE 1.2 *Histogram of Satellite Costs in Low Earth Orbit.*

1.2.3 Selected Mission Prioritization

After considering the various trades, Freebird's missions were chosen and prioritized as follows.

1. Tending assets in LEO
2. Tending assets in GEO
3. Rescue astronauts from other crewed platforms

Tending assets in LEO is the vehicle's most important mission. Next is tending assets in GEO. While the desire for GEO missions is greater, the added complexity caused them to be placed slightly lower than those in LEO in importance. Following this was the capability to rescue astronauts. This was considered less important because a rescue mission is very different to design for than the tending missions. In addition, a system that spends large portions of its time performing tending missions away from the space station simply can not operate as a dedicated rescue vehicle. Missions beyond GEO to Lagrangian points or the moon were not designed for because there are currently no assets in these locations. Further, politics suggests that there is not going to be any demand for increased traffic in these regions in the foreseeable future.

1.3 Mission Objectives

The prioritized needs led to two primary and two secondary mission objectives. These missions define the vehicle design.

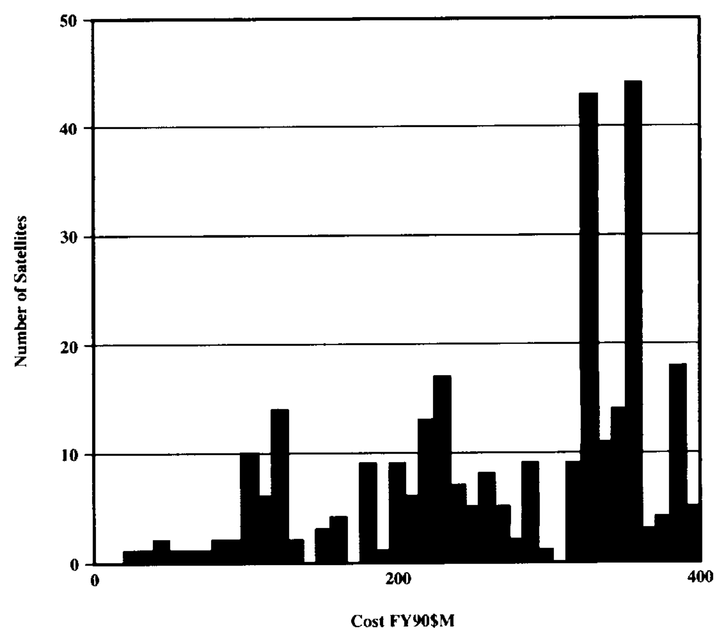

FIGURE 1.3 *Histogram of Satellite Costs in Geosynchronous Orbit.*

1.3.1 Primary Mission Objectives

The primary objectives are those for which the vehicle was specifically designed. They initially establish what the system capabilities must be. The primary objectives for Freebird are

1. The vehicle is capable of tending assets in LEO, between inclinations of 13.2° and 90°.
2. The vehicle is capable of tending assets in GEO, at equatorial inclinations.

Tending includes repair, reboosting, and deorbiting. It does not include deployment. This is because it is not fuel efficient to launch assets to an inclination of 51.6° and then boost them to GEO at equitorial inclinations. In addition to this, using Freebird to deploy assets will lessen its availability for repair and deorbit. Deployment is a mission already adequately performed by existing vehicles, whereas repair and deborbiting are not.

In some cases, especially repair missions, these objectives can be fulfilled more efficiently with humans on board. As a result, the vehicle will be designed to carry humans. However, it is unfeasible to send humans to GEO or polar orbits, as they can not perform ExtraVehicular Activities (EVA's) there (see Part II: Section 9.8.4 on page 187). Thus, the vehicle is capable of operating without astronauts on board as well.

1.3.2 Secondary Mission Objectives

Once the primary objectives were established, the following secondary objectives were added to the vehicle's capabilities.

1. The vehicle can rescue stranded crews from crewed orbital missions.
2. The vehicle can ferry cargoes between orbits.

They were included because their desirability outweighs the increase in complexity and cost of the design. The ability to rescue does impose significant added complexity to the design (see Section 1.2.1 on page 11). Launching a vehicle that does not have this capability does not make sense. If it were required, even once, during the lifetime of the system, it would be well worth the effort that went into the design. However, due to the constraints imposed by the primary objectives, Freebird's rescue capabilities will be limited in scope. It will not be a dedicated rescue platform, but rather, will be capable of performing an evacuation of either the shuttle, Soyuz within 24 hours of a declared emergency. In addition, it will not be capable of performing rescues while it is fulfilling its primary mission objectives. Any astronauts rescued will be brought back to ISSA for later return to earth.

Requiring the vehicle to be capable of carrying cargo between orbits adds little to the complexity of the vehicle. This is because tending assets already requires the vehicle to carry cargo between orbits.

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Chapter 2

Mission Scenarios

The established mission objectives allow for the definition of specific mission scenarios. These scenarios define the missions that Freebird will be able to perform and how it will perform them.

The concept for the Freebird design is detailed in Section 2.3.2 on page 44. Freebird is a semi-modular vehicle which consists of a crew module, a "smartbox" (which contains the electronics and fuel cells), and a detachable propulsion pack. Figure 2.5 on page 44 details these modules.

Freebird is capable of three basic missions. These include crewed and uncrewed missions to LEO, uncrewed missions to GEO, and rescue missions. In addition, scenarios for storing the vehicle at ISSA and resupplying it at the end of each mission are presented.

2.1 Tending Assets in LEO

Freebird's chief area of operation will be LEO (Low Earth Orbit) which is defined to be between the altitudes of 200 km and 1000 km and between the inclinations of 28.5° and 90°. This area includes a majority of the assets in LEO, as seen in Figure 1.1 on page 10. Freebird's primary function is to tend these assets.

2.1.1 Uncrewed LEO Missions

An uncrewed repair of a satellite in LEO is the primary mission for which Freebird is designed. In this mode of operation, the crew module will be disconnected from the base unit and left at ISSA (International Space Station Alpha), while the vehicle is deployed. Once on location, it will effect any repairs by teleoperation of its RMS (Remote Manipulation System), and then return back to ISSA for resupply and storage.

Orbital maneuvers will be accomplished via a simple plane change, or if an altitude change is required, by performing a combined plane change and Hohmann transfer. These maneuvers allow for fuel efficiency while still allowing the vehicle to complete the maneuvers in a matter of hours rather than days or months [1]. Once the orbital transfer is complete, the vehicle will rendezvous with the target asset. The vehicle is capable of orbital transfer and rendezvous by autonomous control (with the use of on-board computers) or by teleoperation (by ground based human controllers or computers). Rendezvous is expected to take approximately 6 hours to complete.

Once Freebird has matched orbits with the target satellite, it must grapple and repair it. This will be done by teleoperation of the RMS by ground based human controllers through a video downlink from cameras mounted on the vehicle's superstructure and the RMS. The RMS consists of two arms mounted on the sides of the vehicle which will facilitate the grappling of the satellite with one arm while effecting repairs with the other.

The types of repairs that Freebird will be able to perform in this mode depends on the specific asset to be repaired. For a satellite that was designed to be repaired in the manner described above, Freebird could be used to replace any of a variety of different modules, such as thruster fuel, batteries, or solar arrays. Many satellites, however, will not be repairable in this manner. An alternative option for such satellites is for Freebird to attach a booster to them for return to ISSA. Freebird could also grapple the satellite with a mission specific berthing adaptor and transfer the satellite back to ISSA. Once at ISSA, the satellite can either be repaired or returned to Earth. Another option is to send Freebird in a crewed configuration so that astronauts can perform an EVA (ExtraVehicular Activity) in cooperation with Freebird's RMS, to conduct a more extensive operation. As a last resort, then Freebird can deorbit the satellite by altering its orbit (either with a booster, or by grappling it) so that it decays into the atmosphere.

The length of time that Freebird can spend on location in this mode is determined by the amount of RCS (Reaction Control System) fuel and the amount of fuel for the electrical power system that is carried on board. Freebird can conceivably spend several weeks at the target satellite, effecting repairs 24 hours a day (with the ground controllers operating in shifts), before it needs to return for resupply and storage.

2.1.2 Crewed Configuration

Freebird can operate with a crew of 3 astronauts in LEO in inclinations between 28.5° and 74.7°. The astronauts can be sustained for missions up to 4 days in length. Calculations show that the orbital transfer will take two hours each way, that it will take 6 hours to rendezvous with the satellite, and 6 hours to dock with ISSA. This allows the astronauts to spend 3 days on location performing repairs, with a contingency. The vehicle will be resupplied remotely, once the astronauts have been returned to ISSA.

The capabilities of the vehicle in the crewed configuration are the same as in the uncrewed configuration. In addition, with astronauts on site, more complex repairs can be performed. The astronauts can work in conjunction with the RMS to capture and repair the satellite. In the crewed configuration, the RMS will be operated by an on board astronaut while the other two astronauts perform EVA. Astronauts, however, cannot perform EVA's at higher inclinations due to radiation (see Section 9.8.4 on page 187). Thus, the crew module is not required above inclinations of 75°.

Astronauts will embark from ISSA for Freebird missions and return to ISSA upon completion. Because taking astronauts from ISSA will significantly drain its resources, crewed missions will be scheduled to coincide with Shuttle resupply missions. The Shuttle is scheduled to arrive for resupply 4 to 5 times a year. At these times, there will be 11 astronauts available (as opposed to the 6 normally present). Thus, Freebird can perform up to 5 crewed missions every year.

2.2 Expandability

Without crew or cargo, Freebird weighs 5000 kg and can carry 20 000 kg of fuel. Thus, it has a MF (Mass Fraction) of approximately 5 which allows it to travel between inclinations of 28° and 75° (see Section 1.4.1 on page 29). With the crew module, the vehicle weighs 8000 kg which lowers the MF to 3.5 and allows up to 17° of inclination change within LEO.

To travel beyond these limits or to carry additional cargo, many strategies may be employed. One possible strategy is the addition of 2 external fuel tanks (launched by Proton) which each hold 18 000 kg of fuel and thereby increase the MF to approximately 11 without the crew module and up to 6.6 with it. In this case, the base unit would rendezvous and dock with the tanks prior to departing for a mission. This easily allows a trip to 90° with only the base unit and allows the vehicle to perform the required 23.1° of plane change with the crew module attached.

Another, more fuel efficient, strategy for extending Freebird's capabilities is to launch the fuel necessary for the return trip to the Freebird's initial destination prior to Freebird's departure from ISSA, carry enough fuel aboard Freebird for a one-way trip, and refuel for the return trip on orbit. The rocket equation specifies

$$\frac{M_i}{M_f} = e^{\frac{\Delta V}{Isp \bullet g}} \qquad \textbf{(Equation 2.1)}$$

where $\frac{M_i}{M_f}$ is the MF, ΔV is the necessary change in velocity, Isp is the specific impulse of the fuel and g is the acceleration due to the earth's gravity at sea level [1]. This equation states that doubling ΔV squares the MF. Therefore, if Freebird only carries enough fuel for a one way trip, then the ΔV required is halved, reducing the MF to the square root of its original value. The fuel that is saved with strategy can be quantified by the expression

$$F_L^* = 2\left[(F_C^* + 1)^{\frac{1}{2}} - 1\right] \qquad \textbf{(Equation 2.2)}$$

where F_L^* is the total fuel per kilogram of vehicle weight that is required when the fuel for the return trip is received at the destination and F_C^* is the fuel per kilogram of vehicle weight that is required for a round trip without refueling. This is not an immediately intuitive expression. Consider the example of a transit to polar orbit of the base unit, which has a mass of 5000 kg. The MF for one way trip to polar at 390 km is 3.139, which means that the fuel required for the mission without refueling during the mission is 44 255 kg (requiring 3 Proton launches to the station). On the other hand, by launching fuel to polar orbit for the return trip means that only 21386kg of fuel is required (which can easily be done in 2 Proton launches, one to polar and one to ISSA). The logistics of refueling should be no more complex anywhere else than they are at ISSA, as Freebird refuels autonomous of the station in any case. In addition, rendezvous with the fuel tank will be no more difficult than rendezvous with a satellite in the same orbit. For these reasons, this is the preferred option for increasing Freebird's range.

The external fuel tanks will be the same design as the propulsion module, without the main engines. The fuel tanks may have a minimal RCS system to allow them to maintain attitude while Freebird is performing a rendezvous. Once the fuel has been used, the tanks will be deorbited.

2.3 Tending Missions in GEO

Freebird's base unit can only reach GEO by expanding its fuel supply as discussed in Section 2.2. Because launching 10 000 kg of fuel to GEO for a return trip is prohibitively expensive (only the Titan IV and Energia are currently capable of lifting this much mass to GEO), the external tanks will be used. Freebird will use a combined Hohmann transfer and plane change to reach the target asset, placing it in GEO in approximately 6 hours. An additional 6 hours will be required for rendezvous. Freebird will be limited to uncrewed operations in GEO. As in LEO (see Section 2.1.1 on page 17), it can effect repairs with the RMS by teleoperation and can deorbit irreparable satellites. Freebird will return to ISSA with another combined Hohmann transfer and plane change.

2.4 Rescue Missions

As mentioned in Section 1.3 on page 12, Freebird is not a dedicated rescue vehicle. Instead, it is capable of performing a limited role as a rescue craft when it is not being used for its primary tending operations. To facilitate this, Freebird will be maintained in a fueled, stand-by state, from which it can be powered and operational within 2 hours. Freebird could be used to rescue astronauts from either the Space Shuttle or Soyuz within 24 hours of a declared emergency if they were close enough to the station (within 17° of inclination). Astronauts rescued by Freebird will be returned to ISSA for later return to Earth.

As the Shuttle normally operates at an inclination of 28.5°, it is possible that the rescue of Shuttle astronauts will not be an option. Freebird's base unit, however, could be deployed with a cargo of food, air, power, and other needed supplies to extend the Shuttle's stay time on orbit. Once this has been done, a more permanent solution could be implemented, depending on the nature of the emergency: either enough fuel could be launched to orbit to allow Freebird to rescue the astronauts, another shuttle could be launched to rescue the crew, or the crew could effect repairs using the extra supplies brought by Freebird. If the astronauts need to evacuate the Shuttle for some reason, Freebird could instead transport up to 2 Soyuz ACRV's (Assured Crew Rescue Vehicle) to the Shuttle which could be used to board the Shuttle astronauts and return them to the ground. Another option that may be feasible in some situations is to maneuver Freebird to the Shuttle with its crew module. The astronauts could then board it for use as a safe haven for up to 24 hours. This option is not very useful, however, because Freebird would not have the fuel to carry the astronauts back to ISSA and it is not capable of a reentry. However, if fuel were deliverable in the interim, Freebird could refuel and then carry the astronauts back to ISSA.

Obviously, Freebird cannot save astronauts from any life critical failures that require an extremely rapid response, such as a hull breach. However, it can be relied on to alleviate many mission critical and certain life critical failures, such as Soyuz launching too far from ISSA to rendezvous, or the Shuttle doors being uncloseable for reentry.

2.5 Storage

Because Freebird will be based at ISSA, there must be adequate facilities to store and maintain it while it is not in use. Protecting a vehicle against debris impacts for 10 years would require a prohibitively massive structure (see Section 4.4 on page 279). To alleviate this problem, the VSF (Vehicle Storage Facility) was conceptualized. The VSF is not part of the Freebird design, but it is a necessary piece of infrastructure in order for Freebird to operate successfully for its specified vehicle lifetime.

The VSF is a structural enclosure that is attached to ISSA which contains the base unit and crew module when they are not being used to protect them from micrometeorite damage. In addition, it has a passive thermal control system.which is required because Freebird's own thermal control system will not be sufficient while enclosed within the VSF. The VSF must allow Freebird to enter, dock, and exit as easily as possible as well as allow the crew to ingress and egress from the crew module. Finally, because Freebird will require a small amount of power, the VSF must be able to support this. The VSF could draw this power from ISSA or from solar panels.

2.6 Servicing

Freebird will need certain supplies after every mission it completes and must be maintained in a regular fashion to keep it operating normally.

2.6.1 Resupply

To be ready to perform its rescue mission, Freebird must be replenished at the conclusion of each of its missions. It will need up to 20 000 kg of main propulsion fuel in addition to RCS fuel and fuel for the fuel cells. If the crew module was used, air, food, and water will have to be replenished as well. The various fuels will be launched to a coplanar orbit with ISSA by Proton. Proton can launch 15 000 kg of main propulsion fuel as well as up to 800 kg of RCS fuel and 200 kg of fuel for Freebird's electrical power system. These fuels will all be contained in various cells within a common fuel tank. Upon return from a mission, Freebird will rendezvous with the fuel tank and grapple it. The fuel tank will interface with the vehicle and fuel will be pumped aboard. Because the fuel will be transferred to Freebird, it will not be in the launch tank for a long period of time which means that only minimal insulation is required on the tanks and helps maximize the amount of fuel that can be carried in a single launch. Refueling will be performed via teleoperation, so that it does not require valuable astronaut EVA time. The vehicle's capability to refuel remotely is what allows it to refuel at its destination as well as at ISSA. Once a launch tank has been emptied, Freebird will deorbit it.

Expendables for the crew module will be drawn from ISSA. Food, air, and water, as well as the astronauts themselves, will be brought to the station by the Shuttle. From ISSA, they will be loaded to the crew module for the mission.

2.6.2 Maintenance

To ensure that Freebird has a vehicle life of 10 years, it will have to be maintained on orbit. In particular, the engines selected (see Section 5.2.4 on page 91), will need replaced approximately every 3 years. To facilitate this, the section of the vehicle that contains the main engines as well as the fuel tanks and RCS thrusters is detachable. This allows a new propulsion module to be launched to orbit and put in place with a minimum of effort. In addition to this, the components of the "smart box" are constructed and mounted in modules, which can be easily removed from the exterior of the vehicle by astronauts in EVA. Thus, should one component fail, it can be replaced. A complete preventive maintenance schedule will be developed for Freebird to ensure that nothing fails during a mission (see Chapter 4.2 on page 276).

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Part II: Pages 25-252
Vehicle Design

Chapter 1

Design Approach and Constraints

Beyond the technical considerations, the design of this vehicle was predominantly determined by several factors: the imposed class structure, the amount of time that was available to the class and the knowledge and experience of the students.

1.1 Class Structure and Program breakdown

The students that undertook this project were enrolled in a required single term senior system design class at MIT entitled "Space System Engineering." The purpose of the course is to give students a chance to work on a large group project and to gain some experience in the design process associated with large systems. The faculty associated with the course consists of both a senior and junior instructor in addition to a teaching assistant -- usually a recent graduate student who previously took part in the course. The prerequisites for the class essentially consist of the undergraduate Aerospace Engineering curriculum. This year, the class consisted of forty students.

The class was broken down into nine different groups. The groups consisted of:

1. Systems
2. Structures
3. Power and Propulsion
4. Environmental Control and Life Support Systems (ECLSS)
5. Communication, Command, Control and Telemetry (C^3T)
6. Guidance, Navigation and Control (GNC)
7. Thermal
8. Verification and Validation
9. Cost and Producibility

Most of the groups had the obvious responsibilities associated with their discipline. Systems also included some project management duties. Verification and Validation worked with each of the groups to determine required test plans as well as the design reliability of the vehicle. Cost and Producibility worked with groups to create a working budget for the project and to maintain a semblance of realism with respect to manufacturing and availability concerns.

In addition to the groups above, a Project Steering Committee (PSC) was created to act as a small decision-making body and to direct interaction between various groups. A member of each group (two from Systems) was assigned the responsibility of PSC representative.

1.1.1 Time Constraints

The class met three times a week in two hour sessions and lasted for fourteen weeks, of which ten were spent working on the group project. Most students were also enrolled in a full course complement and attending to job and graduate school concerns.

This ten week period was further constrained by presentations required to take place during class time to promote full attendance.

1.1.2 Student Knowledge and Experience

The analytical and theoretical knowledge brought to this class by the students is significant, however experience with the design of complex systems is limited. As mentioned earlier, the course requirements for "Space Systems Engineering" consist of the entire undergraduate curriculum at MIT. This curriculum is largely theoretical and deals with most disciplines in an isolated manner. In addition to course work, many students brought knowledge from summer jobs and internships to the class.

For situations where the library facilities were insufficient or outdated, contacts were created in industry to help the project. For up to date information and pricing, hardware suppliers were consulted. Through faculty contacts, NASA engineers were consulted regarding specific issues.

1.2 Design Methodology

The design methodology used for this vehicle initially consisted of determining a set of needs for the vehicle to satisfy. A graph of the mission envelope of a spacecraft with varying mass ratios was overlaid on the satellite distribution chart to determine the basic fuel requirements of these missions, as shown in Figure 1.1. The bulk of missions that would be profitable were determined to be low earth orbit (LEO) repair missions in varying inclinations up to a polar orbit (90° inclination) and geosynchronous orbit (GEO) orbits at an inclination of 0°. These two missions were determined to be the design drivers.

A pictorial representation of the design process that shows the iterative nature of the process and corresponds to the description below is shown in Figure 1.2.

The next step was to ascertain the hard requirements that could be derived from these missions. In particular, the ΔVs or mass ratios required for each mission were recognized as an aspect of the mission that would greatly influence vehicle design.

The class then brainstormed, in subsystem groups, individually and collectively, for ideas to satisfy that which were recognized as the difficult problems in the preliminary vehicle design. The ideas grouped themselves into several categories including the level of modularity of the vehicle, the propulsion system to use for various missions and those missions for which the crew was required.

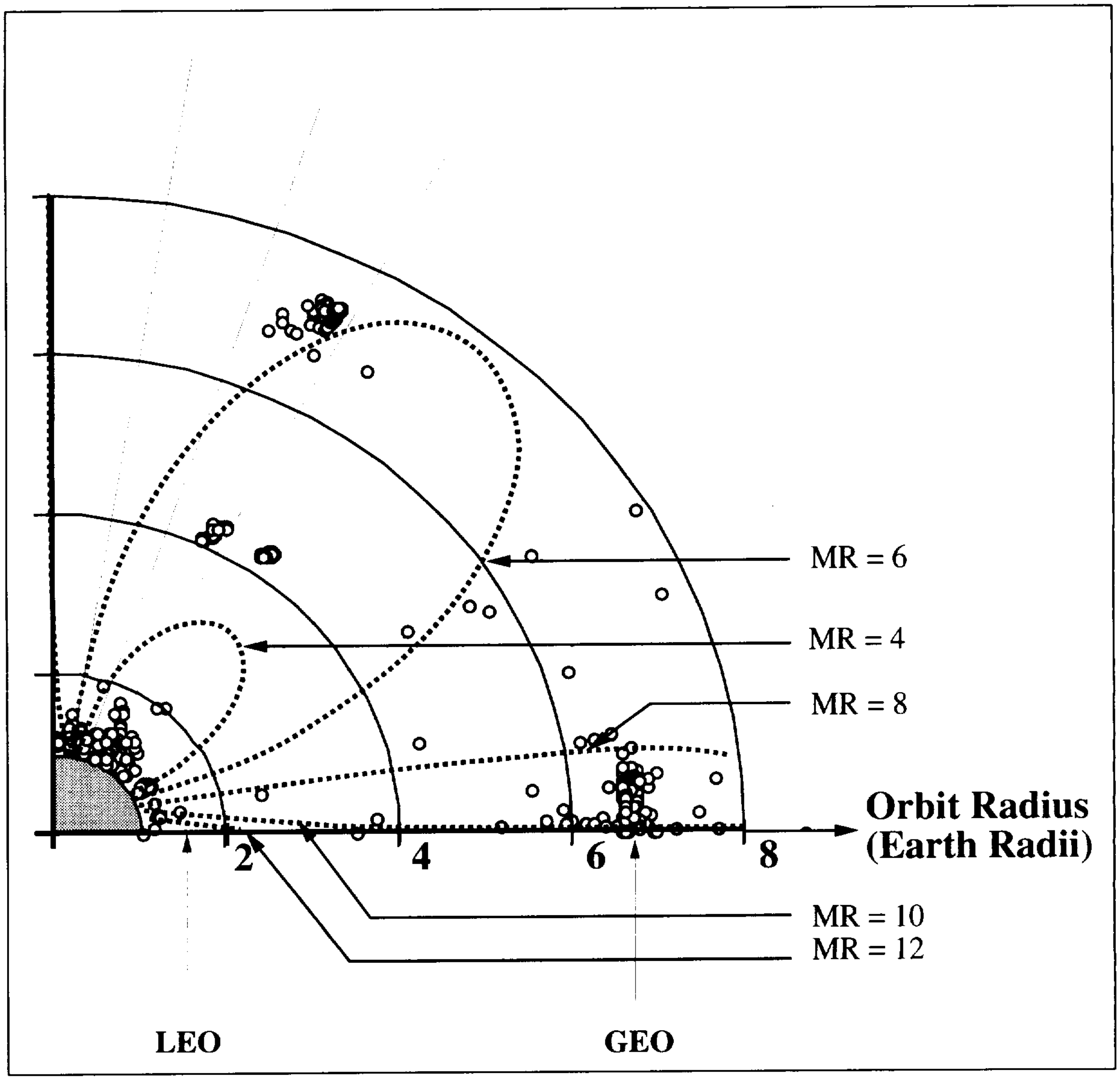

FIGURE 1.1 *Mass Ratio Mission Envelopes.*

The groups most closely associated with each problem then made recommendations based on first level research. The recommendations were based on application of an implicit trade methodology (detailed in the next section).

The Project Steering Committee made many of the major final decisions based on the system recommendations. Decision making consisted of the relevant subsystem group member discussing the issues that were of obvious concern, followed by the other groups determining how this decision would impact their systems. For example, when the Thermal group made a recommendation to use a passive cooling system for the crew module, the Structures group was concerned about the added mass and impact on the debris shielding layers, Power was concerned about the energy savings of passive versus active, and GNC was concerned about whether the vehicle would have additional attitude requirements.

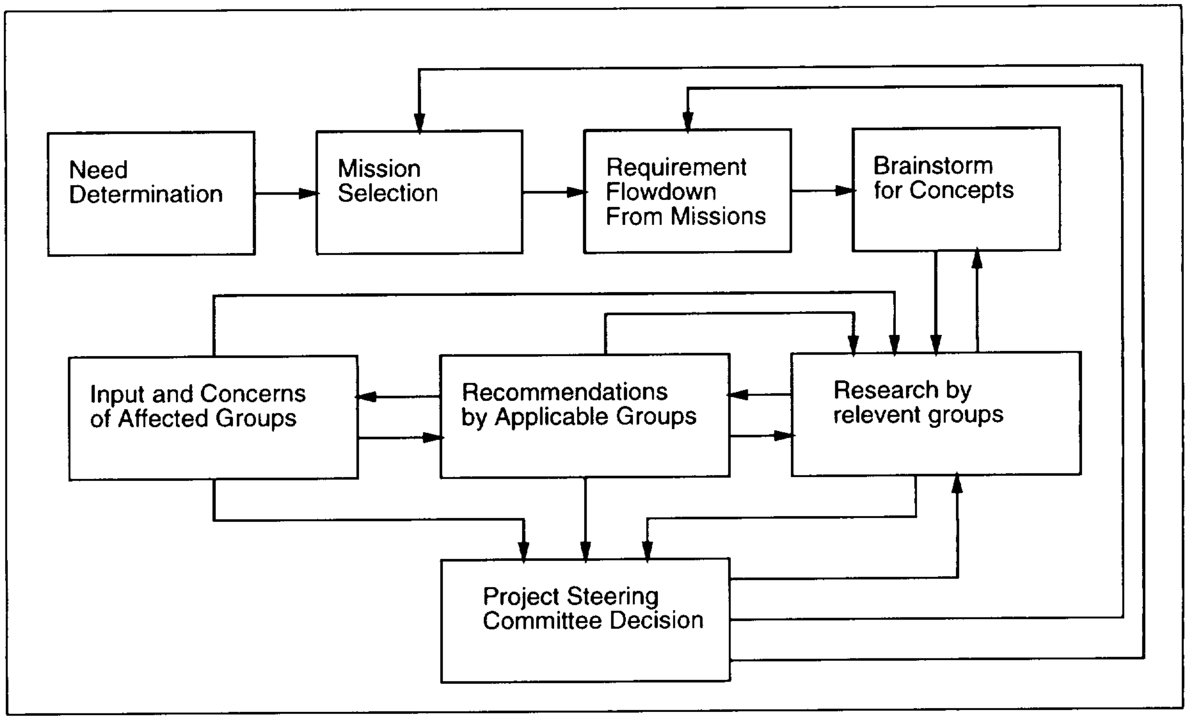

FIGURE 1.2 *Design Process.*

Once the decision was made, it was considered a baseline for design. When circumstances required, major and minor decisions were held up to scrutiny and reevaluated as necessary.

1.3 Trade Methodology

Trades were performed on several criteria. The most important of these dealt with the availability, maturity, price and ultimately the feasibility of the technologies required for a specific approach.

When a recommendation was made to PSC regarding a specific technology for the vehicle, it was usually based on available literature garnered from companies, faculty, and library sources. In each case, the scheduled production and lead time was noted and taken into account.

The maturity consideration of a technology was largely handled by Verification and Validation. In situations where the technology was not considered mature enough to fly "off-the-shelf", a test plan was put together. In a few cases, notably the RL10A - 4 engines, the technology had been used for many years but not in the manner -- extended restarts and burns -- around which Freebird was designed. Test plans were put together in these situations as well.

The Cost and Producibility group worked with other groups to determine if performance goals could be met with less expensive hardware. In many cases, the expense associated with a product was tied into the testing required to validate the part for crewed missions.

Finally, the overall feasibility of a technology was based on a combination of the factors above.

1.4 Design Constraints

The design of Freebird was driven by many factors. The primary design drivers were cost, environment, and operating envelope. The costs associated with Freebird are covered in Chapter 12 on page 227 and Chapter 7 on page 285. The constraints imposed on the vehicle design imposed by the desired operational envelope are presented here.

1.4.1 Fuel Requirements

A summary of necessary ΔV's and Mass Ratios for the various missions is provided in Table 1.1. In this table,

TABLE 1.1 *Summary of ΔV's and Mass Ratios.*

Target Altitude [km]	Target Inclination [°]	ΔV one way [km/s]	Mass Ratio one way	Mass Ratio round trip
36 000	0.0	4.825	2.985	8.908
200	90.0	5.193	3.244	10.523
160	28.5	3.207	2.069	4.279

all ΔV's and Mass Ratios are calculated from an initial position in a circular orbit at 300 km and an inclination of 51.6°, the planned orbit for International Space Station Alpha. The target altitudes and inclinations listed are the limiting cases for the mission scenarios presented. GEO assets are all located at 36 000 km and 0° to a close approximation. The limiting factor for orbital transfer in LEO is inclination change. For these changes, lower altitudes also require more fuel than higher ones. The LEO assets that require the greatest propulsive mass are located at polar inclinations. Assets in altitudes less than 200 km are not of high enough value for it to be worthwhile for Freebird to tend them. This established the limit of Freebird's envelope in this direction at 200 km and 90°. In order to meet rescue needs, Freebird will need to travel to the shuttle's orbit at 28.5° at an altitude as low as 160 km in order to rendezvous with it.

1.4.2 Environmental Issues

The space environment can be largely characterized by six major categories [1]:

1. Vacuum environment
2. Neutral environment
3. Plasma environment
4. Thermal environment
5. Radiation environment
6. Micrometeorite/debris environment

Table 1.2 summarizes the environments and their effects on spacecraft.

The primary design-limiting environments are the thermal, radiation, and micrometeorite/debris environments, and are summarized in the following pages. Of somewhat secondary importance in the initial design phase are the vacuum, neutral, and plasma environments.

Protecting against vacuum effects is difficult, but much of these effects can be mitigated by outgassing necessary components on Earth. Of all the neutral environment effects, atomic oxygen degradation is usually the most serious. Carbon composites and polymers must be adequately shielded, usually with Multi-Layer Insulation (MLI), although visible degradation of Kapton, often used in MLI, has been observed [1]. Interaction

TABLE 1.2 *Space Environments and Effects [1]*

Environment	Effects
Vacuum	UV Degradation Outgassing
Neutral	Aerodynamic Drag Physical Sputtering Atomic Oxygen Reaction Spacecraft Glow
Plasma	Spacecraft Charging Arc Discharging
Thermal	Spacecraft Heating
Radiation	Single Event Upsets Solar Array Degradation
Micrometeorite/Debris	Impact Damage

between a spacecraft and the plasma environment is primarily governed by the spacecraft's magnetic field and surface electric properties. Arcing between surfaces can occur when potentials differ by anywhere from 150 Volts to 500 Volts. Properly grounding a spacecraft considerably reduces arcing problems. Few difficulties have been encountered with vehicles whose power systems operate at a voltage less than approximately 30 Volts.

Thermal Heat Flux Environment

The external environment provides the spacecraft with three major sources of thermal radiation: direct solar flux, the Earth's albedo, and the Earth's infrared emissions. See Figure 1.3

Solar Flux

The solar flux is a measure of the amount of the sun's energy per unit area received at a given distance from the sun. In the case of Freebird, this distance is assumed to be equal to the distance of the Earth from the sun for two reasons. One, since Freebird is orbiting Earth, it is on average at the Earth's distance from the sun. Secondly, the difference between LEO and GEO is insignificant compared to the distance between Earth and sun. The value of the solar flux is called the Solar Constant and has been determined to have an average value of 1358 W/m^2[2].

Earth's Albedo

Approximately one third of the solar flux that reaches the Earth is reflected back into space [2]. This albedo factor is mostly due to the Earth's atmosphere. The amount of albedo energy that reaches Freebird varies inversely with the square of the distance from the center of the Earth.

Earth's IR Radiation

The earth also radiates energy at infrared wavelengths. The amount of radiation reaching Freebird again depends inversely on the square of the distance from the center of the Earth. The flux radiated at the Earth's surface is 273 W/m^2 [2].

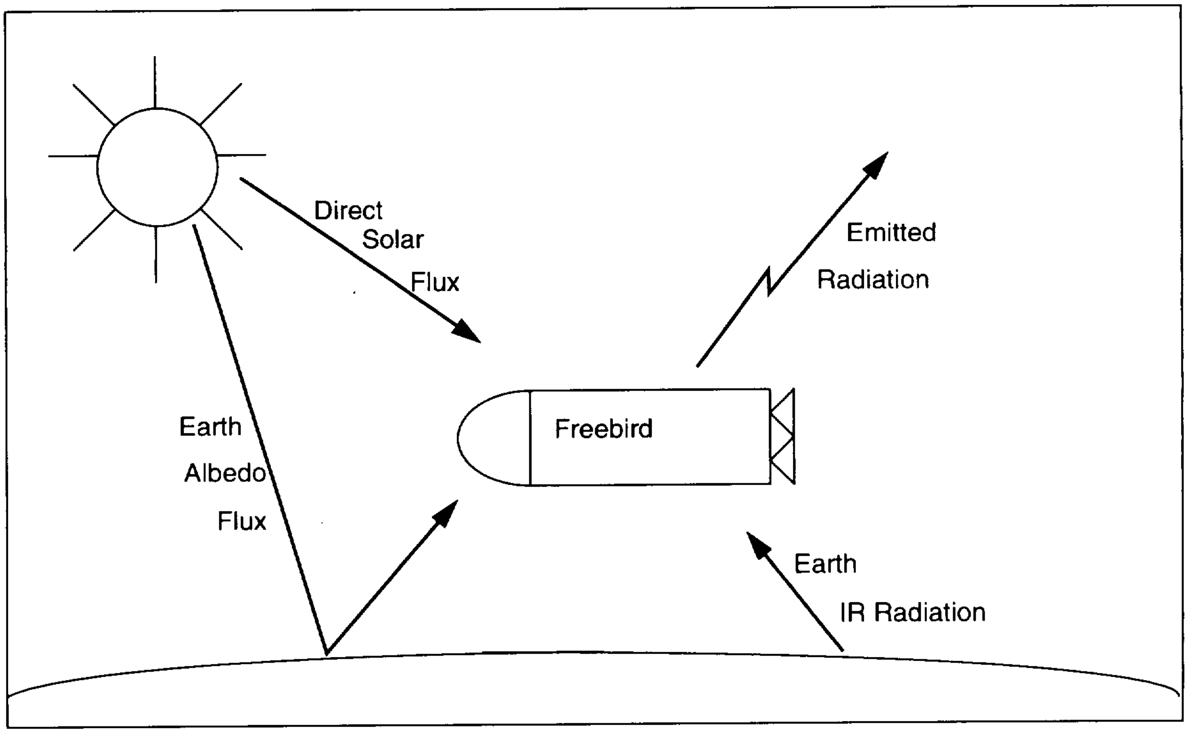

FIGURE 1.3 *External Heat Fluxes Affecting Freebird.[3].*

Radiation Environment

During its missions, Freebird will encounter the harsh space radiation environment, in which there are three primary components of space radiation: trapped particles in the Van Allen Belts, solar flare protons, and galactic cosmic rays (GCR). Figure 1.4 depicts the active space radiation environment, characterizing the flux and energy relationships for the various types of space radiation.

Of the types of space radiation, Van Allen Belt trapped radiation is the most hazardous, due to its proximity to Freebird's anticipated orbital altitudes. The belts are rings of protons (inner belt) and electrons (outer belt) trapped near Earth due to the Earth's magnetic fields. They lie in the region 30° above and below the equator, from approximately 200 km to up to 10 Earth radii. The proton-rich inner belt peaks in intensity near 3 and 5 Earth radii, while the electron-rich outer belt peaks near 2 and 8 Earth radii[3]. Figure 1.5 is a semi-quantitative view of the proton and electron distributions in the inner and outer belts, respectively.

Another type of hazardous space radiation comes from solar flares. Several times per year, the sun emits a flare, or an increase in the flux of high-energy ($1\text{-}10^3$ MeV) particles emitted. These solar flares can last from several hours to several days, and have the potential to damage vehicles in space. Because solar flares are quite unpredictable, vehicles must be prepared to tolerate a sudden increase in high energy particles. Such particles have the ability to penetrate space vehicle outer surfaces, degrading the structural integrity of the surface and/or reaching vulnerable components inside the vehicle, and therefore must be considered when shielding a vehicle from the space radiation environment [5].

The third type of space radiation, Galactic Cosmic Rays, becomes important at higher orbital altitudes, outside of the realm of the Van Allen Belts. They consist of bare, very high-energy ($10^2\text{-}10^{13}$ MeV) nuclei emitted at a relatively low flux (~4 particles/cm^2-sec). While bombardment is not a problem here as it is with solar flares,

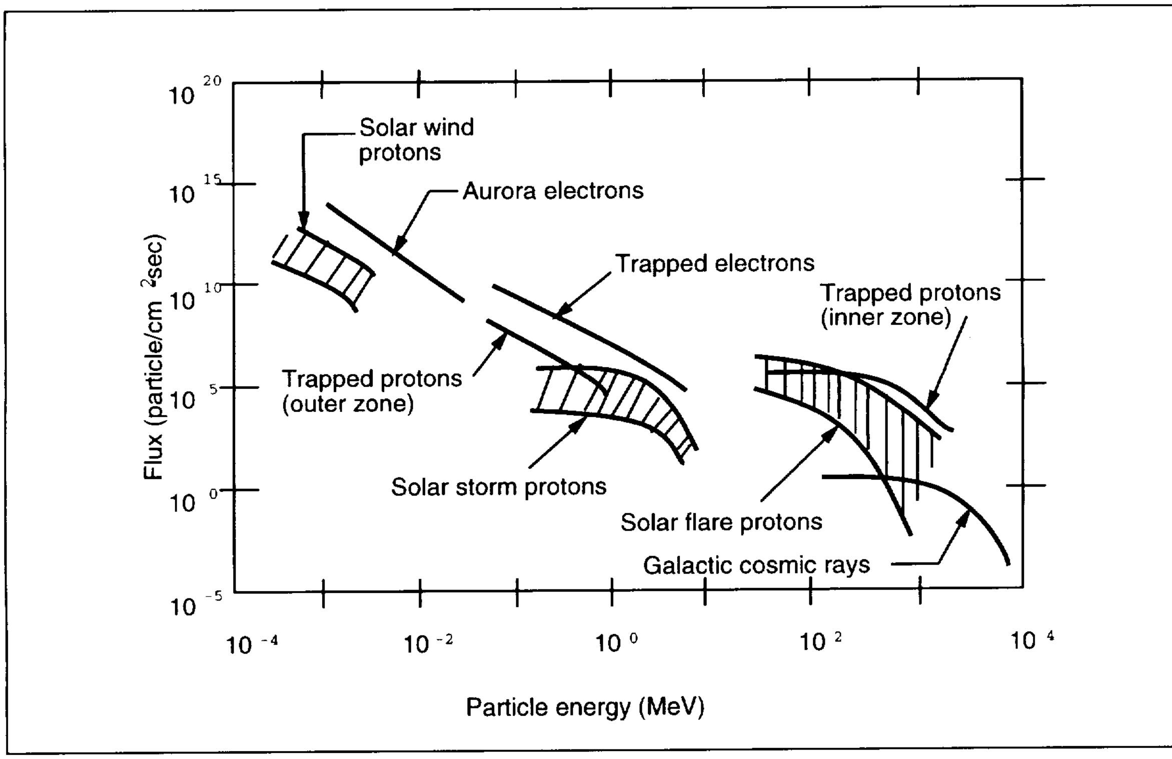

FIGURE 1.4 *The Space Radiation Environment [4].*

GCR's are still a small threat because of the high energy of the particles. A single particle has the potential to damage an electronic component and cause it to malfunction [3]. Although the high-energy particles also pose a threat to other areas of a space vehicle, it is the electronics that are most vulnerable to GCR's.

Vehicles in space must be adequately shielded against all of the types of space radiation they are anticipated to encounter. For all missions, but particularly for deep space and GEO missions, the amount of total radiation encountered depends on the time spent traversing the Van Allen Belts, the current level of solar activity, and the duration of the entire mission. Based on the cyclical nature of solar activity and past data concerning Van Allen Belt radiation and GCR's, it is possible to get a rough approximation of the space environment for a particular mission time frame and therefore an idea of the shielding necessary for that mission.

Based on the space radiation profile detailed above, along with Freebird's mission scenarios presented in Chapter 2 on page 17, it is possible to design a radiation protection scheme for Freebird; the scheme is detailed in Section 3.4.2 on page 70.

Micrometeorite and Debris Environment

The natural micrometeoroid environment has always been a driver for space vehicle structural designs. However, since the orbits have become more populated by satellites, it has also become increasingly populated with larger, man-made debris. Thus, there are two types of micrometeoroid debris that must be protected against: smaller, faster natural debris and larger, slower man-made debris [3]. The smaller (less than 1 cm), natural debris travels at an average relative impact velocity of 20 km/s. The requirements which must be met to withstand impacts from these particles is negligible compared to the required shielding for the man-made debris. Impact from debris greater than 10 cm diameter can be tracked on radar, and thus avoided by maneuvering the spacecraft. Therefore, the micrometeoroid debris that Freebird must be protected against consists of man-made

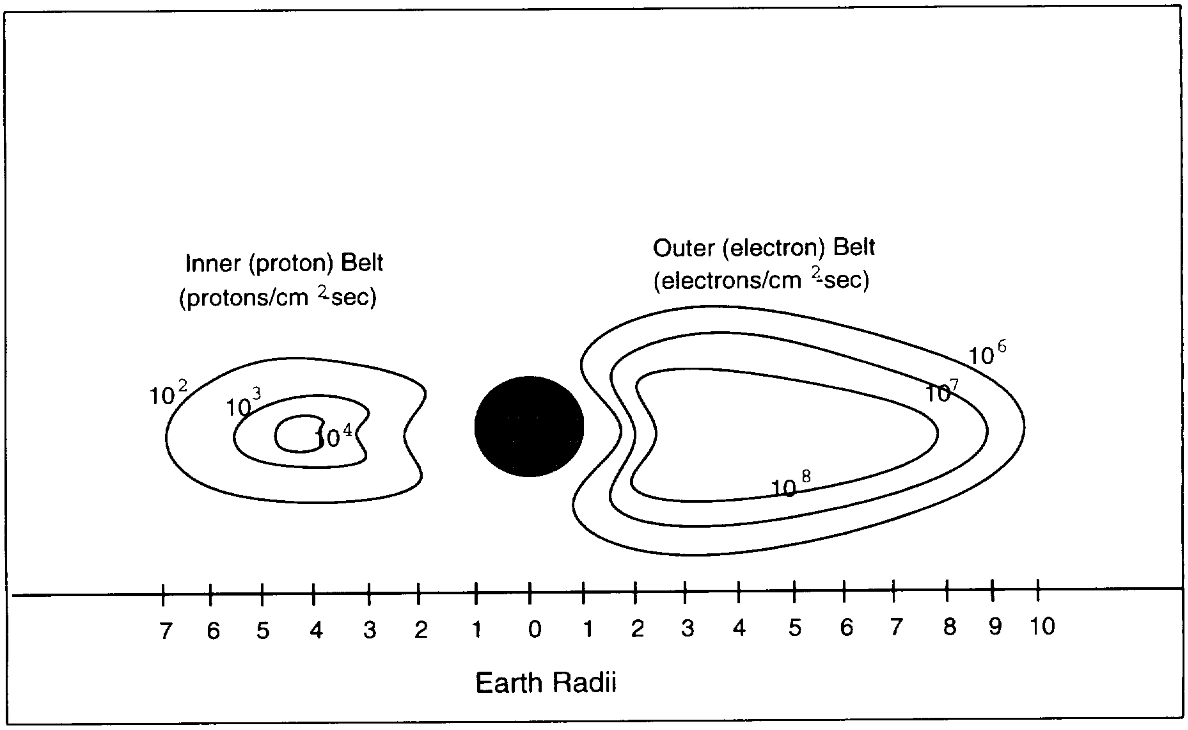

FIGURE 1.5 *The Van Allen Belts [5].*

(usually aluminum) bits between one and ten centimeters in diameter. The average relative velocity of these projectiles is 10 km/s.

Mathematical models have been developed to predict the probability of an impact with micrometeoroid debris and to extract the critical projectile diameter against which to design. Wertz and Larson [3] discuss some of these models in more detail than will be discussed here. Figure 1.6 on page 34 is a reconstruction of a graph showing the distribution of objects at various altitudes [6]. From it an approximate average number of objects was extrapolated. This average is approximately 50 objects, which corresponds to 500 km. According to a model cited in Wertz and Larson, the collision probability of a one centimeter diameter particle with a spacecraft of cross sectional area between 50 and 200 m^2 in a 500 km orbit is 0.001.

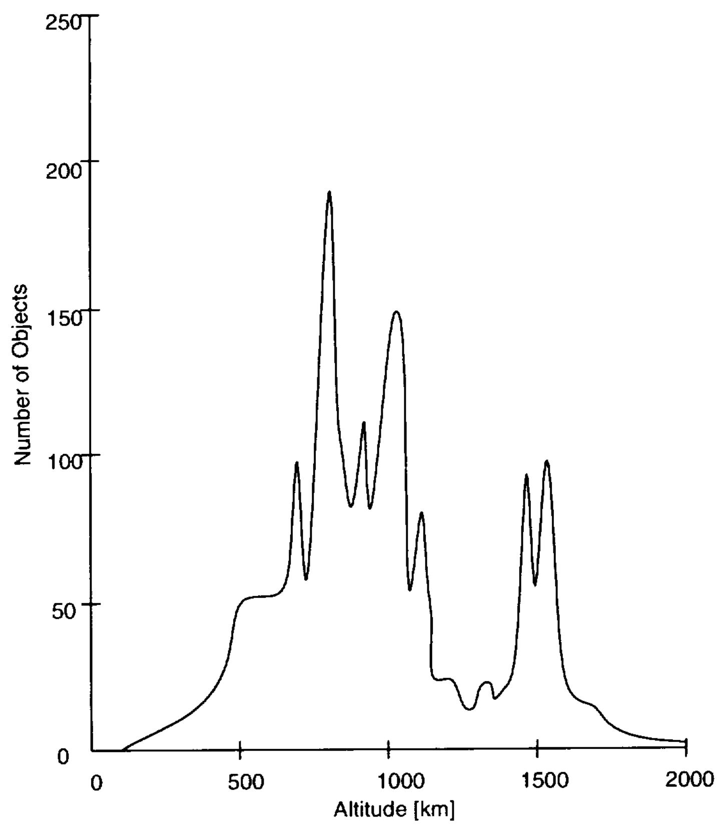

FIGURE 1.6 *Approximate Altitude Distribution of Objects in Low Earth Orbit [6].*

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Chapter 2

Systems Engineering

Systems Engineering was responsible for system wide integration and issues that fell on the borders and outside the domains of other groups. Much of the management, organization, and final document integration for the class was also handled. Systems created templates and guidelines for all groups to follow when documenting their portion of the project. A timeline for the class to follow was also under their purview.

In addition to these management issues, Systems Engineering created and kept track of the overall requirements. This involved due consideration of the various options that were suggested and a thorough analysis of which trade-offs were being made at each stage. The final concept selection was a joint effort of each of the subsystems, via a Project Steering Committee meeting.

2.1 System Level Requirements

The following system level needs were established for Freebird. They flow down from the established primary and secondary mission objectives (see Section 1.3 on page 12).

- The vehicle must support the required number of humans and any and any cargo for the duration of the mission.
- The vehicle must be capable of reaching its destination and returning safely.
- The vehicle must safely interact with its environment.
- The vehicle must be reusable.

2.1.1 Support of Astronauts

The following top level requirements flow down from the first system need. The vehicle must:

1. Be capable of supporting a crew of 3 for a 4 day mission.
2. Be capable of supporting EVA's.
3. Be capable of supporting a crew of 8 for a 24 hour mission.

The longest manned mission the vehicle will perform will be a repair mission. Experience with the shuttle shows that an extensive repair mission can run as much as 3 days in length. It also generally requires 3 astronauts. Adding an additional 24 hours for travel time and contingency, totals to 3 astronauts for 4 days. The reason for having astronauts on site during repair missions is so that they can perform EVA's to assist the robotic systems of the vehicle with the repair. For this reason, the vehicle must be capable of supporting EVA's.

Supporting 8 astronauts for 24 hours is a requirement for rescuing astronauts from another platform. The largest number of astronauts that will need rescued at any one time is currently a full shuttle crew of 7. Provision was made to allow an astronaut from Freebird to be on scene for the rescue, requiring room for an additional person. The 24 hour time allows for a 2 hour power up of Freebird, orbital transfer to and from the shuttle, docking, and 2 hours to transfer the astronauts from the shuttle to Freebird.

Freebird was designed with a removable crew module (See Section 2.3 on page 40). This allows it to be removed during missions in which it is not carrying astronauts, eliminating excess mass and reduces the amount of fuel that Freebird needs to complete its mission.

2.1.2 Operational Envelope

From the second basic need, flow several system requirements. The vehicle must:

1. Be capable of carrying 500 kg of cargo to a polar inclination at an altitude of 210 km.
2. Be capable of carrying 500 kg of cargo to an equatorial inclination at an altitude of 36 000km.
3. Be capable of carrying the crew module within LEO between inclinations of 28.5° and 74.7°.
4. Be controllable remotely.
5. Be controllable by on board astronauts when the crew module is attached.
6. Be capable of retrieving astronauts from another platform in LEO within 24 hours.

The vehicle was designed so that it can travel to a polar orbit with 500 kg of cargo without the crew module. This is the limiting cases of the vehicle's operational envelope, as noted in Section 2.1.2 on page 18. It must also be capable of carrying this cargo to GEO. The rescue mission establishes additional requirements on the vehicle with crew module. It must be capable of rescuing astronauts within 24 hours of an emergency being declared and it must be capable of carrying the crew module to 28.5°. Freebird must be controllable remotely because it will perform many of its missions without a manned crew. In order to make full use of its abilities with a crew, it must be controllable by on board astronauts as well.

2.1.3 Environmental Interactions

The need for the vehicle to safely interact with the environment established the following top level requirements. The vehicle must:

1. Safely maneuver around, rendezvous, and dock with α Station, Space Shuttle, Soyuz and satellites.
2. Survive radiation, debris, atmospheric, and thermal environments between altitudes of 210 and 36 000km.
3. Be capable of communicating with α Station, Space Shuttle, Soyuz, and ground stations

The vehicle must be capable of withstanding all external environmental elements that it will encounter within its operating envelope, as noted in Section 2.1.3 on page 38. In addition, it must be capable of docking with α Station and other space platforms to allow ingress and egress of astronauts and to provide rescue capability. Docking with satellites is required for completion of tending missions.

2.1.4 Reusability

The need for the vehicle to be reusable provides 2 final top level requirements. The vehicle must:

1. Have a vehicle lifetime equal to that of ISSA.
2. Be ready of launch by the time ISSA is operational.

In order to receive the maximum use from Freebird, it is necessary to have it operational for as long as possible. For this reason, there is a requirement that the vehicle be ready for launch by the time ISSA is operational. In addition, it must be operable as long as ISSA. At this time, the date that ISSA is expected to be operational is 2002 and the designed lifetime is ten years.

2.2 System Functionality

In order for Freebird to accomplish the given missions as intended, it must perform certain tasks. These tasks, or functions, are divided into subgroups which define the responsibilities of the subsystems. Table 2.1 on page 39 lists the top level functions of each subsystem, from which the lower level functionality of each is derived.

TABLE 2.1 *Subsystem Top-Level Functions*

Subsystem	Function
Structures, Materials & Mechanisms	Provide Environmental Protection
	Maintain Physical Integrity
	Provide Mechanical Interfaces
Power	Generate Power
	Control & Distribute Power
Propulsion	Generate Thrust
	Manage Fuel/Refueling
Thermal Control	Monitor Temperature
	Supply Heat
	Reject Heat
Guidance, Navigation & Control	Determine Attitude & Position
	Control Attitude & Position
Command, Communication, Control & Telemetry	Computation & Control
	Command & Telemetry
	Communication
Environmental Control & Life Support	Monitor Conditions
	Control Atmosphere
	Provide Sustenance
	Manage Waste

These functions are represented graphically in Figure 2.1 on page 40, along with relevant internal and external interfaces. Mechanical interfaces are depicted with solid lines, communication/data interfaces with dotted lines, and power interfaces with dashed lines. Some lines are broken, indicating an impermanent interface, or one required for only certain kinds of missions. External interfaces are examined in more detail in Chapter 14.

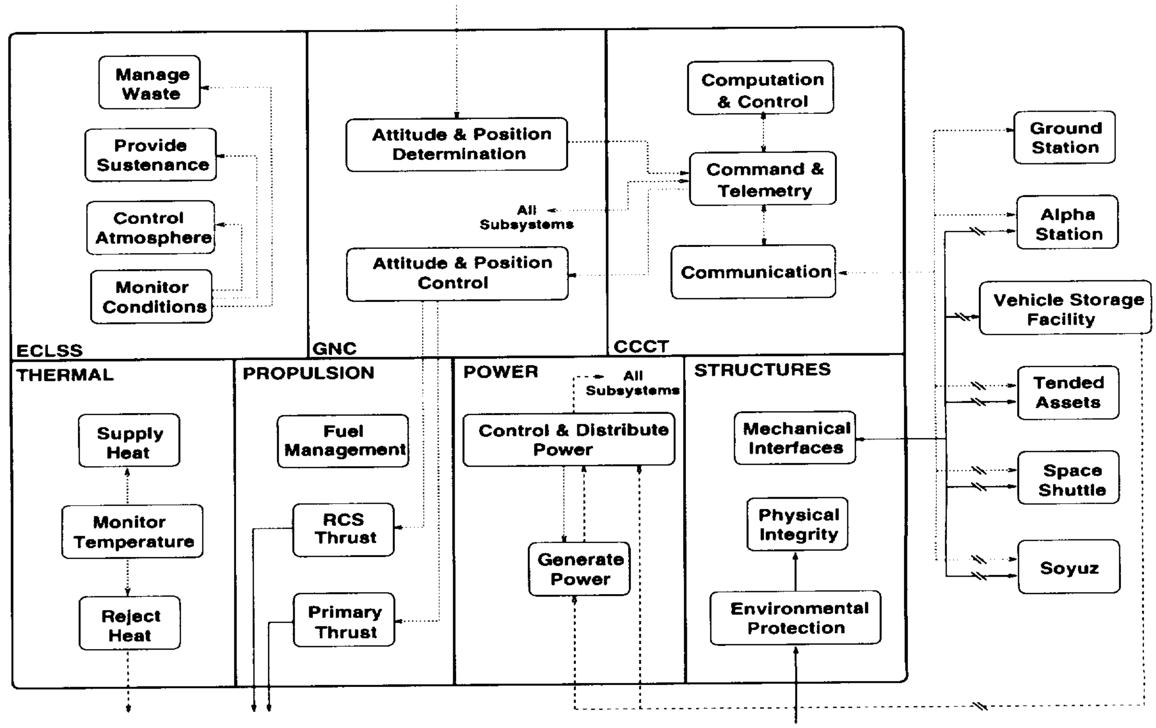

FIGURE 2.1 *System Functional Block Diagram.*

2.3 Vehicle Design Concepts

In the process of designing this vehicle, many different ideas were discussed. Outlined here is a shortened list of leading concepts as well as the basis that was used to select between these options. Finally, an overview of the selected design is presented.

2.3.1 Options

The major concepts that were proposed differed primarily in levels of system modularity, in the range of intended missions and in the type of propulsion used. In addition, the launch vehicle that was to be used constrained the vehicle geometry and had an impact on other issues such as the design of retractable antennas and the amount of fuel that could be lifted into orbit.

Modularity

The primary decision for this vehicle was whether it should be a modular system, consisting of several interchangeable parts, or whether it should consist of a single, integrated design.

A modular design is one that consists of a number of semi-functional units which are assembled into a specific configuration dependent on the required mission. The idea was to have a payload module on the front of the vehicle which was mated via an interface module (a "smart box" consisting of GNC and C3T electronics as well as mechanical interfaces) to a propulsion module as shown in Figure 2.2. On a mission specific basis, a configuration of payload and propulsion modules could be fitted to a interface module. For instance, a non-time-critical autonomous GEO mission to deorbit a satellite might have an electric propulsion system attached to the interface module and some sort of simple grapple to attach to the satellite. On the other hand, a nearby crewed mission would require a higher thrust propulsion system (perhaps cryogenic) and a pressurized crew module as payload.

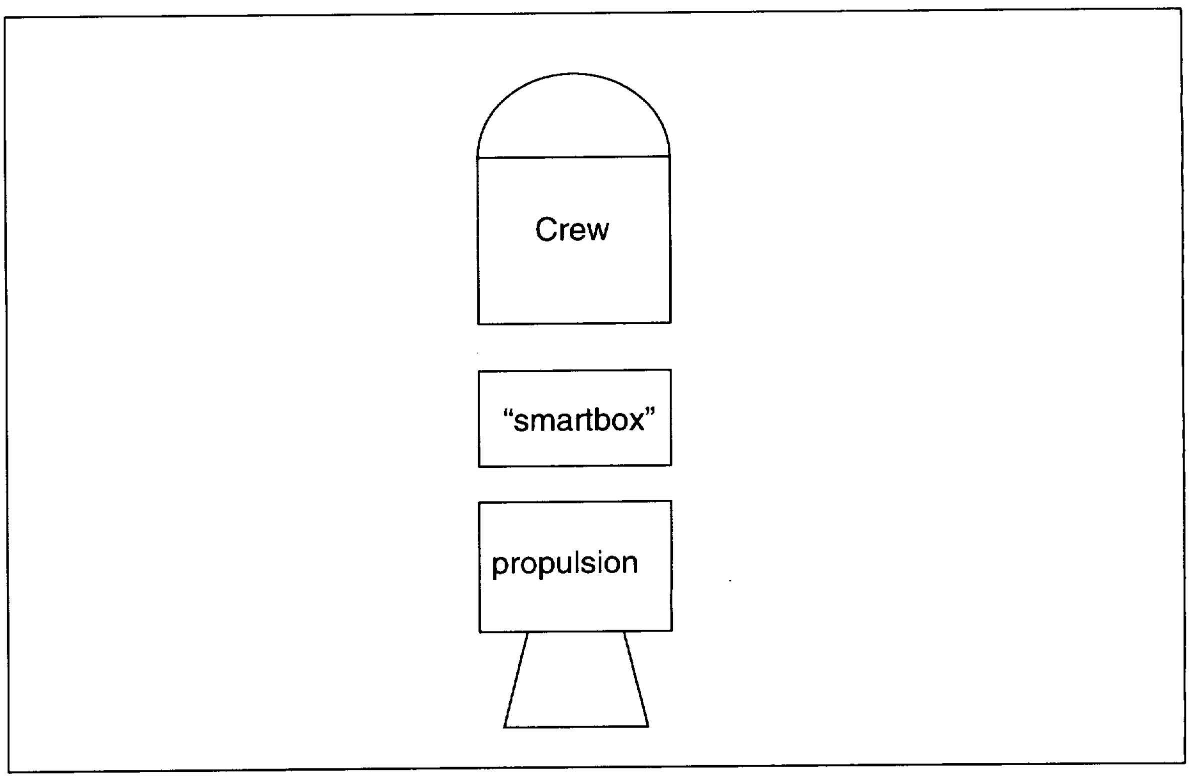

FIGURE 2.2 *Modular Design Concept Representation.*

The other extreme is a single vehicle, as shown in, Figure 2.3, designed to do all of these tasks. This vehicle would consist of a high thrust engine and a crew module as well as autonomous capability. All of the necessary payload modules would be integrated onto the single vehicle.

Expandability

Closely tied to the options on the level of modularity of the vehicle were the options regarding expandability. It was recognized that the large difference in fuel required to get to GEO made the ability to add additional thrust or fuel a necessity, in all but the most fully integrated (nonmodular) systems. The options presented consisted

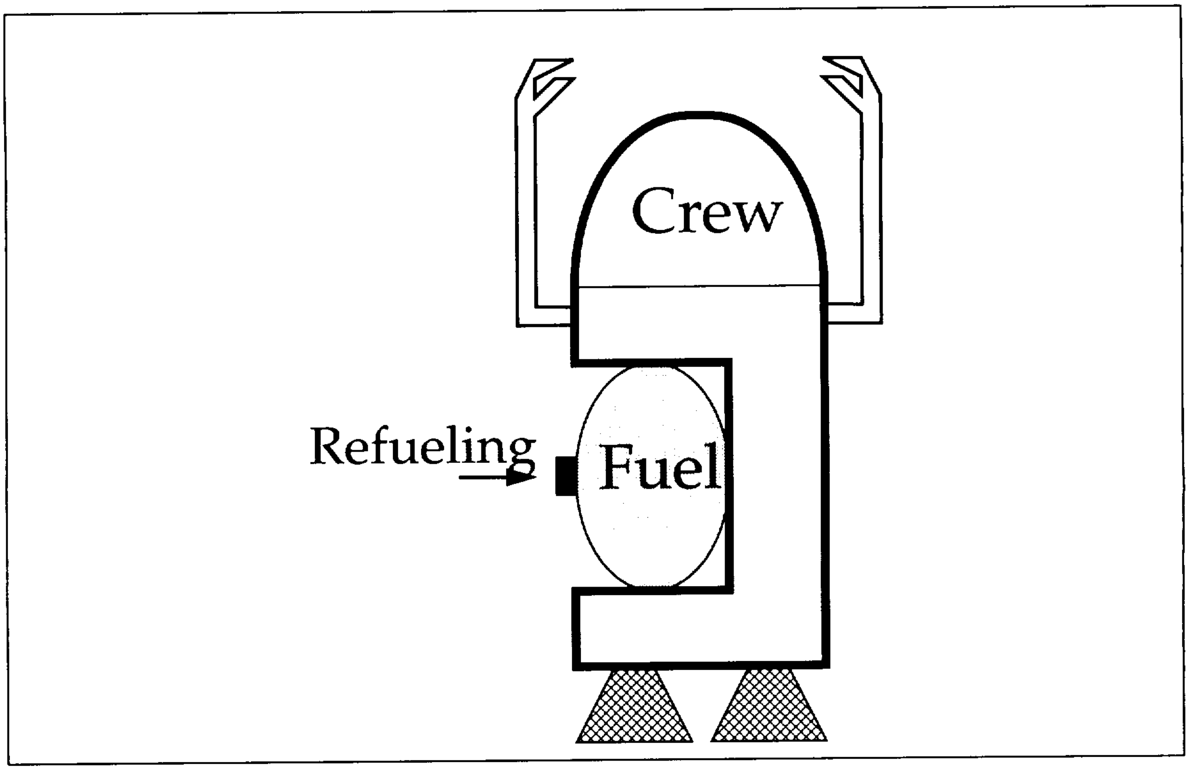

FIGURE 2.3 *Integrated Design Concept Representation.*

of extra fuel tanks that could be added to the system or having entire boosters (fuel tanks and engines) that would attach to the base vehicles, see Figure 2.4.

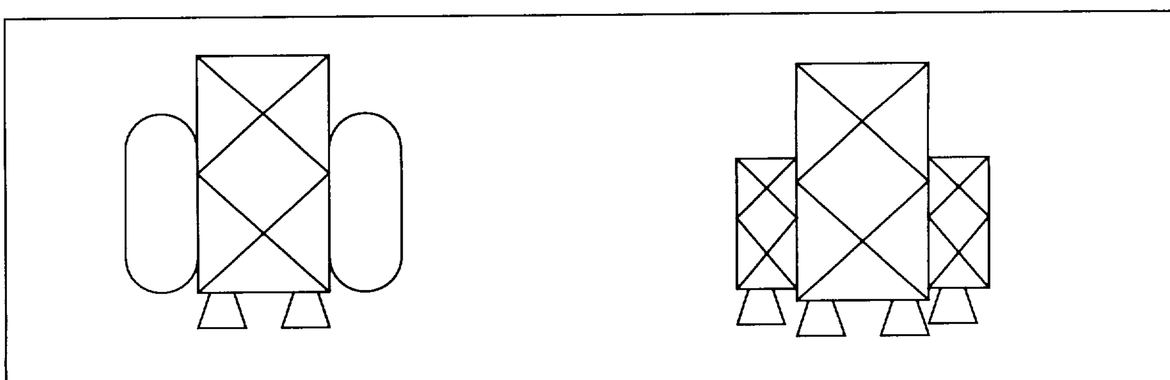

FIGURE 2.4 *Fuel Tanks Concept Versus Booster Packs Concept.*

Propulsion System

The decision on propulsion system had several aspects. The first was whether to consider liquid, solid, hybrid or electric propulsion. Throttleability was a requirement of the vehicle (to allow accurate attitude and location control), so solids were unusable. Electric engines did not have a high enough thrust for any manned missions

-- the travel times would be on the order of months. Hybrids have not yet been flight tested in any systems, and are not intended to be tested in the near future. Based on this short listing, the propulsion system choice became a decision between different liquid propellants: hypergolic fuels (such as SOME HYPERGOLIC FUEL) and cryogenic fuels (such at liquid oxygen and liquid hydrogen).

Launch Vehicle

Finally, the choice of launch vehicle affected the design of the system considerably. Not only at stake were the geometry and mass limitation, but also the ability to use EVAs to help assemble or deploy the vehicle. The size of the fuel tanks was the primary design driver in the choice of launch vehicle -- most simply would not fit the tanks. The second major driver was the cost of the launch and the requirement of having a backup American launch vehicle (if a foreign vehicle was the primary choice). Shown in Table 2.3 on page 47 is the summary of launch vehicle considerations and how the leading candidates compared.

Trades

The basis for determining the concept to design included three major factors: efficiency/low cost, safety of the system and the maturity of the technology required for the design. Note that there was an implicit assumption of forced technology maturation - if a less than mature technology was chosen, it could be tested a verified explicitly for this vehicle at a significant additional cost. In the paragraphs below, each of the options is discussed with particular emphasis on the basis by which the decision was made.

Modularity

The trade-off at stake with a modular design is that system reliability and complexity increases with modularity, as does the intrinsic efficiency of the system. A highly modular design creates a situation where a vehicle can be assembled highly optimized for a specific mission, causing a minimization of fuel wasted in transporting under- or non-utilized components. Transporting a crew module is completely unnecessary for autonomous missions. However, the introduction of additional levels of complexity in the vehicle -- fuel, power, mechanical interfaces -- decreases the reliability of the vehicle. In addition, requiring assembly in orbit either requires considerable investment in robust autonomous operation assets, or astronaut EVA. Both of these options are expensive.

Expandability

As mentioned earlier, extra fuel or thrust is required to get to GEO. The usage of fuel tanks allows a minimum extra dry mass to be added to the vehicle for these extended missions. However, adding only fuel tanks requires a propellent (liquid) interface on the vehicle (to supply the fuel to the existing engines). Adding booster packs does not require a propellent interface -- the fuel can be used exclusively by the booster engines. Booster packs also have an advantage in that they distribute the thrust required for the mission over more engines. This leads to longer intervals between engine replacement.

Propulsion System

The choice between propellants was quickly reduced (by minimum thrust requirements on manned missions) to a choice between two liquid technologies: cryogenic and hypergolic. The reason for choosing cryogenic lies in the greater specific impulse (Isp). A higher Isp creates a larger mission envelope. However, since one of the secondary objectives of the vehicle is to provide rescue capability, long term storage of cryogenic fuels was a major issue. Since the fuels must be kept at extremely low temperatures, additional insulation and other concerns needed to be addressed. Hypergolic propellants are a much more mature technology, but suffer from a lower Isp. In addition, hypergolic fuels are more corrosive and dangerous. However, long term storage of hypergolics is less of a concern.

Launch Vehicle

The trade-offs for the launch vehicles consisted of very pragmatic concerns and included some that were intangible, or at least difficult to quantify. Given that cost was the major driver, the choice of the Proton (if the Freebird fit into the payload envelope) seemed logical. However, the uncertain state of the Russian space industry was a major concern as further explored in Chapter 8 on page 289. Other foreign launch vehicles were considered to be suitable backup vehicles, namely the Ariane series. Political considerations also required that the Freebird fit in an American launch vehicle. The Shuttle was originally rejected, due to the cost under-utilization of its capabilities and the problem of launching cryogenic fuel on a manned vehicle. However, it was reexamined for the additional flexibility of having human intervention available for problems in vehicle assembly.

2.3.2 Concept selection

The concept that was finally selected is that of a semimodular vehicle. The major sections consist of a crew module, a "smartbox", a propulsion pack, extra fuel tanks and a barn for storage purposes. An exploded view of the crew module, the smartbox and the propulsion tank is shown in Figure 2.5.

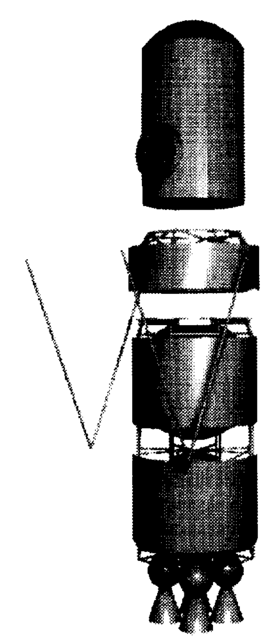

FIGURE 2.5 *Selected Freebird Concept.*

Some of the smaller pieces of hardware have been left off of this figure for simplicity.

The axis system used on this vehicle is shown in the Figure 2.6.

The crew module consists of a pressure vessel that contains all of the necessary life support equipment and human interfaces. The smartbox is a mechanical link between the crew module and the propulsion pack and consists of the required C^3T and GNC components. It provides a mounting point for all of the antennas, guidance instrumentation and thrusters (including thruster fuel tanks) as well as two remote manipulators. The use

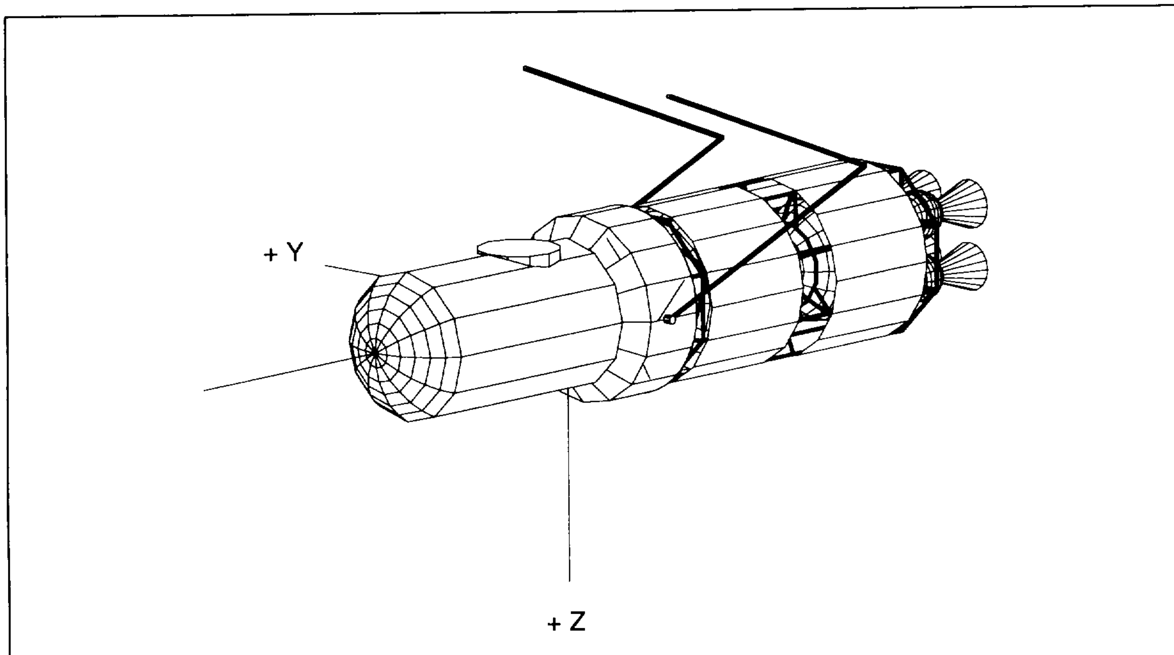

FIGURE 2.6 *CAD drawing showing Freebird coordinate system.*

of two manipulators allows one to hold a satellite and the other to be used as an attach point for astronauts or to manipulate objects for repair purposes. The propulsion pack consists of a set of fuel tanks and four RL10-A4 cryogenic engines. Extra fuel tanks can be attached to the propulsion pack to provide the system a further range. The VSF is a simple structure nearby the space station that will provide storage for the vehicle (or sections of the vehicle) when it is in not use. The rationale behind the VSF is to extend the life of the vehicle by reducing the micrometeorite and radiation damage to the primary structure while reducing the shielding weight that is necessary on the vehicle. A view of each of the components -- crew module, "smartbox", propulsion pack, and extra fuel tanks -- are shown in Figure 2.7.

The will be four basic configurations used in the operation of the vehicle:

1. nearby remote (Mass Ratio up to 5)
2. nearby crewed (Mass Ratio up to 3.5)
3. distant remote (Mass Ratio up to 8)
4. distant crewed (Mass Ratio up to 6.1)

The derivation of these preliminary mass ratios is covered in Chapter 2: Mission Scenarios. Also located there are the operational manner in which these missions will be completed.

The simplest mission will involve a nearby remote mission: the base unit (consisting of the smart box and the propulsion pack) will travel to the satellite and affect the repair. If a crew is required, the crew module will be mated to the base unit. If more fuel is required (for distant missions), extra fuel tanks will be mated to the propulsion pack.

For autonomous missions, the crew module will be removed and stored at the International Space Station Alpha (in the VSF). When a crew is required, the base unit will remotely mate to the crew module using the remote manipulators attached to the smart box.

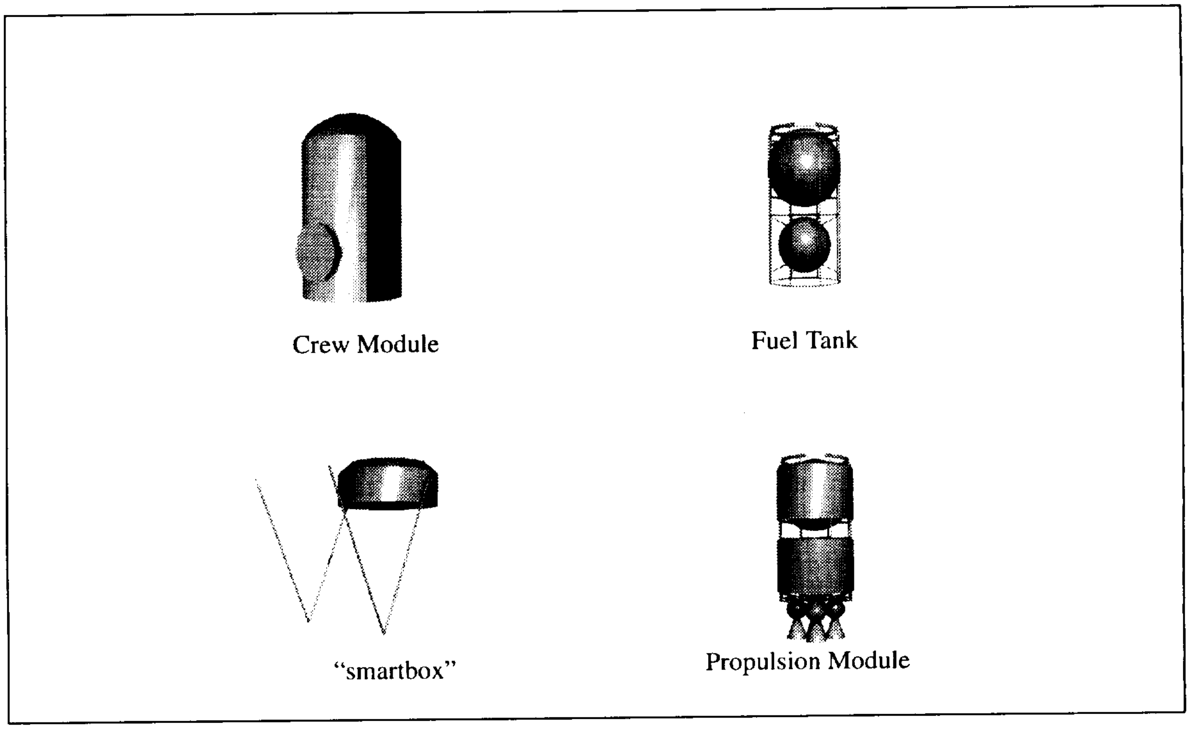

FIGURE 2.7 *Individual Modules in Freebird Design.*

Refueling consists of filling the propulsion pack tank from launched tanks of propellent. Refueling the base tank requires a fluid propellent interface, so no more complexity is introduced by using the same interface for feeding the fuel from extra tanks to the engines. For this reason, extra fuel tanks were chosen as the Expandability option.

As mentioned earlier, the selected propellent was cryogenic, to allow for more efficient orbital transfer. After more research, it was clear that the concerns regarding the on orbit lifetime of the fuel could be satisfied via venting and passive thermal management. The Reaction Control System (RCS) thrusters use an independent hypergolic propellent system. The choice was based on the ease of use of hypergolic propellants in RCS situations, the maturity of the technology and the ability to use empirical evidence from the Shuttle, which has a similar design.

2.4 Vehicle Design Constraints

Several hard constraints were set on the vehicle design by various agencies, in addition to those set by internal agents. The constraints fell into five different categories: safety, launch vehicle, technology maturity, cost and mass. These constraints significantly influenced the design of Freebird.

2.4.1 Safety

Freebird is a crewed vehicle and therefore must comply with NASA's reliability requirement of 0.9944. The Verification and Validation group performed an analysis of technology reliability and assigned reliabilities to each subsystem on that basis, as shown in Table 2.2 on page 47. The life support system was assigned the highest reliability in order to assure crew safety. For a more detailed examination of the reliability issue, see Chapter 11.

TABLE 2.2 *Subsystem Reliability Allocations*

Subsystem	Reliability Allocation
Structures, Materials & Mechanisms	.999
Thermal Control	.999
Power & Propulsion	.999
Environmental Control & Life Support	.9994
Guidance, Navigation & Control	.999
Command, Communications, Control & Telemetry	.999
System Reliability	**.9944**

Since Freebird will be docking with and operating in the vicinity of the space station, certain maneuvering precision requirements apply, as well as limitations on thruster emissions to avoid contamination of the local environment. These restrictions will be discussed further in Chapter 5 (Propulsion) and Chapter 7 (GNC).

2.4.2 Launch Vehicle

Launch vehicle lift capabilities and payload fairings limited the overall size, the mass and the physical configuration of the vehicle, thus affecting the functionality of the vehicle when it reaches orbit Since the capabilities and geometries of launch vehicles are effectively set, the Freebird design was constrained by the launch vehicles currently available. The Titan IV will be used to launch the base unit, consisting of the "smartbox" and the propulsion pack. The crew module will be launched on the Model C Proton, with the Titan III as a backup. Future fuel launches will be conducted on the Model C Proton, and future propulsion pack launches will use the Ariane V. These launch vehicles were selected because they were the cheapest vehicles which could provide sufficient lift capability to get the vehicle to the desired orbit. Table 2.3 on page 47 shows the relevant characteristics of the 4 launch vehicles designated for use by the Freebird program, and Figure 2.8 on page 48 shows the payload fairings of these vehicles.

TABLE 2.3 *Launch Vehicle Characteristics*

Launch Vehicle	Launch Use	Mass to Orbit [kg][a]	Fairing Volume [m3][b]	Cost Per Launch [$M]
Ariane V	Base Unit	22,000	197	90-100
Titan IV	Base Unit Backup	20,000	208	250-300
Proton C	Crew Module/Fuel Supply	20,000	58	35
Titan III	Crew Module/Fuel Supply Backup	14,000	68	130-160

a. Circular orbit at 390km, 51.6° inclination.
b. Primary cylindrical volume only.

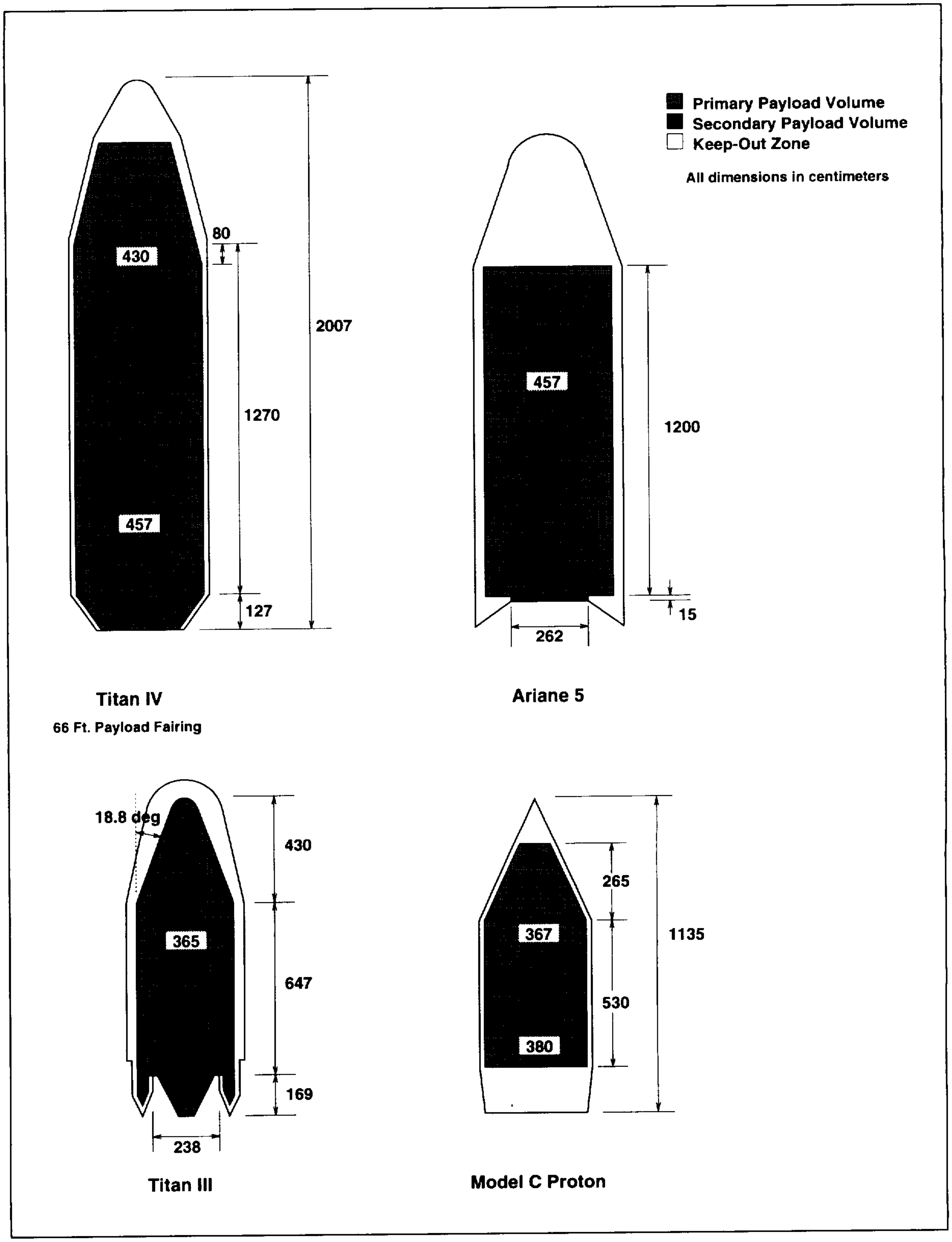

FIGURE 2.8 *Launch Vehicle Payload Fairings.*

2.4.3 Maturity of technology

Since this vehicle is intended to be in use soon after the year 2000, another major constraint was the requirement that the design use proven technology, or technology which is expected to be flight-qualified in time for a technology freeze date one year before launch. Nuclear propulsion, for example, was discarded due to its highly experimental status.

2.4.4 Cost

In order to maintain economic viability, it was decided that Freebird's development and production costs should be limited to $1 billion, including hardware and integration, but not the costly system verification and validation testing. Decisions on the launch vehicle, technology maturity and levels of redundancy were partially driven by this consideration. The vehicle was designed so that operating costs for a repair mission would be considerably less than replacement costs for that satellite.

2.4.5 Mass

As with any space system, mass was a major driver in the design of Freebird. In order to keep launch and operating costs at acceptable levels, it was determined that the base unit should have a mass no greater than 4,000kg and the crew unit should be limited to 3,000kg. These numbers were derived by looking at the average Based on comparisons to other vehicles and an analysis of Freebird's specific situation, these masses were allocated as shown in Table 2.4 on page 49. These numbers include a contingency factor of approximately 15% to allow for uncertainty and growth, as well as 200kg of satellite repair cargo to be carried in the crew module.

TABLE 2.4 *Subsystem Mass Allocations*

Subsystem	Base Unit Allocation [kg]	Crew Module Allocation [kg]
Structures, Materials & Mechanisms	1,850	1,500
Thermal Control	50	75
Power	600	---
Propulsion	1,300	---
Environmental Control & Life Support	---	1,425
Guidance, Navigation & Control	100	---
Command, Communications, Control & Telemetry	100	---
Total Allocation	**4,000**	**3,000**

References

1. National Aeronautics and Space Administration. 1993. *Launch System Highlights for JPL Mission Planning*. Jet Propulsion Laboratory, CA.

Chapter 3

Structures, Materials, and Mechanisms System

This chapter discusses the design and analysis of all structural aspects of Freebird. In addition, issues of the space environment, such as debris and radiation are addressed. Mechanisms, such as the Remote Manipulation System (RMS) and various adapters, are also explained in this chapter.

3.1 Structures, Materials, and Mechanisms Requirements

In order to meet the overall mission requirements, Freebird's structure must:

- provide a pressurized volume for eight crewmembers.
- withstand all loads during launch, ascent, and on-orbit maneuvers.
- provide adequate protection to crew and components from orbital debris and radiation.
- provide adequate interfaces with International Space Station Alpha (ISSA), the Shuttle, and each module-specific launch vehicle.
- meet launch vehicle volumetric and mass constraints.
- include simple and effective interfaces between modules.
- be equipped to capture satellites and stranded astronauts.
- accommodate attachments and volume for anticipated cargo and other equipment.
- provide a Common Berthing Adapter and extravehicular activity capability.
- achieve a reliability of 0.9990.

3.2 Interfaces

Freebird comes into contact with many different types of objects: its launch vehicle, the International Space Station Alpha, the Shuttle, the target satellite, as well as its various modules. Thus, in addition to the conventional system interfaces, there are many specialized mechanical interfaces to consider.

3.2.1 System Interfaces

A subsystem functional block diagram illustrates the components of the subsystem and how they interact with both each other and the overall system (See Figure 3.1.). In the diagram, the solid lines represent mechanical interfaces, the dashed lines power connections, and the shaded lines interfaces with the vehicle computer system. As structural aspects of a system are relatively autonomous, the diagram is not very complex.

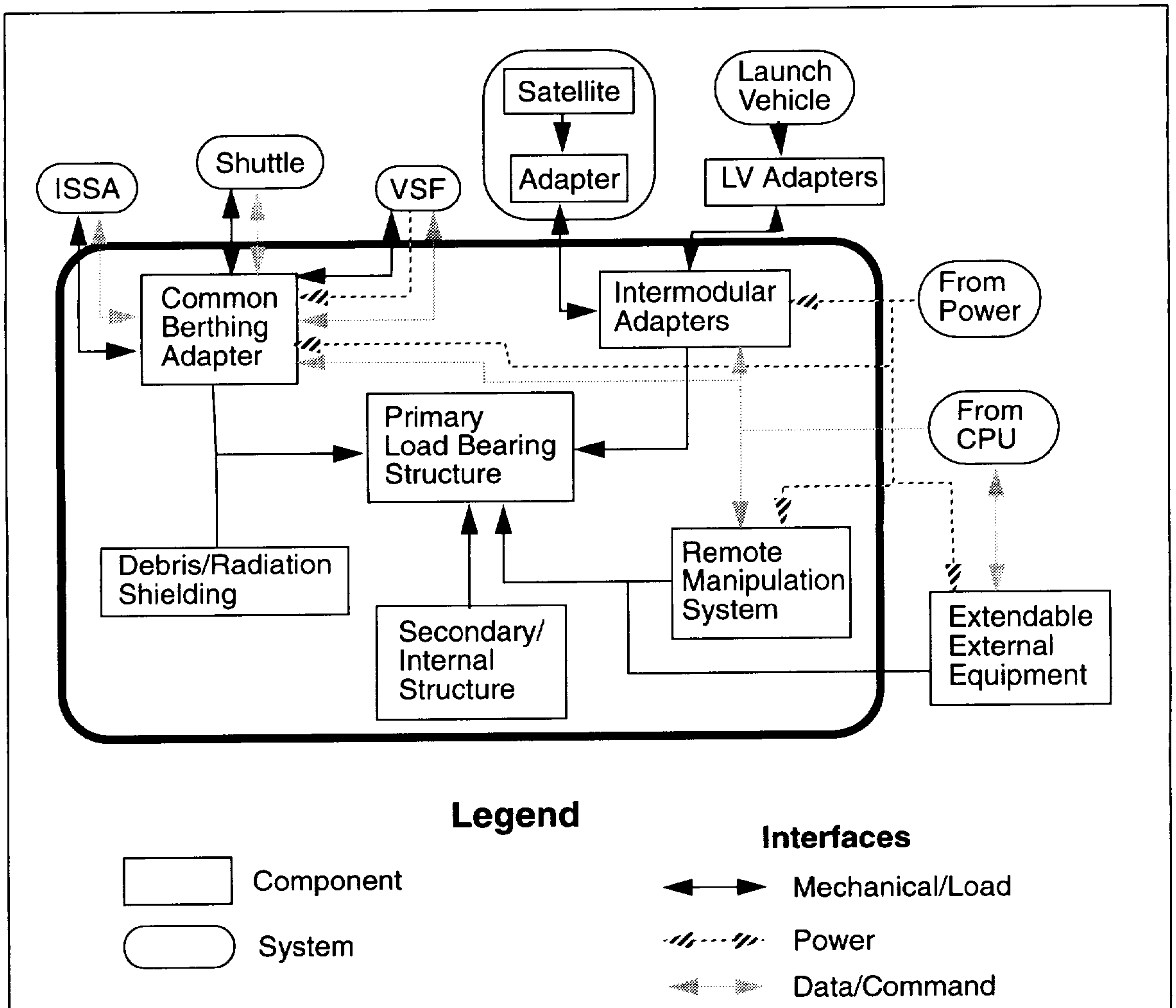

FIGURE 3.1 *Structures, Materials, and Mechanisms System Functional Block Diagram.*

3.2.2 Launch Vehicle Interfaces

All of Freebird's launch vehicles (LVs) use a cantilevered adapter ring for the attachment of cargo. As earlier stated, the possible launch vehicles include the Arianne 5, the Proton Model C, and the Titan III and IV. The LV adapter for the base unit and propulsion pack replacements must be able to fit the appropriate launch vehicle cargo adapters. Because the base unit must be launched "right-side-up" (+x direction toward the nose of the LV), an adapter was designed to transfer the launch loads from the LV adapter to the load-bearing structure of the propulsion pack, around the engines. In order to carry the extreme launch loads the propulsion pack experiences, it will be launched with additional longitudinal struts, attached at various places along the propulsion pack, to bear the bulk of the launch loads. These struts will be joined to the LV adapter, so that the launch loads carried by the propulsion pack itself will be minimal. The adapter consists of a plate which is mated to the LV cargo attachment. To this plate are attached struts which come out at an angle, avoiding the engine nozzles, then connecting to each other via additional members which form an octagon slightly larger than the propulsion pack itself. The joints of this octagon attach to the longitudinal struts which support the rest of the vehicle. Also joined to the additional longitudinal struts are weaves which assist in supporting the fuel tanks during launch. This adapter connects to the propulsion pack via explosive bolts, which are detonated upon reaching orbit.

The mating of the crew module to its launch vehicle adapter is accomplished by a simple connection between the cargo adapter ring and the intermodular interface already present on the crew module. This connection will be released after achieving orbit.

3.2.3 Other Vehicle Interfaces

So that it may dock with the International Space Station Alpha and the Shuttle, Freebird must be equipped with a Common Berthing Adapter (CBA) (see Figure 3.2). Lockheed is currently under contract to design the CBA. It is also used on the Vehicle Storage Facility (VSF). The capture latches and the alignment guides serve to assist in the docking procedure. The active half has powered bolts to secure the connection and is included on Freebird. (The passive half is planned for the ISSA U.S. Node.) The CBA has an outer diameter of 2m with a 20.32cm diameter window in its center [1].

The CBA will be located on the crew module, at the rear of the capsule, on the negative z side (the top of the vehicle). This configuration minimizes the loads associated with being docked.

3.2.4 Intermodular Interfaces

The modular design of Freebird requires interfaces between each module. These interfaces must transfer all mechanical loads without failure as well as accommodate power and data transmission. Several load bearing interfaces were researched, from launch vehicle cargo adapter rings to berthing mechanisms. Because the Freebird concept requires modules that can be removed and replaced easily several times during the vehicle lifetime, a mechanism similar to the berthing mechanisms of both Hermes and the ISSA U.S. Node was chosen.

Two interfaces are required on Freebird, one to connect the crew module to the "smartbox" and one from the Smartbox to the propulsion module. As with berthing adapters, each interface will have an active and passive half. The active half has several features which will be described later, but several of the mechanisms require power. Since the "smartbox" houses all fuel cells, the active half of each interface was mounted to the "smartbox", while a passive half was placed one each on the crew module and propulsion pack.

A schematic of a typical interface is pictured in Figure 3.3. Noteworthy features include:

- axial load path structural ring,
- guide and attenuation assembly (not shown),
- capture latch mechanism,
- powered bolt latching assembly.

The axial load path structural ring is the solid Aluminum 7075 ring on which all other interface mechanisms reside or to which they connect in the case of a passive half structural ring. Its serves to distribute axial

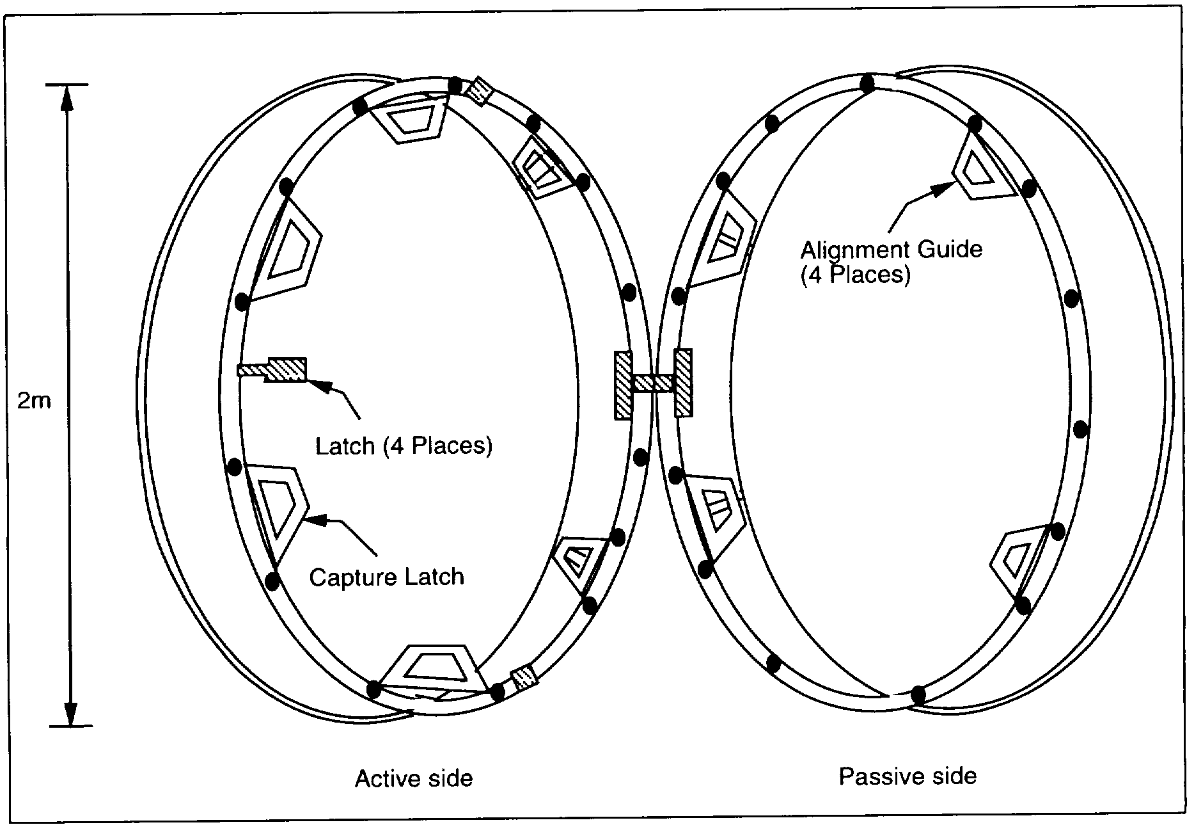

FIGURE 3.2 *The Common Berthing Adapter.*

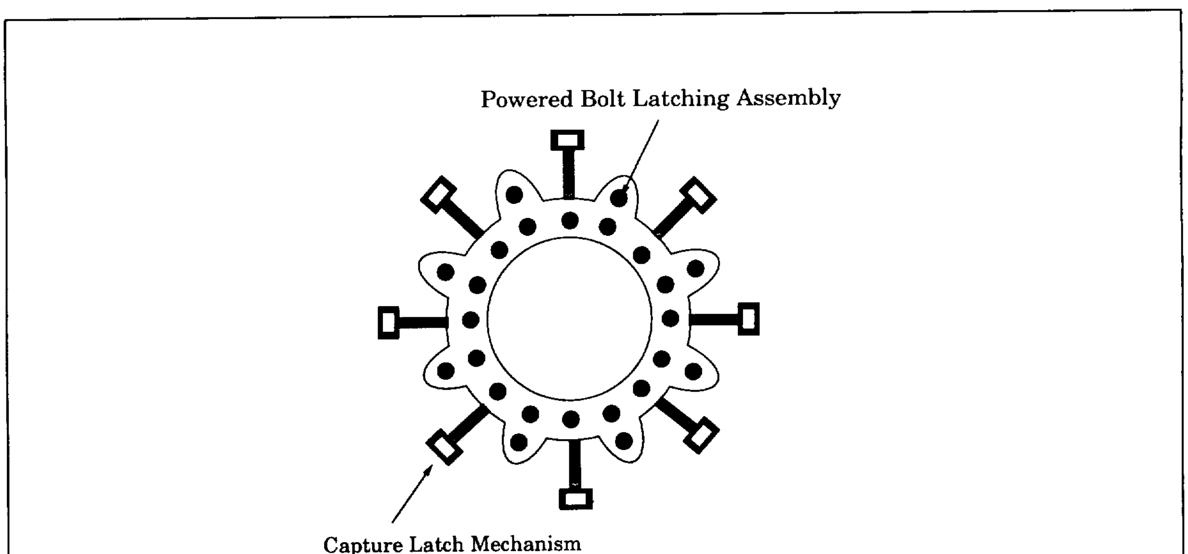

FIGURE 3.3 *Intermodular Interface Schematic.*

compressive loads to the primary structure of each module. The crew module/"smartbox" interface has an outer diameter of 3.3 meters, while the "smartbox"/propulsion pack interface has an outer diameter of 3.9 meters. These dimensions were chosen to best transfer the loads to the primary structure of each separate module.

The *guide and attenuation assembly* (not shown) provides exactly what its name implies. The assembly guides two mating interfaces into proper alignment while reducing the relative velocity to sufficiently low levels where no damage will be incurred and mating and latching can take place properly. The assembly is also capable of providing an impulse to aid in separation.

Eight *capture latch mechanisms* are spaced equally around the structural ring on the active portion of each interface. These latches act to secure the interfaces while the powered bolts make the final mechanical connection. They also compress the attenuation system as the two interface rings are drawn together. During module separation, the latches allow for a controlled separation of the interface rings as the attenuation system pushes the rings apart via spring loading. Figure 3.4 illustrates a typical capture latch mechanism.

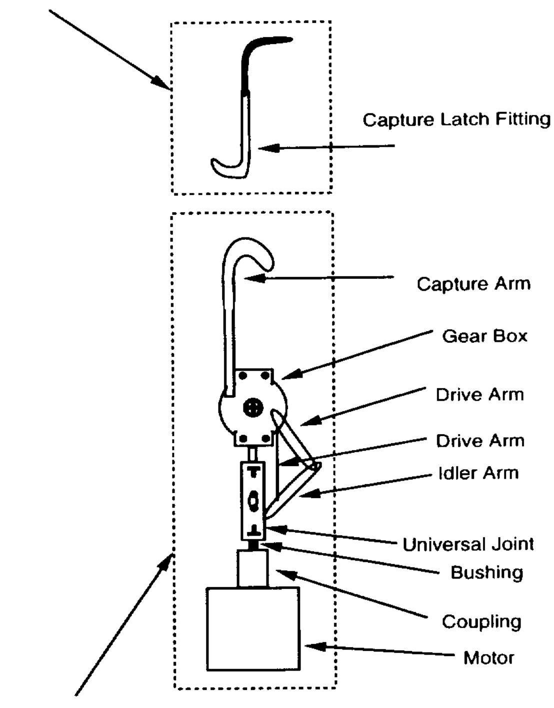

FIGURE 3.4 *Typical Capture Latch Mechanism.*

Powered bolts provide the necessary compressive force to fully mate the active and passive interfaces as well as the means for carrying shear and tensile "pullout" loads throughout the lifetime of the vehicle. In order to determine the number of bolts and their shear and pullout requirements, the interface load environment was characterized. A factor of safety of 1.4 was applied. Table 3.1 summarizes bolt shear requirements and drivers, assuming 24 powered bolts. These shear loads are very low for common bolts, with some bolts capable of handling shear loads near 15kN.

TABLE 3.1 *Powered Bolt Shear Capacity and Drivers*

Interface	Bolt Shear Driver	Shear Capacity Required [kN]
Crew Module/"Smartbox"	RCS thrusters causing roll	1.4
"Smartbox"/ Propulsion Pack	laterally applied launch loads	2.3

The "pullout" requirements were the drivers behind the actual number of bolts. The drivers for the "pullout" requirements are summarized in Table 3.2. Bolt pullout limits can typically range from around 15kN to 42.2kN. [2] The use of 24 bolts achieved the optimal balance of weight and margin of safety. Because the "smartbox" and propulsion tank truss structures use eight axial struts in a regular octagonal fashion (see Section 3.3 on page 57), 8 pairs of powered bolts are placed in an octagon matching the truss members, and eight individual bolts are spaced equally between pairs. The placement of the eight latching mechanisms is the same as the placement of the 8 individual bolts (See Figure 3.3.).

TABLE 3.2 *Powered Bolt Pullout Capacity and Drivers.*

Interface	Bolt Pullout Driver	Pullout Capacity Required [kN]
Crew Module/"Smartbox"	dual off-axis main engine firing	22.9
"Smartbox"/Propulsion Pack	dual off-axis main engine firing	22.9

The bolts are tapered at the ends to compensate for some misalignment or flexure of the structural rings. Each powered bolt actuator can provide typically 101.7N•m of torque to drive the bolt into the passive bolt hole on the opposite interface [2]. A powered bolt is pictured in Figure 3.5.

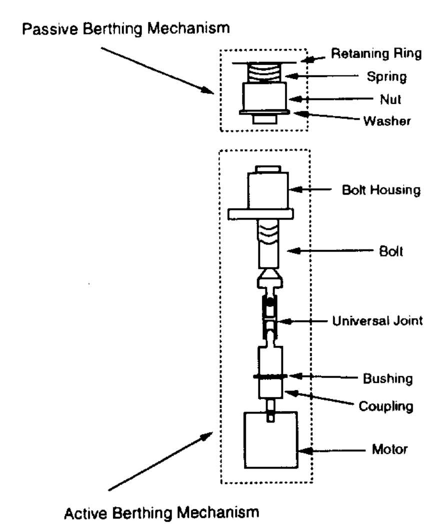

FIGURE 3.5 *Typical Powered Bolt.*

A means for connecting power and data lines was not explicitly designed. However, once the powered bolts have locked in place, and telemetry has verified the connection, some connectors will be mated, transferring power and data.

3.2.5 Target Satellite Interface

In order to perform its primary objectives of tending and repairing satellites, Freebird must have the ability to grapple the target satellite. For this purpose, a Remote Manipulation System (RMS) must be included on the vehicle. For the details on the RMS, refer to Section 3.6.1 on page 73.

In the teleoperated configuration, Freebird must also have the ability to physically connect to the adapter ring interface on the target satellite when it is advantageous to take the satellite to the ISSA for repairs. For this operation, the RMS must be either programmed or controlled remotely to stabilize the satellite, then attach it to the "smartbox." Because the intermodular interface is larger than standard cargo adapter interfaces, the costumer must provide an adapter, to be fitted to Freebird prior to mission, to connect the asset to Freebird.

3.3 Primary (Load Bearing) Structure

The primary structure of Freebird is comprised of all load-bearing components. It must carry all loads applied to Freebird during its entire lifetime. This structure is also responsible for macroscopic physical integrity.

3.3.1 Material

Since the main functions of the primary structure are to bear most of the loads that Freebird undergoes and to serve as the final layer of debris/micrometeorite shielding, strength and ductility were the principal drivers in material selection. In addition, since the primary structure comprises a significant part of the vehicle, weight and cost were also of major concern. In consideration of all these factors, aluminum was the clear choice from among the most commonly used aerospace, materials such as steel and titanium. Some of the attributes that make aluminum so attractive are its relatively high strength to weight ratio, ductility, availability, and ease of fabrication. Steel and titanium each have higher strengths, but each suffer from disadvantages such as higher density, lower fracture toughness, and more complicated manufacturability.

Although aluminum's extensive use was a large influence in the selection process, other materials, not as commonly used, were also investigated. For instance, composites, such as graphite/epoxy, have very high specific strength and stiffness. Also, advances in their development have helped to bring them more into mainstream use. Therefore, Graphite/Epoxy T300/E $[\pm 45^\circ / 0^\circ_4]_s$ was chosen for the truss members of the base unit.

Even in taking aluminum as the choice material for the crew module, however, many alloys and tempers must be considered. Recent developments in aluminum-lithium alloys have produced very promising results: 8-10% less density and approximately 10% higher stiffness [3]. However, shortcomings of their ductility, fracture toughness, and stress corrosion resistance weigh heavily against them. Table 3.3 lists several aluminum alloys and their structural properties. For this specific load-bearing application, where strength is of high importance, experience shows the aluminum alloy 7075-T6 to have the appropriate characteristics of high strength as well as high structural efficiency, E/ρ.

TABLE 3.3 *Structural Properties of Aluminum Alloys*

Aluminum Alloy	Tensile Yield Strength [10^6 N/m^2]	Compressive Yield Strength [10^6 N/m^2]	Young's Modulus [10^9 N/m^2]	Density [10^3 kg/m^3]	E/ρ [10^6 m^2/s^2]
2024-T3	310	255	72	2.8	25.7
2219-T87	365	365	72	2.9	24.8
6061-T6	262	255	68	2.7	25.1
7075-T6	496	489	71	2.8	25.4
8090-T851 (Al-Li)	393	393	79	2.6	30.4

3.3.2 Construction

There were four methods of construction considered: monocoque, semi-monocoque, sandwich, and truss. These are illustrated in Figure 3.6.

- A monocoque design consists of a single skin of a certain thickness without longitudinal stiffeners or circumferential stringers. It is the least efficient design option because of its low stiffness-to-weight ratio. It also tends to buckle at relatively small compressive loads if it is long with a thin skin.
- A semi-monocoque design incorporates stiffeners with a skin, resulting in a more efficient design. The stiffeners and stringers help to prevent buckling as well as to maintain the cross-sectional shape. The stiffeners primarily carry the loads due to axial forces and bending moments, while the skin takes on more of the shear stresses due to shear loading and torsion.
- A cylinder comprised of a sandwich construction can achieve very high stiffness-to-weight ratios. The components include the outer and inner facings and the core. A sandwich panel is similar to an I-beam with the facings corresponding to the flanges and the core analogous to the web. The facings carry axial stresses while the core sustains shear stresses. A sandwich construction would have been ideal from a stiffness and weight standpoint. However, because of manufacturing difficulties, it was not chosen [1].
- A truss structure is a collection of members, such as beams, forming a rigid, open framework. Here, composites can be used as the truss members because of the significant savings in weight, but this also results in an increase in cost. To provide for debris and radiation protection, panels can be placed in the open area between truss members. These body panels can be removable so to allow access to the interior of the structure. A truss design is flexible in that the design can be modified to accommodate all types of loading.

Since the crew module is pressurized, a semi-monocoque design has been selected. This design does not have any longitudinal stiffeners but it does have circumferential stringers to provide additional torsional support. The primary material used is Aluminum 7075. The stringers are also Aluminum 7075. For the "smartbox" and propulsion pack, the primary structure consists of Graphite/Epoxy composite truss members with removable Aluminum 7075 panels. The composite truss members need to be outgassed on Earth in order to prevent on-orbit outgassing. The members are also surrounded by Multi-Layer Insulation (MLI) to prevent the composite members from interacting with atomic oxygen. The panels are bolted at each truss node to allow an EVA astronaut to easily repair and/or replace a component within the primary truss structure. The "smartbox" is completely encased by this truss/panel structure while the propulsion pack only has certain sections of its truss structure surrounded by panels. These paneled sections exist in order to provide additional torsional rigidity to the propulsion pack's primary structure and additional radiation and debris shielding to the fuel tanks within the primary structure.

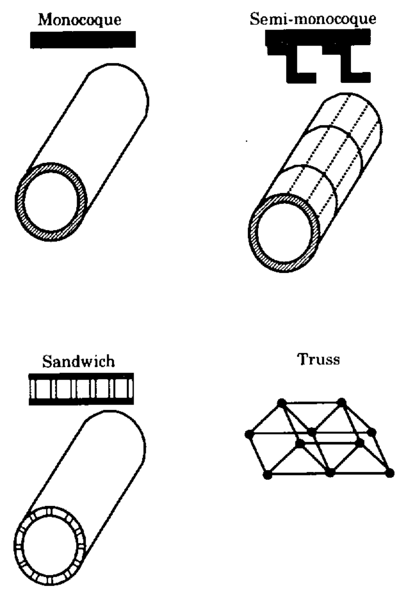

FIGURE 3.6 *Construction Method Examples.*

3.3.3 Analysis

In order to properly size Freebird's primary structure, its load environment was first quantified. Table 3.4 shows the maximum load factors associated with launch.

TABLE 3.4 *Launch Vehicle Load Environment.*

Launch Vehicle	Launch Loads [g's]	
	Axial	Lateral
Proton C	+6	±3
Titan III	+2.5/-5.0	1.7
Ariane V	+4.5	0.2
Titan IV	+3.3/-6.5	±1.5

The maximum loads that the structure will experience during its ten years of normal operation were quantified and subsequently used as the design-limiting case. Two different factors of safety (FOS) were applied: 1.4 for all crew-critical loads and 1.25 for all others. Because designing for launch vehicle loads resulted in an enormously heavy propulsion pack, that module was sized to meet lifetime loads only, with the understanding that it would have to be reinforced during launch. Table 3.5 summarizes the actual load design drivers. The basic design philosophy followed was to define the ultimate load environment and design the vehicle to yield at that load. Thus, the dominant factor governing the mass of the vehicle is the specific strength, σ/ρ, of the materials chosen.

TABLE 3.5 *Summary of Design Drivers for Each Module*

	Rolling Moment	Pitch/Yaw Moment	Axial Load
Crew Module	main engine-induced roll	lateral launch load induced moments	4 main engines firing simultaneously
"Smartbox"	main engine-induced roll	dual off-axis engine firing	4 main engines firing simultaneously
Propulsion Module	main engine-induced roll	dual off-axis engine firing	4 main engines firing simultaneously

Crew Module

The crew module was first modeled as a pressure vessel. The simple pressure vessel equation was used to calculate a thickness by setting the meridianal stress equal to the yield stress of the material.

$$t = \frac{pR}{2\sigma_{yield}} \qquad \textbf{(Equation 3.1)}$$

where t is the thickness, p the internal pressure, R the radius of the vessel, and σ_{yield} the yield strength of the material. Next, the vessel was sized for strength. The effective ultimate axial load was calculated using the following equation:

$$P_{eff} = \left[P_{axial} + \frac{2M}{R}\right]FOS \qquad \textbf{(Equation 3.2)}$$

where M is the applied moment, and P_{axial} the applied axial force. Equating the compressive yield stress with the effective load, P_{eff}, divided by the area of the cylinder cross-section allows for a calculation of the wall thickness required to withstand the maximum compressive stress.

The pressure vessel was then checked for buckling stability using design rules found in for the theoretical cylinder buckling stress, σ_{cr}

$$\sigma_{cr} = 0.6\gamma\frac{Et}{R} \qquad \textbf{(Equation 3.3)}$$

$$\text{where } \gamma = 1.0 - 0.901\,(1.0 - e^{-\phi}), \qquad \textbf{(Equation 3.4)}$$

$$\phi = \frac{1}{16}\sqrt{\frac{R}{t}}, \qquad \textbf{(Equation 3.5)}$$

and E is the Young's Modulus of the material[4]. Equating the buckling stress with the stress associated with Equation 3.2 allows one to solve for the required thickness. For each of several materials examined, the strength required proved the most stringent requirement, although ultimately, the thickness required for radiation protection proved to be the design driver.

In order to save weight, the debris shielding walls are also used to carry loads, thus reducing the required thickness of the pressure vessel wall. In order to transmit torsional loads, five circumferential stringers are spaced equally up the length of the vehicle. These stringers connect the debris bumper segments together as well as connect the pressure vessel wall to the bumper (see Figure 3.8). To size the thickness of these stringers, the maximum torsional load was divided by an average radius to obtain a force. This force was then equated with product of the yield stress and the cross-sectional area to obtain a minimal thickness. Minimum gauge (~1mm) sheets of Aluminum 7075 proved to be one order of magnitude too thick, providing a very large margin of safety.

"Smartbox"

The primary structure of the "smartbox" is a truss. A vessel construction similar to the crew module is inappropriate, due to the proliferation of equipment that must extend beyond the vessel wall. An octagonal truss was chosen to allow maximum strength while preserving a large open surface area through which equipment may pass. The truss work connecting the upper and lower interfaces is a Warren truss with eight verticals/axials. To strengthen the truss against shear loads, some additional reinforcements were added. For a detailed description, see Section 3.3.4 on page 62.

The preliminary sizing of the struts was done by observing that the greatest stress in a strut will be:

$$\frac{\sigma_{max}}{FOS} = \frac{M_{max}z_{max}}{I_{min}} + \frac{M_{max}y_{max}}{I_{min}} + \frac{P_{max}}{A} \qquad \textbf{(Equation 3.6)}$$

where A is the area of one strut, M_{max} is the maximum moment about the y and z axes, P_{max} is the maximum axial force, y_{max} and z_{max} the largest y and z coordinates of all truss members, and I_{min} is simply the area moment of inertia, equal to $4AR^2$ for an octagon. This equation is easily solved for cross-sectional area, which can be calculated by equating σ_{max} to $\sigma_{compressive\ yield}$. The cross-sectional shape was assumed to be a circular tube, a typical space truss cross-section. The outer diameter was set at 3 inches or 7.62cm, based on a preliminary design of Space Station Freedom's truss structure. [5] This choice allows a simple calculation of the thickness required, which then yields an area moment of inertia for the struts. Once a material is chosen, it is easy to calculate the maximum length a strut can be before it buckles:

$$l = \sqrt{\frac{\pi^2 EI}{P_{eff}}} \qquad \textbf{(Equation 3.7)}$$

where P_{eff} is the effective axial load. No strut is allowed to be longer than this critical length without reinforcements. For simplicity, all struts in the truss structure were preliminarily sized to be exactly the same as the axial struts, thus providing a conservative estimate of the mass.

Propulsion Pack

To minimize weight, the propulsion pack was chosen to be an open truss instead of a continuous vessel. For continuity, a scaled version of the octagonal structure of the "smartbox" was chosen. All structural elements were again assigned the same dimensions as those of the axial struts, using the maximum moments and loads appropriate to the propulsion pack. To provide torsional rigidity, debris shielding, and radiation shielding, two Aluminum 7075 cylinders, each three meters long and one millimeter thick, were placed around the main fuel tanks.

As mentioned above, the propulsion pack was designed to withstand all loads encountered in its mission environment. This required the design of some launch reinforcement struts. Using the same sizing technique as above, inserting launch loads where appropriate, an extra 350kg of struts were designed. This additional structure must also carry the vibrational loads during launch.

A detailed thermal, fatigue, and frequency analysis was not performed because these effects are secondary in an initial design phase. The majority of the structure is covered with Multi-Layer Insulation (MLI), so thermal issues are not critical. The largest stress fatigue occurs during launch, after which the loads are much lower. There are also a low number of load cycles during launch, compared to the number needed to fatigue most materials. The frequency vibrations are carried by the extra reinforcements on launch, and therefore do not seriously impact the propulsion pack design.

3.3.4 Design

Freebird's modules were each designed to withstand the module-specific expected loads, as analyzed in Section 3.3.3 on page 59.

Crew Module

The crew module consists of an aluminum semi-monocoque pressure vessel of radius 1.64 m, surrounded by thermal, radiation, and debris protection layers. The pressure vessel is cylindrical in shape, with a 60° cap, and has the shape and cross-sectional wall profile shown in Figure 3.7 and Figure 3.8, respectively.

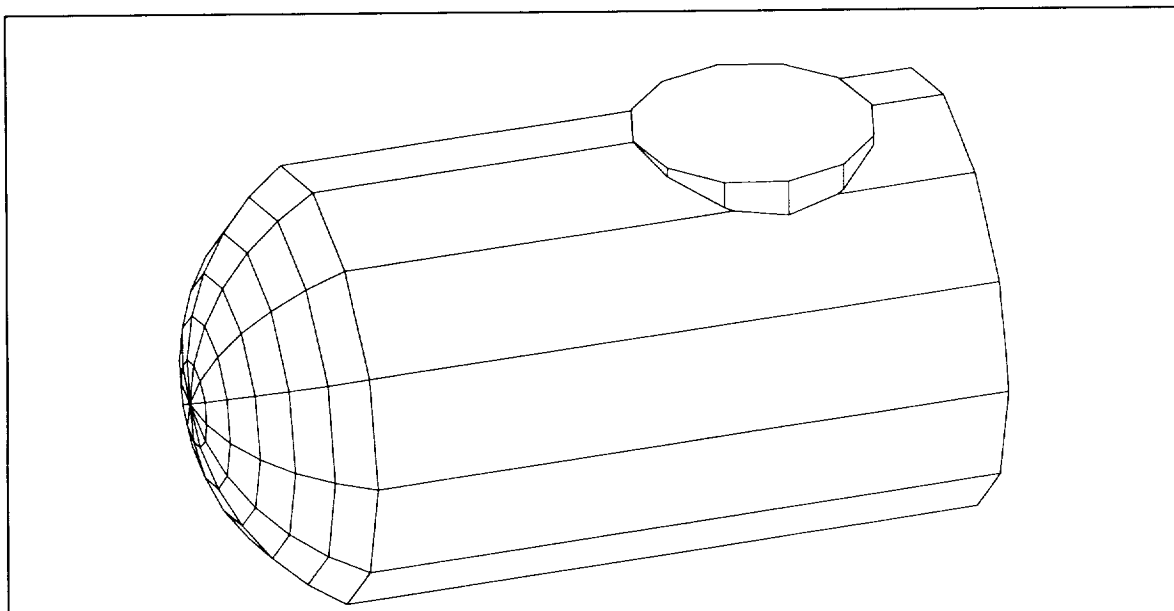

FIGURE 3.7 *Crew Module Outer View.*

In order to support the maximum internal cabin pressure of 14.7psi required for the crew, the pressure vessel walls consist of a 0.14mm thick Aluminum 7075 skin with five aluminum circumferential stringers of cross-sectional area 3cm^2. Surrounding the cabin walls is a 2cm thick open region through which heat pipes, electric wiring, and data cables run. The final layers, components of a Mesh Double Bumper, act primarily as a barrier to the potentially hazardous space debris environment and secondarily as radiation shielding. The Mesh Double Bumper, described in greater detail in Section 3.4.1 on page 66, is a four-layer system designed to protect the vehicle from orbital debris. The innermost of the four layers is a solid aluminum rear wall, 2.25mm in thickness. After a 4cm space is a 1.79mm-thick layer of Kevlar fabric, followed by another void, approximately 22cm in thickness. This void is partially filled with Multi-Layer Insulation (MLI), and is used for thermal protection, as described in greater detail in Part II,Section 6.4.1 on page 107. The next layer is another continuous aluminum wall, 0.93mm in thickness, and the final, outermost layer is a 0.71mm-thick later of aluminum mesh. Summing the thicknesses of the various aluminum layers in the crew module cross section

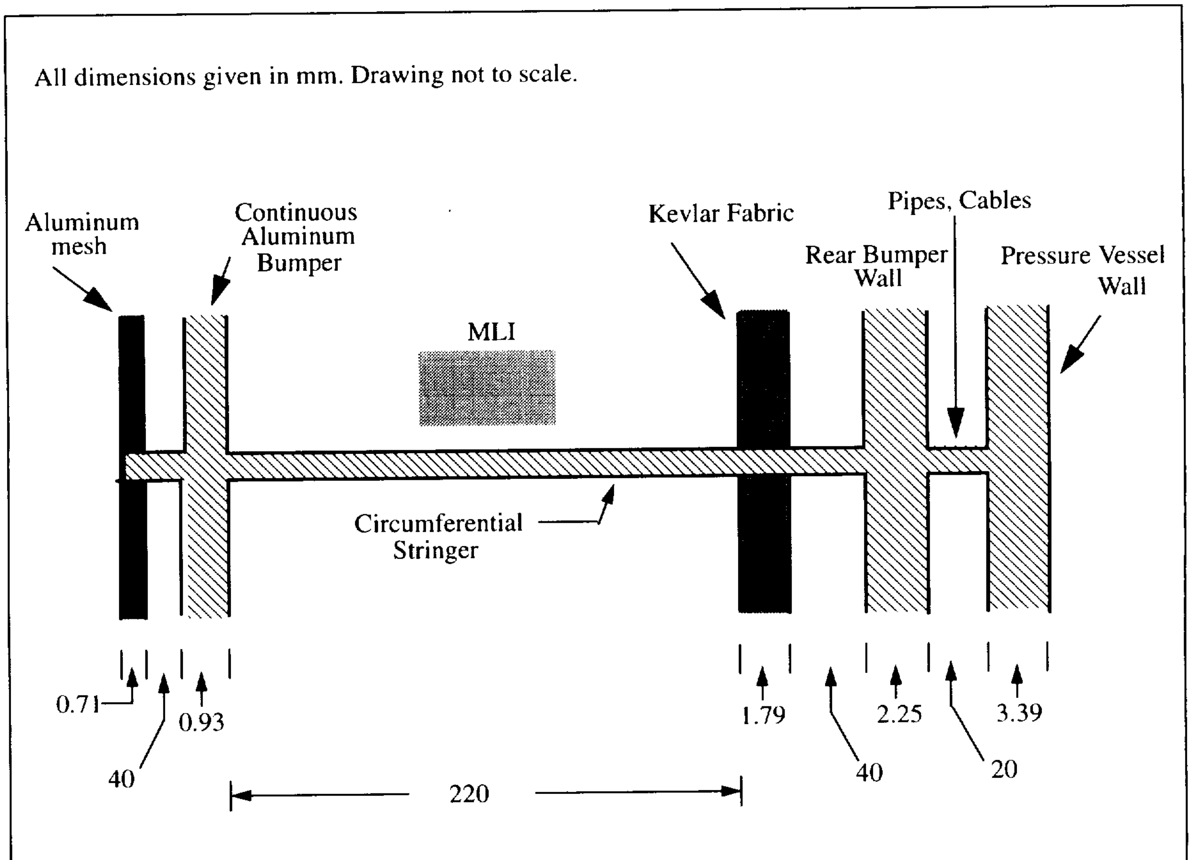

FIGURE 3.8 *Crew Module Wall Profile.*

gives a net aluminum thickness of approximately 4.03mm, which is not quite thick enough for acceptable crew radiation protection. In order to alleviate this problem, the pressure vessel wall is thickened by 3.25mm, to 3.39mm, for a total of 7.28mm of aluminum. This supplementary aluminum also acts as structural reinforcement. For a more detailed description of the radiation environment and vehicular radiation protection schemes, refer to Part II, Section 1.4.2 on page 29 and Section 3.4.2 on page 70, respectively. With the overall configuration outlined above, the crew module adequately supports Freebird's crew and keeps them safe from the harsh space environment.

"Smartbox"

The "smartbox," shown in Figure 3.9, is a short cylindrical truss configuration, surrounded by debris shielding, discussed in Section 3.4.1 on page 66, designed to hold the vehicle's C^3T, GNC, and Power hardware. The trusswork is 2.0m in length and 3.9m in diameter, tapering to a diameter of 3.3m at the "smartbox"/crew module interface. The "smartbox" truss is constructed of T300/E $[\pm45° /0°_4]_S$ Graphite/Epoxy struts. This configuration of graphite/epoxy was selected due to its high specific strength ($\sigma/\rho=5.2 \times 10^5$ m^2/s^2) [6]. The cylindrical truss consists of two parallel octagonal planes supported by eight vertical stringers, connected with diagonal members for torsional support. There is one full-length axial strut as well as two sets of "shear stars" within the octagonal framework for additional support. The shear stars are sets of eight radial struts designed to help bear the torsional and shear loads.

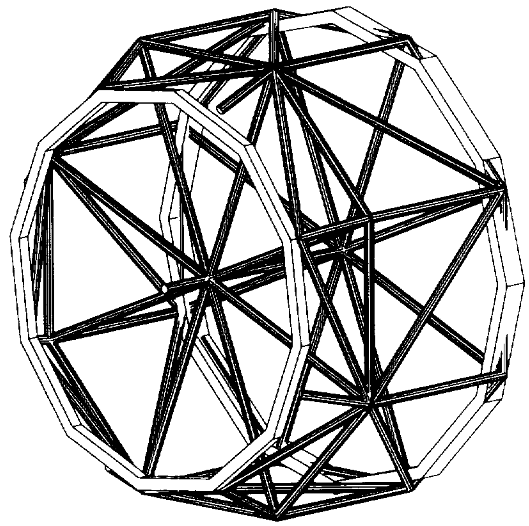

FIGURE 3.9 *"Smartbox" Truss Configuration.*

Propulsion Pack

The propulsion pack is a cylindrical truss structure with internal supporting members, designed to support one 2.1m radius spherical Liquid Hydrogen (LH_2) tank and one 1.5m radius spherical Liquid Oxygen (LOX) tank. The structure additionally supports the four main engines and 28 RCS thrusters. The module is 9.98m in length from the interface to the tip of the engines and is 4.5m in diameter. The truss configuration, pictured in detail in Figure 3.10, consists of an octagonal cylinder with eight vertical stringers and two circumferential skins for torsional and shear support. The LH_2 tank is supported further by two sets of four truss members in a square orientation, also shown in Figure 3.10. Similarly, the LOX tank is supported by two circular configurations, each with four struts to distribute the load to the primary structure. A circular orientation was chosen for this tank because its smaller size prohibits effective use of the square support. A final strut set for engine reinforcement is placed around the main engines and RCS thrusters, in order to facilitate engine load distribution to the vertical stringers. All truss members are again made of T300/E $[\pm 45°/0°_4]_s$ Graphite/Epoxy, for the same reasons as in the smart box. Additionally, the truss member cross-section is the same as that of the smart box truss members, except that the inner diameter is slightly smaller. Refer to Table 3.6 for a summary of truss dimensions for the "smartbox" and propulsion pack. Finally, in order to successfully support the propulsion equipment during launch, 8 extra Graphite/Epoxy struts are added for structural reinforcement. These struts have the same outer diameter and material choice as all of the primary truss members, but are slightly thicker. Again, for exact dimensions, refer to Table 3.6.

TABLE 3.6 *Truss Member Dimensions for the "Smart Box" and Propulsion Pack*

Truss Member Location	Outer Diameter [cm]	Inner Diameter [cm]	Thickness [cm]
Smart Box	7.62	7.48	0.14
Propulsion Pack (primary truss)	7.62	7.46	0.16
Propulsion Pack (Launch Reinforcement Struts)	7.62	5.99	1.63

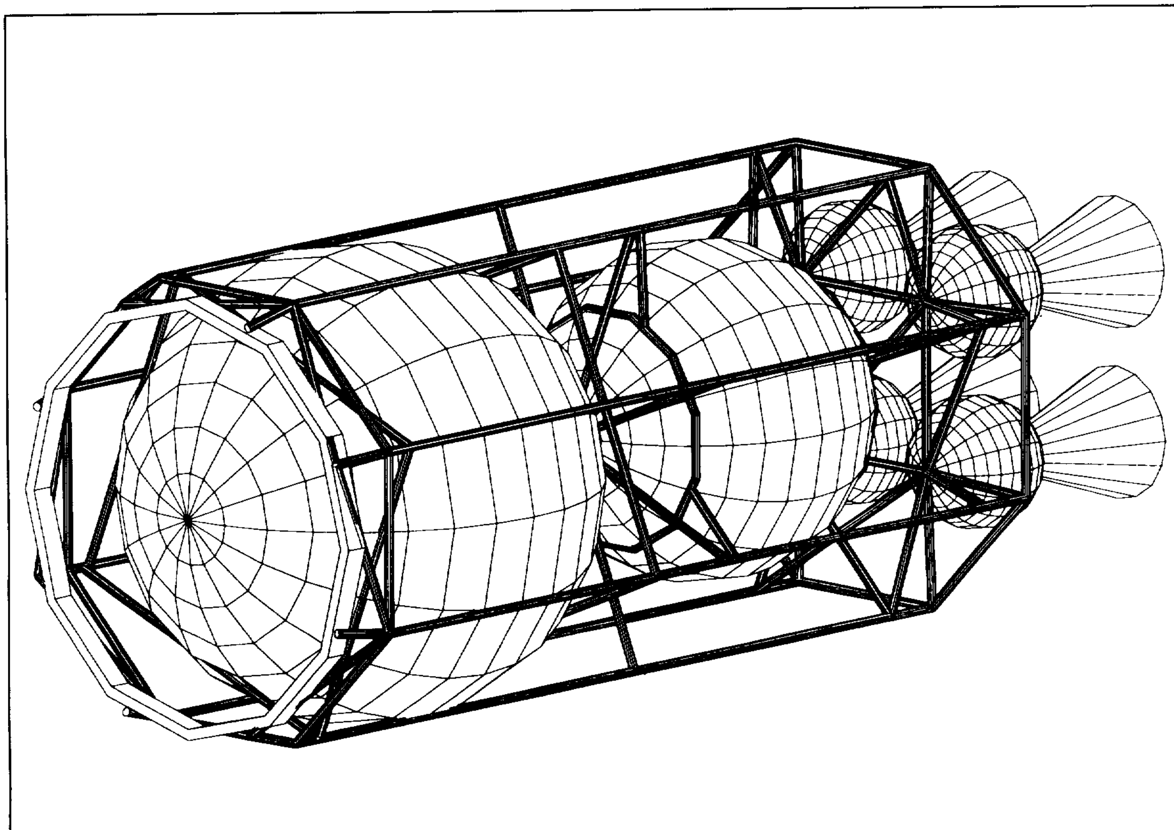

FIGURE 3.10 *Propulsion Pack Truss Configuration.*

3.4 Environmental Protection

In order to maintain vehicle integrity and ensure a useful lifetime, Freebird's essential components must be protected from the mission environment.

3.4.1 Debris/Micrometeorite Impact

In order to insure the integrity of Freebird during its lifetime and to meet the specified performance reliability of 0.9990, Freebird has an impact protection layer. This layer is outside of the primary structure, and is designed to withstand the micrometeoroid environment as described in Section 1.4.2 on page 29. By designing to withstand impacts from 1cm objects at a counterorbital velocity of 10m/s, Freebird's debris shielding has a reliability of 0.9996.

Three different debris impact protection designs where investigated: the Whipple Bumper Shield (WBS), the Multi Shock Shield (MSS), and the Mesh Double Bumper (MDB). All three designs incorporate the same general philosophy, shown in Figure 3.11. The shields consist of two or more layers of material, separated by a certain distance. The first layer takes the brunt of the impact, slowing and breaking the particle into smaller, less damaging pieces. The last layer's purpose is to survive perforation from the fragments and impulse loading[7]. Any intermediate layers that may exist serve to further slow and vaporize the particle.

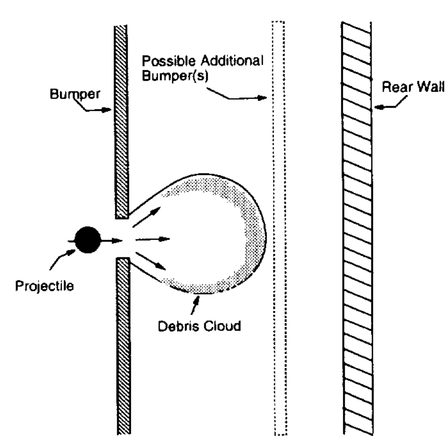

FIGURE 3.11 *General Debris Shield Configuration.*

The Whipple Bumper is the only design currently in use. It is the simplest design, consisting solely of two aluminum layers spaced a certain distance apart. These layers often have some Multi-Layer Insulation (MLI) between them, both to help absorb the shock and to assist in thermal insulation. The WBS configuration is identical to that of Figure 3.11, without the optional additional bumpers.

The Mesh Double Bumper was designed to be an improvement on the Whipple[8]. It begins with the same basic design, two aluminum layers, but then adds two more layers. This is depicted in Figure 3.12. The first addition is an aluminum mesh which goes in front of the first layer. This dense mesh is designed to break up the debris particles in much the same way a continuous plate would, but without the additional mass. A problem with the original Whipple is that in breaking up the particle, the first aluminum bumper itself is broken up, and sometimes this secondary debris can be more damaging than the original particle. The mesh layer diminishes these occurrences. The mesh used is dense enough that the particles that go through unhindered are easily defeated by the next bumper, the continuous aluminum. The second additional layer is between the two continuous aluminum plates. It is a layer of high-strength fabric, such as Kevlar or Spectra, which serves to decrease the impulsive loading on the rear wall of aluminum.

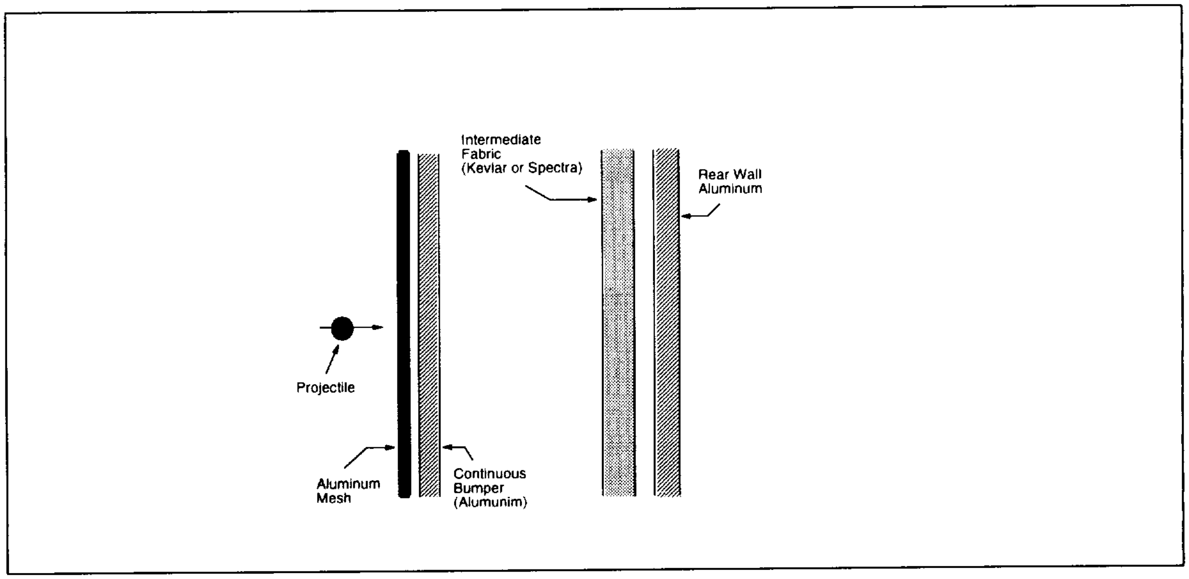

FIGURE 3.12 *The Configuration of a Mesh Double Bumper Debris Shield.*

Instead of an aluminum outer shell, the Multi-Shock Shield utilizes several layers of Nextel, a ceramic fabric. As stated previously, one of the main disadvantages of using an aluminum outer bumper is the spall ejected due to the perforation of the aluminum. The Nextel fabric produces non-damaging ejecta, which allows for a thinner rear wall. The MSS is configured with a number of ultra-thin, spaced sheets of Nextel before a rear wall, again of aluminum[9]. Figure 3.13 shows the geometric configuration. The successive shocks raise the thermal state of the projectile and often cause it to vaporize[8]. A further advantage to this type of shield is that it is easier to fit to a form because it is made primarily out of a fabric.

The shielding masses necessary to protect against a 1cm aluminum projectile at 10km/s were compared for the different configurations to find the optimum shield. The bumper and rear wall thicknesses necessary for a Whipple Bumper can be calculated by using the following equations:

$$t_b = \frac{c_b m_p}{\rho_b} = \frac{c_b d \rho_p}{\rho_b} \qquad \textbf{(Equation 3.8)}$$

$$t_w = c_w \sqrt{\frac{d}{S}} \cdot \sqrt[6]{\frac{\rho_p}{\rho_b}} \cdot \sqrt[3]{M} \cdot V_n \cdot \sqrt{\frac{70}{\sigma}} \qquad \textbf{(Equation 3.9)}$$

where t_b and t_w are the bumper and rear wall thicknesses (cm), respectively, c_b is defined as 0.25 for our case, c_w is defined as $0.16 \text{cm}^2 \cdot \text{s/g}^{2/3} \cdot \text{km}$, m_p is the areal density of the expected projectile (g/cm^2), ρ_b is the bumper density (g/cm^3), ρ_p is the projectile density (g/cm^3), d is the expected projectile diameter (cm), S is the spacing

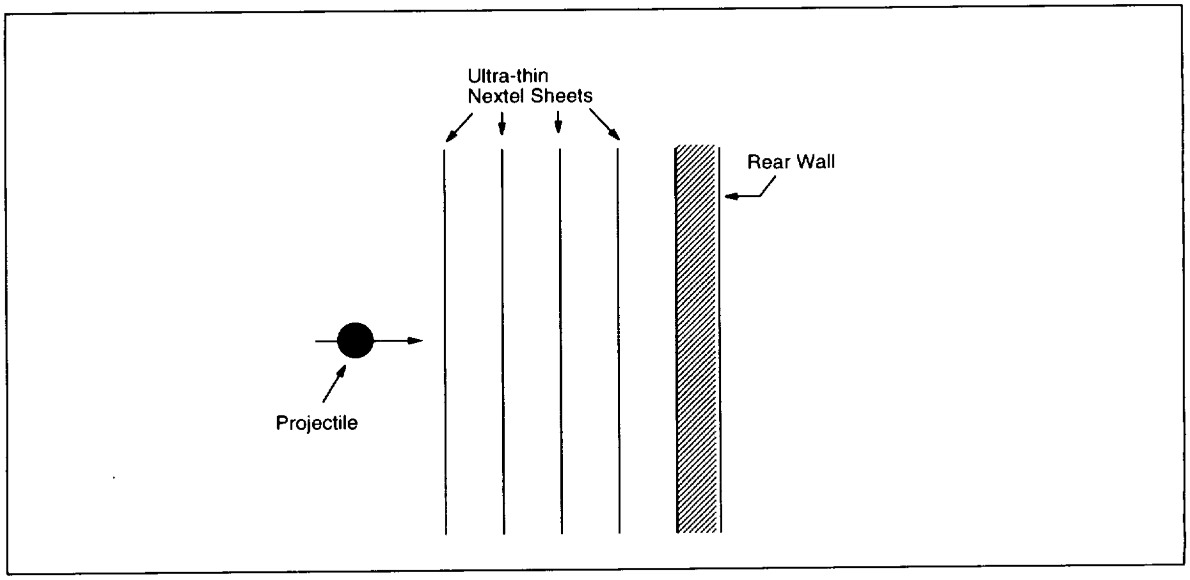

FIGURE 3.13 *The Configuration of a Multi Shock Shield.*

between bumpers (cm), M is the expected projectile mass, V_n is the expected projectile velocity (km/s), and σ is yield stress of the rear wall (ksi). [8] For the Mesh Double Bumper Shield, the following equations characterize the necessary thicknesses[10]:

$$t_{mesh} = \frac{c_{mesh} d \rho_p}{\rho_{mesh}} \quad \textbf{(Equation 3.10)}$$

$$t_b = \frac{c_b d \rho_p}{\rho_b} \quad \textbf{(Equation 3.11)}$$

$$t_f = \frac{c_f d \rho_p}{\rho_f} \quad \textbf{(Equation 3.12)}$$

$$t_w = \frac{9M \cdot V_n \cdot S^{-3/2}\sqrt{\frac{40}{\sigma}}}{\rho_w} \quad \textbf{(Equation 3.13)}$$

In these equations, t_{mesh}, t_b, and t_f are the thicknesses of the mesh outer bumper, the aluminum continuous bumper, and the fabric layer, respectively; c_{mesh}, c_b, and c_f are constants defined as 0.057, 0.093, and 0.064, respectively; ρ_f is the density of the Kevlar fabric.

For the Multi Shock Shield, the wall thicknesses were calculated by the following formulas[8]:

$$t_N = \frac{0.19 d \rho_p}{\rho_N} \quad \textbf{(Equation 3.14)}$$

$$t_w = \frac{41.7 M \cdot V_n \cdot \sqrt{\frac{40}{\sigma}}}{\rho_w} \quad \textbf{(Equation 3.15)}$$

In these equations, the variables stand for the same quantities as in the previous equations, with the subscript N standing for the Nextel bumpers. The thickness calculated for the Nextel bumpers with this formula is equally distributed over the four bumpers.

In these calculations, the material properties of the walls and the bumpers are those of Aluminum 7075. The projectile diameter and velocity used were 1.0cm and 10.0km/s as explained in Section 1.4.2 on page 29. After a number of iterations, it was found that the most efficient configurations for MDB and MSS were both achieved with a total spacing of 30cm. The calculated masses are in Table 3.7. The masses calculated here are preliminary masses, used only to make a design choice.

TABLE 3.7 *Preliminary Shielding Thicknesses and Resulting Masses*

Attribute	Whipple Bumper Shield	Mesh Double Bumper	Multi Shock Shield
Rear Wall Thickness [mm]	6.3	2.25	2.0
Bumper Thicknesses [mm]]	2.5	$t_b = 1.0$ $t_{mesh} = 0.8$ $t_f = 1.8$	1.8 over 4 bumpers or 0.45 per bumper
Crew Module Shield Mass [kg]	1500	750	650
"Smartbox" Shield Mass [kg]	1400	700	625
Total Shielding Mass [kg]	2900	1450	1275

These calculations suggest that the MSS is the most mass efficient design, and should be utilized for Freebird. However, mass is not the only consideration for the design. MSS is a new concept and would therefore have increased development costs compared to a WBS. Also, aluminum is much less expensive than Nextel ceramic cloth. Although these are valid arguments against using the MSS, the chief reason for not choosing it is in its basic design. The cloth nature of the Nextel outer layers would make it difficult to secure Freebird's many external attachments, such as handles, cameras, and antennae. Therefore, the MDB configuration was chosen. Although MDB also has development costs associated with it, the fact that the shielding mass is approximately halved overcomes any hesitation to choose this new technology. Thus, the debris shielding for Freebird will be of the Mesh Double Bumper configuration with the following bumper thicknesses:

- $t_{rear\ wall} = 2.3$mm
- $t_{Al\ bumper} = 0.93$mm
- $t_{Al\ mesh} = 0.72$mm
- $t_{Kevlar} = 1.8$mm

Based on the above thicknesses, the final debris shield masses are 668kg for the crew module and 186kg for the "smartbox." The bumpers are held together by the stringers, or, in the case of the "smartbox," by the truss members of the primary structure. The debris bumper configuration for Freebird is depicted in Figure 3.8.

3.4.2 Radiation Shielding

Because each module contains items that have different radiation damage threshold levels, Freebird's radiation shielding is module-specific. Table 3.8 offers radiation damage threshold levels for the main classes of components found on the vehicle; each module is therefore shielded to protect the most vulnerable component contained in it.

TABLE 3.8 *Radiation Damage Thresholds [11]*

Component Class	Radiation Damage Threshold [Rad]
Sensitive Human Organs	~50
Electronics	10^1-10^3
Ceramics and Glasses	10^6-10^8
Lubricants and Hydraulic fluids	10^5-10^7
Polymers	10^7-10^9
Structural Metals and Alloys	10^9-10^{11}

Several shielding materials were considered for their radiation absorption capability, atomic weight, density, cost and availability, seen in Figure 3.14 and Table 3.9. Figure 3.14 shows the amount of radiation allowed through four different materials to BFO's (Blood Forming Organs) during a 1989 solar proton event. It is given not for its numerical content, but to comparatively show the radiation shielding properties of several materials considered for Freebird's radiation protection program. While aluminum has been widely used on space vehicles in the past, further research points to lithium hydride (LiH) as a better choice than aluminum to supplement structural materials that are not thick enough to provide adequate radiation protection. LiH has a very high hydrogen content and a low atomic weight, and therefore has greater proton stopping ability and less generation of secondary nucleons than heavier, lower hydrogen-content materials[12]. Water, with virtually the same radiation stopping capability as LiH, is also a good material for radiation shielding, but is obviously not applicable for Freebird's mission scenarios because it is a liquid.

TABLE 3.9 *Properties of Various Radiation Shielding Materials*

Material	Atomic Weight [g/mol]	Density [g/cm^3]	Cost	Availability
Lead	207.2	11.35	mid	good
Aluminum	26.98	2.8	mid	good
Lithium Hydride	8.96	0.82	mid	average to good
Water	18.02	1.0	low	very good

The crew module's radiation protection is designed to limit the astronauts' exposure to under 25 rem/mission (the NASA exposure limit), or, for the design-limiting four-day mission, 6.25 rem/day. In LEO a rem is approximately equal to a rad. Summing the individual wall thicknesses needed for structural support and debris protection in the crew module design yields a net aluminum thickness of 4.03mm. However, since 0.71mm of this is dense aluminum mesh, a net thickness of 3.91mm is used to determine radiation shielding

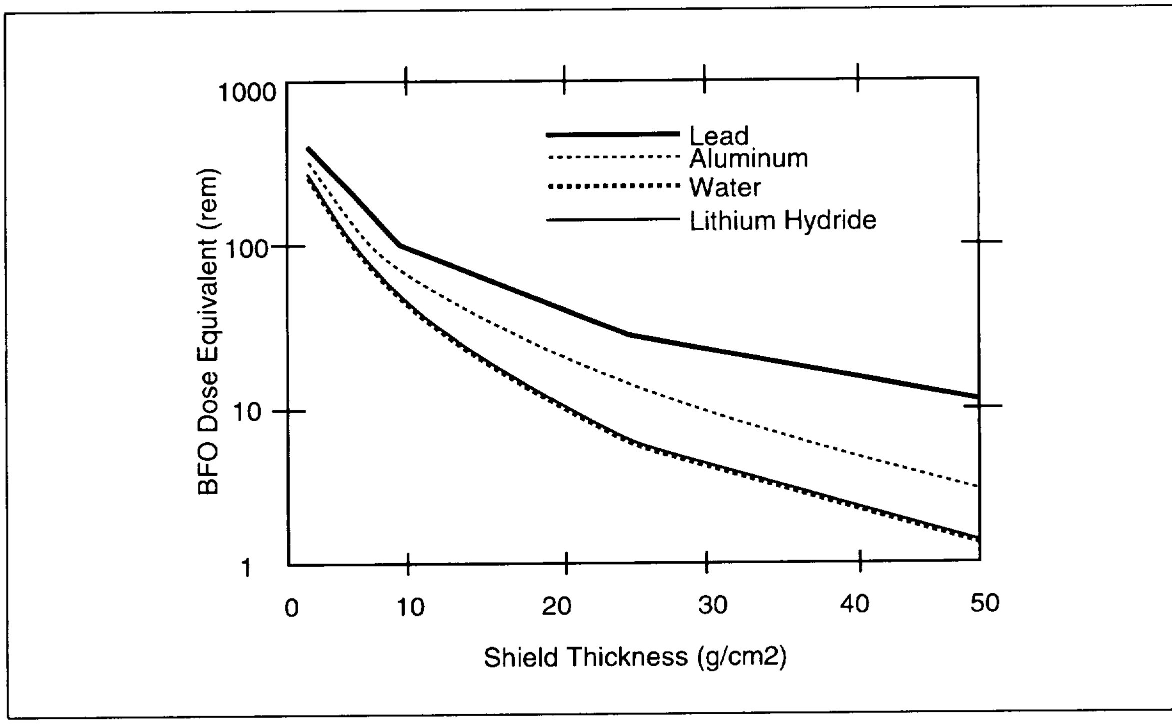

FIGURE 3.14 *Blood Forming Organ Dose Equivalent Versus Depth Functions for Sum of 1989 Flare Fluences for Various Materials [12].*

requirements. In order to facilitate comparison between shielding materials, radiation stopping capabilities are usually given in units of g/cm^2, as such units, when multiplied by the material's density, give equal masses for given allowable dose amounts. Freebird's net aluminum thickness of 3.91mm, multiplied by the density of aluminum, 2.8g/cm^3, yields a mass/surface area number of 1.01g/cm^2. A total thickness of 2.0g/cm^2, or 7.14mm, of aluminum is required to protect astronauts in LEO missions[13]. 3.25mm of aluminum are therefore added to the pressure vessel wall thickness to insure that Freeebird's astronauts do not exceed their radiation exposure limits. Aluminum is used as the supplementary radiation protection material rather than Lithium Hydride because it offers a greater ease of manufacture, as the shielding and the pressure vessel can be manufactured simultaneously. Furthermore, as seen in Table 3.10 [11], for the worst case scenario of a 600km equatorial orbit, the 2.0g/cm^2 provides a more than adequate margin of safety for radiation absorption doses.

TABLE 3.10 *Radiation Dose Rates at a 600 km Orbit for Aluminum Shielding [11]*

Equivalent Shielding Thickness [g/cm^2]	**Radiation Dose Rate [rem/day]**	
	Polar Orbit	**Equatorial Orbit**
	Polar Orbit	Equatorial Orbit
0.1	3 x 10^3	10^4
1.0	1.3	6.5
10.0	0.4	2.0

Due to the nature of the structure of the "smartbox,", it is impossible to provide a simple layer of radiation shielding for the entire module. Each component on the "smartbox" has its own radiation shielding, either on the component itself, or in the form of an aluminum container thick enough to provide adequate radiation protection. All electronics used on Freebird are radiation hardened, and can therefore withstand 10^3-10^5 rads before being damaged. With the anticipated 12 missions/year, 4 days/mission, and 10-year expected vehicle lifetime, the electronics can receive 10^3-10^5 rads over 500 days, or 2-2000 rads per day. This dose allowance would require the aluminum box wall thicknesses to be 3.57cm thick for the most sensitive components if they had no individual shielding. Due to volume and mass constraints on the "smartbox," this box thickness is unacceptable. In order to alleviate this conflict, the components are placed in aluminum boxes with varying wall thicknesses up to 1.5 cm, individually evaluated for radiation protection necessity and component-based shielding, and placed accordingly in the "smartbox." The more sensitive components are placed toward the interior of the "smartbox," utilizing the surrounding boxes for shielding. Furthermore, it should be noted that the vast majority of Freebird's electronic components are closer to the upper limit for radiation tolerance, and only a small percentage require the special interior placement in the "smartbox."

The propulsion pack contains liquid hydrogen, liquid oxygen, and hydraulics, all of which can withstand 10^5-10^7 rads before damage. The tanks holding these fuels are of ample thickness, and therefore need no further shielding.

3.5 Internal Structure

The internal structure of the vehicle serves to maintain component configuration and transmit all loads from internal components to the primary, load bearing structure.

3.5.1 Crew Module

The structural members in the cabin are designed to support the cabin layout specified by the Environmental Control and Life Support System (ECLSS). As this is where the astronauts spend most of their time, the crew module is configured to aid the astronaut during their mission. (For more details see Section 9.4 on page 173) These components include: a floor to give the cabin orientation, windows to aid in docking and EVA, a dual storage bay for parts and tools, EVA and docking capability and hard points for the attachment and placement of chairs, restraints, computer consoles, RMS control units, etc. For more information see Section 9.8.3 on page 187. The mass of the internal crew module structure is approximated to be 10% of the mass of the components it is supporting. This approximation leads to an internal structural mass of 159kg. This number was reduced by performing a first cut analysis in which brackets or straps are used to attach other system components directly to the pressure vessel wall as opposed to creating other support structures for said components. For the remaining components that could not be attached to the pressure vessel wall, the 10% approximation was used to determine the mass of the necessary support structure. Thus, the total mass of the internal structure is 125kg, a savings of 34kg compared to the previous design based solely on the 10% approximation.

EVA and docking capability will be attained through the Common Berthing Adapter. For further information see Section 3.2.3 on page 53 and Section 9.8.1 on page 184. Also, see Figure 3.2 on page 54.

3.5.2 "Smartbox"

The structural members in the "smartbox" serve to secure the internal components in place. As a rough estimate, the mass of these supporting members is approximated to be about 10% of the mass that they need to support. According to this calculation, the mass of the structural members should come out to be approximately 77kg. However, the mass of the primary truss structure of the "smartbox" is 93.5kg, about 20% greater than the estimated mass required of the internal structure. Moreover, the interior truss structure was sized to withstand the loads applied to the "smartbox" and its components. Therefore, the primary structure itself can also serve as the supporting structure for the internal and external components. With an additional 10kg for miscellaneous bolts and brackets, the components are thus attached directly to the primary truss structure.

A potential problem with the scenario just described, however, was the number of truss members available for components to be attached to. However, as designed, the internal components take up only 8.3% of the available volume inside the "smartbox." So, not only is there more than sufficient area for attachment points, there is also plenty of room for the aluminum radiation protection boxes as described in Section 3.4.2 on page 70, and for the retraction of external components, such as the antennae.

3.5.3 Propulsion Pack

The internal structure of the propulsion pack consists of additional truss structures: one to which the engines will be attached, one to support the fuel tanks and pumps, and one to attach the additional fuel packs to the main truss structure. Each of these trusses is constructed of the same materials as the primary propulsion pack truss.

There are four engine and four Reaction Control System (RCS) mounts on the base of the propulsion pack structure. They are held in place by a truss structure which is pictured in Figure 3.15. Within this structure are cross beams which hold the pumps and other equipment necessary to operate the engines and RCS. Surrounding this structure (not shown in the figure) is a layer of debris shielding. This shielding is simply an aluminum shell.

The hydrogen fuel tank is supported by cross-beams oriented in squares above and below it. The orientation of the squares are rotated $^1/_4$ turn with respect to each other. The liquid oxygen tank is similarly supported, however the tank is too small to be supported by the square trusses as described above. Therefore, it is supported by circular trusses which are attached to the primary trusses by four members, each. Both of these support systems are described further in Section 3.3.4 on page 62.

The trusses which attach the additional fuel tanks are designed as miniature versions of the main propulsion pack. They are 5.3m long and 3.6m in diameter, in order to fit into the Proton for launch. They are attached to the main propulsion pack by powered bolts on the $\pm z$ axes.

3.6 External Vehicle Equipment

A variety of equipment needs to operate beyond the confines of the "smartbox." These components must often extend, retract, and gimble. Structural reinforcement and operating mechanisms are provided. The location of all external "smartbox" equipment is shown in Figure 3.16.

3.6.1 Remote Manipulation System

Based on Space Shuttle experience it was determined that at least one Remote Manipulation System (RMS) should be present on Freebird. Two arms were chosen, one to grapple the target satellite and one to maneuver working astronauts when Freebird operates in a crewed configuration. For maximum functionality, both arms are capable of being remotely operated for Freebird operations in the teleoperated configuration.

Each robotic arm is essentially a modified version of the Space Shuttle RMS, using some newer technology developed for the Space Station Freedom RMS. Each arm weighs approximately 275kg and has seven degrees

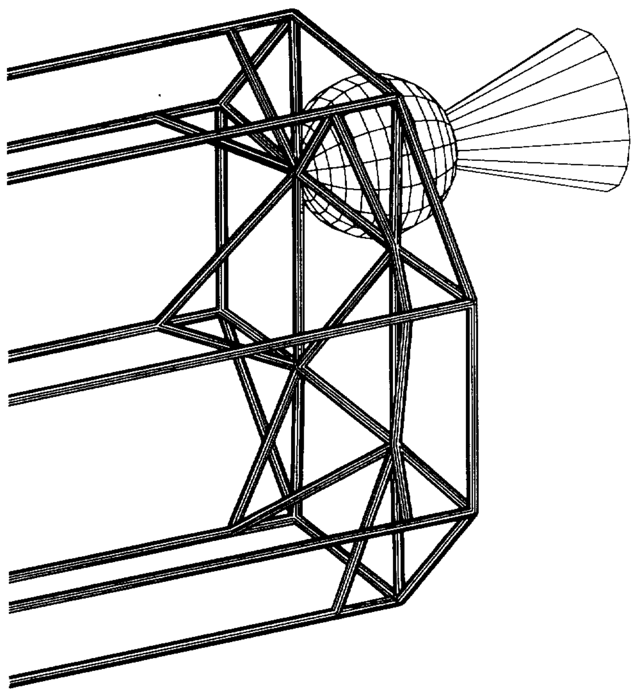

FIGURE 3.15 *Engine Supports on the Aft End of the Propulsion Pack.*

of freedom: three at both the shoulder and the wrist, and one at the elbow. Closed Circuit Television cameras (CCTVs) are located strategically along the length of each arm: one at the shoulder, two at the elbow, and one located on the end effector. In addition, some of the CCTVs are mounted on Pan and Tilt Units (PTUs), and lights are present as well. Each arm is comprised of two segments, each 7m in length, for a total reach of 14m. Each segment is specified to be approximately 15cm in diameter. One arm is capable of moving 20,000kg at an approximate end-point applied force of 80N. Thus, a large payload could be moved at roughly 0.063 radians per second[14].

In order for the arms to have maximum range for all their specific operations, they are placed 180° apart, on the $\pm y$ axes on the "smartbox." In order to have maximum reach, but still stow inside the launch vehicle, each arm rests in a folded position, recessed into the shielding of the "smartbox." During periods of acceleration, each arm folds into its rest position allowing the end effector to attach to the End Effector Retainer (EER). The EER provides a hard point for the effector to hold and restrain the arm during acceleration when the risk of losing control of the arm is greatest.

3.6.2 Antennae

Two 1.25m diameter parabolic antennae are required by the C^3T and GNC systems in order to meet Freebird's communications and navigations requirements. In order to insure a clear pointing path to the Earth at all times and possible vehicular orientations, both antennae are gimbaled in two axes. In addition, two pairs of omnidirectional antennae, each 50cm in length and 5cm in diameter, are necessary. One pair, used for communication during EVA's, is located on the crew module. The second pair, used for telemetry purposes, is

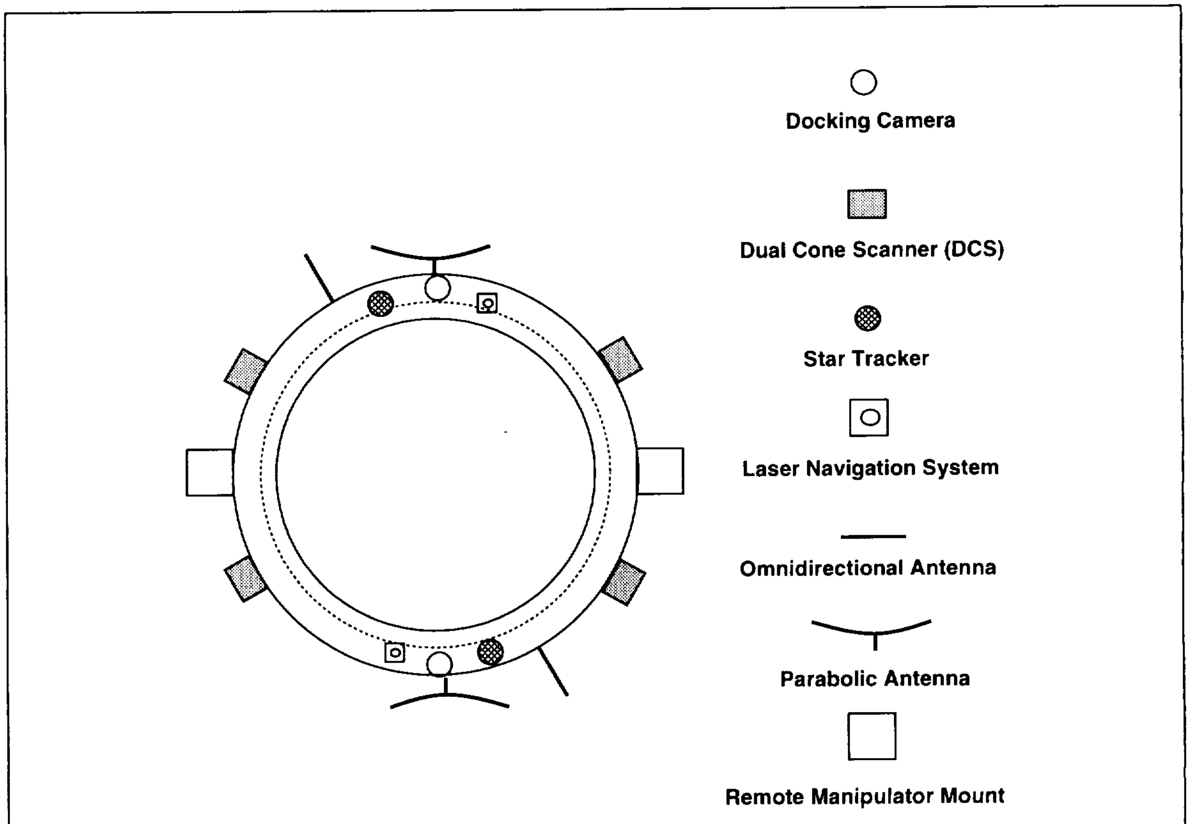

FIGURE 3.16 *"Smartbox" External Equipment Mounting.*

located on the "smartbox." Each of these antennae protrude perpendicularly from the side of the vehicle when in use, and remain embedded in the shielding when passive.

3.6.3 Remote Viewing Devices

To allow Freebird's crew visual access to all areas surrounding the vehicle, several remote viewing devices are strategically placed on the exterior of the vehicle. Two supplementary color video cameras are attached to the vehicle's exterior at hard points, one on each side of the "smartbox." Such placement allows visual access during docking, EVA's, grappling of target satellites, and other orbital maneuvers. In addition, two cameras are attached to each RMS for close-up views of target satellites and EVA activities.

3.6.4 Reaction Control System

Attachments are provided on the propulsion module truss for the reaction control system (RCS). The exact placement and angles of these thrusters are pinpointed in Section 7.7 on page 143.

3.7 Failure Modes and Risk Assessment

The majority of structural failures are, by nature, life-threatening or mission critical. Moreover, the criticality of failure of certain components is dependent on Freebird's configuration at the time. In the teleoperated configuration, a structural failure can only be, at worst, mission critical. In the crewed configuration, however, the same types of failures can be life-threatening. A component level breakdown of the structural failure modes is given in Table 3.11. The criticality of a failure mode is determined by assessing the greatest risk that the failure poses.

TABLE 3.11 *Failure Modes*

Component	Failure Mode	Effect	Prevention or Response	Criticality	
				Crewed Config.	**Teleop.**
Primary Structure	Excessive deformation	Damage to internal structure & components	Factor of safety	1	2
Secondary Structure	Puncture	Loss of atmosphere	Redundancy of bumpers	1	N/A
Internal Structure	Excessive deformation	Harm to crew	Factor of safety	1	N/A
		Damage to other components	Factor of safety	2	2
		Component restraints compromised	Factor of safety	3	3
Shielding	Excessive degradation	Excessive heat or radiation; Damage to primary structure	May be repaired on orbit by EVA	1	2
Remote Manipulator System	RMS malfunction	Arm inoperable/ uncontrollable	Freebird is equipped with 2 RMS's	2	2
Launch Vehicle Adapter	Payload unable to detach	Payload delivery failed; Possible damage to payload	Redundant release mechanism	2	2

TABLE 3.11 *Failure Modes*

Component	Failure Mode	Effect	Prevention or Response	Criticality	
				Crewed Config.	Teleop.
Intermodular Interfaces	Capture hook malfunction	Module attachment/ detachment impossible	Internal Redundancies	2	2
	Powered-bolts malfunction	Module attachment/ detachment impossible	Redundant release mechanism	2	2
	Bolt(s) shear-off/pull-out; hook integrity failure	Intermodular connection compromised or destroyed	Redundancy	1	2
Common Berthing Adapter	Seal leakage	Deterioration of atmosphere	Internal Redundancy	1	N/A
	Capture hook malfunction	Docking/De-docking impossible	Internal Redundancies	1	N/A
	Hatch malfunction	Docking/De-docking impossible	Internal Redundancies	1	N/A

Criticality Key:

1 = life threatening

2 = mission critical

3 = non-mission critical

4 = telemetry critical

3.8 Summary Budgets

See Table 3.12 for a summary of the mass, power, and cost budgets and for each module of this subsystem and Table 3.13 for telemetry information.

TABLE 3.12 *Summary Budgets*

	Component	#	Mass [kg]	Avg. Power [Watts]	Peak Power [Watts]	Hardware Cost [$K]
Crew Module	Pressure Vessel	1	480.0	0.0	0.0	7,619
	Internal Structure	1	125.0	0.0	0.0	1,653
	Radiation/Debris Shielding	1	711.5	0.0	0.0	11,294
	Common Berthing Adaptor	1	130.0	12	200	3,439
Smartbox	Primary/Truss Structure	1	93.5	0.0	0.0	989
	Internal Structure	1	10.0	0.0	0.0	106
	Radiation/Debris Shielding	1	218.3	0.0	0.0	2,310
	Intermodular Interface	2	200.0	24.0	400.0	6,349
	Remote Manipulator Sys	2	550.0	1920.0	2400.0	75,000
Propulsion Module	Primary/Truss Structure	1	216.0	0.0	0.0	2,286
	Radiation/Debris Shielding	1	222.0	0.0	0.0	2,349
	Total	---	2956.3	1936.0	---	113,395

TABLE 3.13 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital	Command
InterModular Interfaces (x2)	Capture Latch	8	10	A	Latch/Unlatch
	Powered Bolts	24	10	A	Forward/Reverse
	Attenuators	4	10	A	Retract/Extend
CBA	Capture Latch	8	10	A	Latch/Unlatch
	Powered Bolts	24	10	A	Forward/Reverse
	Attenuators	4	10	A	Retract/Extend
RMS (x2)	Motor Position	7	100	D	
	Motor Torque	7	100	D	Commanded Torque
	End Effector Position	3	100	D	Commanded Torque
	End Effector Force	4	10	A	
	End Effector Moment	4	10	A	
Para. Antenna Gimbals (x2)	Motor Position	3	10	D	Commanded Position
Omni Antenna Actuator(x4)	Antenna Position	1	.1	A	Commanded Position
GNC Actuators (x6)	Equipment Position	6	.1	A	Commanded Position

C-2.

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Chapter 4

Electrical Power System

4.1 EPS Requirements

The functional requirements of the Electrical Power System (EPS) were derived from the overall system requirements. They are outlined as follows:

- Supply a continuous source of electrical power to all subsystems during the mission life
- Support peak and average loads for all components
- Provide a regulated power bus and converters for all necessary DC voltages
- Protect Freebird from failures and irregularities within the EPS
- Provide command and telemetry for EPS health and status
- Have a reliability of 0.9995

4.2 EPS Interfaces

The interfaces involved in the EPS are:

- Internal
- loads via the power bus
- Telemetry
- Power between the modules

The EPS consists of the power source, a power distribution system, and power regulation devices. The functional block diagram is shown in Figure 4.1. It details the interfaces within the system.

There are also interfaces between the "smartbox" and other modules, to provide power as necessary. When the crew or propulsion module is being attached to the "smartbox", a power junction is established.

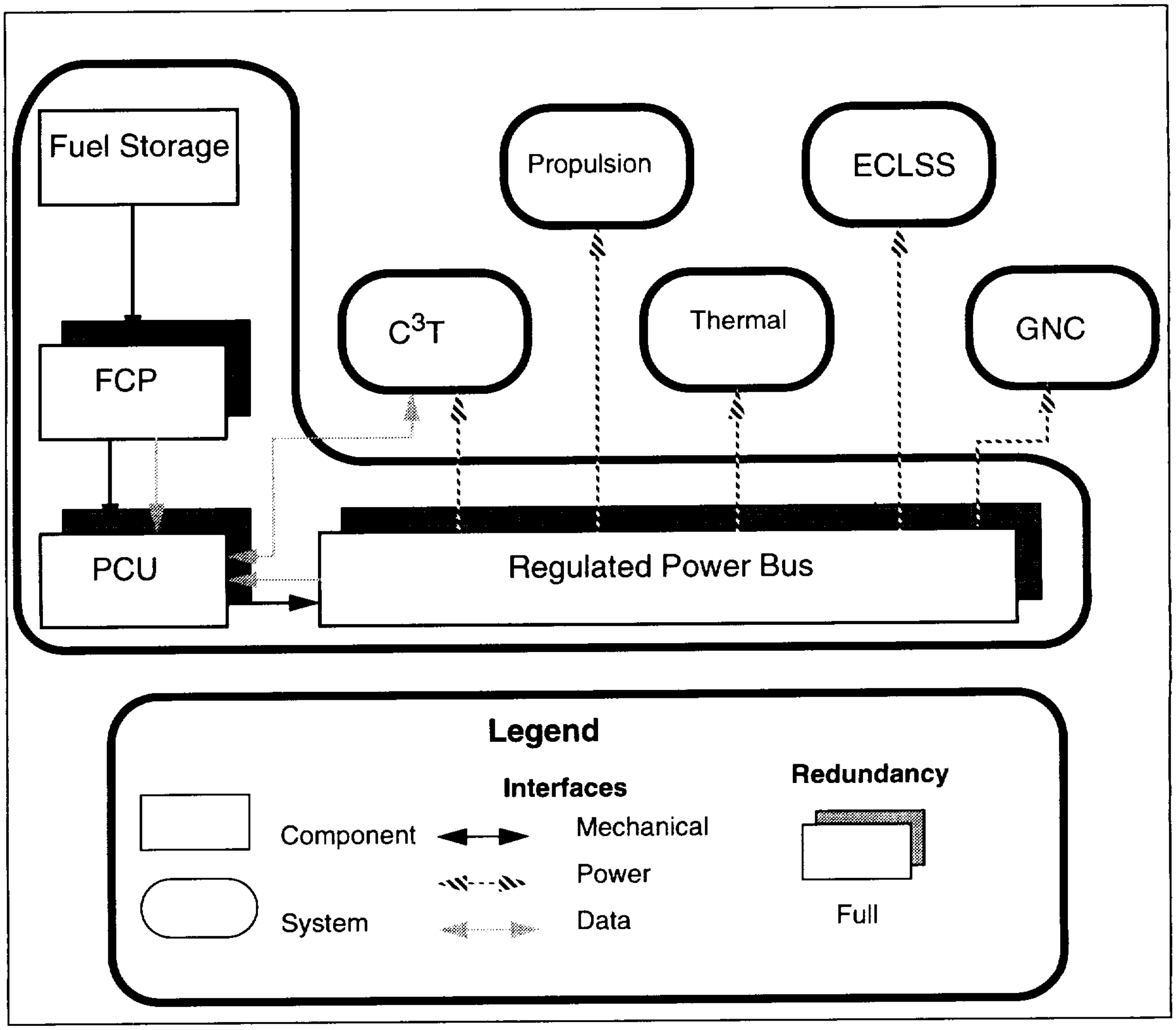

FIGURE 4.1 *Electrical Power System Functional Block Diagram.*

4.3 Power Source

Three major power sources were considered for Project Freebird: batteries, solar cells, and fuel cells. Nuclear power was eliminated because of developmental dangers, high initial costs and qualification difficulties.

Batteries

Batteries fall into two categories, primary and secondary. Primary batteries are single use batteries that may last up to a few months. Since the vehicle will be operating for several years, this option was discarded. Secondary batteries have recharging capability and could last the duration of Freebird's lifetime, however they are very bulky (3-5 times heavier than fuel cells) [1].

Solar Cells

Solar cells generate power in a variety of ways. Photovoltaic solar arrays convert sunlight directly to energy, while dynamic solar arrays use sunlight to power thermodynamic cycles such as Rankine or Brayton cycles.

These arrays are either panel-mounted or body-mounted. In the panel mounted configuration, solar cells are attached to a metallic support structure and insulated with a polyimide. This array is commonly used on three-axis stabilized systems which maneuver the panels to achieve optimum solar incidence. Body mounted arrays are generally used on spin stabilized systems that have their spinning axis perpendicular to the incident light.

Panel arrays were discarded for Freebird because of the g-loads associated with orbital transfers and inclination changes. Unlike satellites who deploy their arrays once, Freebird would have to store the array before every engine burn. Additionally, the arrays would increase the complexity of EVA's and docking maneuvers.

Body mounted arrays were discarded because of their severely limited power output. Since the spacecraft is not spin stabilized, solar cells attached to the body do not see consistent sunlight. As a result, there is not enough available surface area on the spacecraft for the necessary number of cells [2].

Fuel Cells

Fuel cells combine gaseous oxygen and hydrogen to generate electrical power, potable water, and heat. Fuel cell powerplants (FCPs) were developed by United Technologies Corporation in the early 1970's and are the Space Shuttle's sole power source. The benefit of the FCP is that it requires no additional power storage. A given load leads to a release of hydrogen and oxygen from the storage tanks. In this way power can be varied between 2kW and 12kW nominally and up to a maximum of 16kW. FCPs have flight proven reliability and are qualified for the 100 mission lifetime of the Space Shuttle. Lifetime is limited by electrode wear, currently 1200 hours. Present technology has demonstrated test cells that have lasted 20,000 hours and are projected to last 40,000 hours. If qualified for this, the fuel cells will outlast the spacecraft and will not need to be replaced [1][3].

Power Source Selection

Our mission type and duration encourages the use of fuel cells. Freebird has a maximum power demand of 4500W, so the fuel cells will be scaled down versions of the Space Shuttle FCPs, providing 1 to 6kW of power nominally. Table 4.1 details the Space Shuttle FCPs. One FCP will provide enough power for any mission with a second FCP providing redundancy in the case of failure. The fuel cell propellant tanks are detailed in Table 4.2.

TABLE 4.1 *Space Shuttle Orbiter FCP Parameters[3]*

Parameter	Value
Power output	1 to 6kW
Voltage regulation	27.5 to 32.5 V D.C.
Mass	77.1 kg
Max heat rejection	0.7 kW @ 1-kW output, 3.7 kW @ 6-kW output
Max water production	0.1 g/s @ 1-kW output,0.6 g/s @ 6-kW output

TABLE 4.2 *Fuel Cell Propellant Tank Parameters [3]*

Parameter	LOX	LH_2
Geometry	Spherical	Spherical
Volume	$0.16m^3$	$0.3m^3$
Max fill quantity	150 kg	25 kg
Mass	31.8 kg	34 kg

Future Power Technology

Regenerative fuel cells (RFCs) are a possibility for a future power source. RFCs are similar to regular fuel cells except they have an electrolyzer which breaks water back into hydrogen and oxygen so it may be reused. The power for the electrolyzer could come from a source in the Vehicle Storage Facility that Freebird from when not in service.

4.4 Regulation

Techniques for controlling bus voltage are classified into three categories: Unregulated, quasi-regulated, and fully regulated.

Regulation Configuration

An unregulated power bus connects the spacecraft loads directly to the voltage being generated by the fuel cells. This voltage is not constant and requires regulators at each load interface. This method is not mass efficient, requires more parts that may possibly fail and therefore was eliminated from consideration. Quasi-regulated systems are related to battery-powered vehicles. Thus, a regulated power bus was chosen for Freebird [2].

EPS Architecture

The regulated EPS is depicted in Figure 4.2. External redundancy is found in the primary components: fuel cells, the power control unit (PCU), and the power bus. Shunt resistors serve to vent excess power and make the system simpler and more efficient. The PCU is the main component for regulation, control, and telemetry of the EPS.

4.5 Distribution

The distribution aspect of the EPS involves several key functions: providing a wire harness that will bring power to all the electrical applications, converting the common bus voltage to the required level for each application, shielding the electrical cables from sensitive components, and detecting and isolating any faults that might occur in the entire system.

4.5.1 Wire Harness

The wire harness consists primarily of cabling between the elements of the EPS: the fuel cells, the power control unit, and the main power bus. At the location of every load there is an outlet from each of the redundant power busses that interfaces with the voltage converter, if necessary. This scheme can be seen in Figure 4.2. Although the wire harness is kept as short as possible to avoid power losses and voltage drops, it still constitutes approximately 15% of the EPS mass, or about 100kg.

4.5.2 Voltage Conversion

The FCP provides power in a voltage range of 26.5 V to 32 V DC. As a function of the PCU, the common bus voltage is regulated to be 28V DC the current standard for low-power spacecraft system components. Those components operating on a different voltage require a simple voltage converter or inverter which is located between the power bus outlet and the application itself. These converters also function to filter out any excessive noise found on the power bus for more sensitive load components. This arrangement indicates a centralized power distribution system which allows for increased modularity and future modifications to the EPS [5].

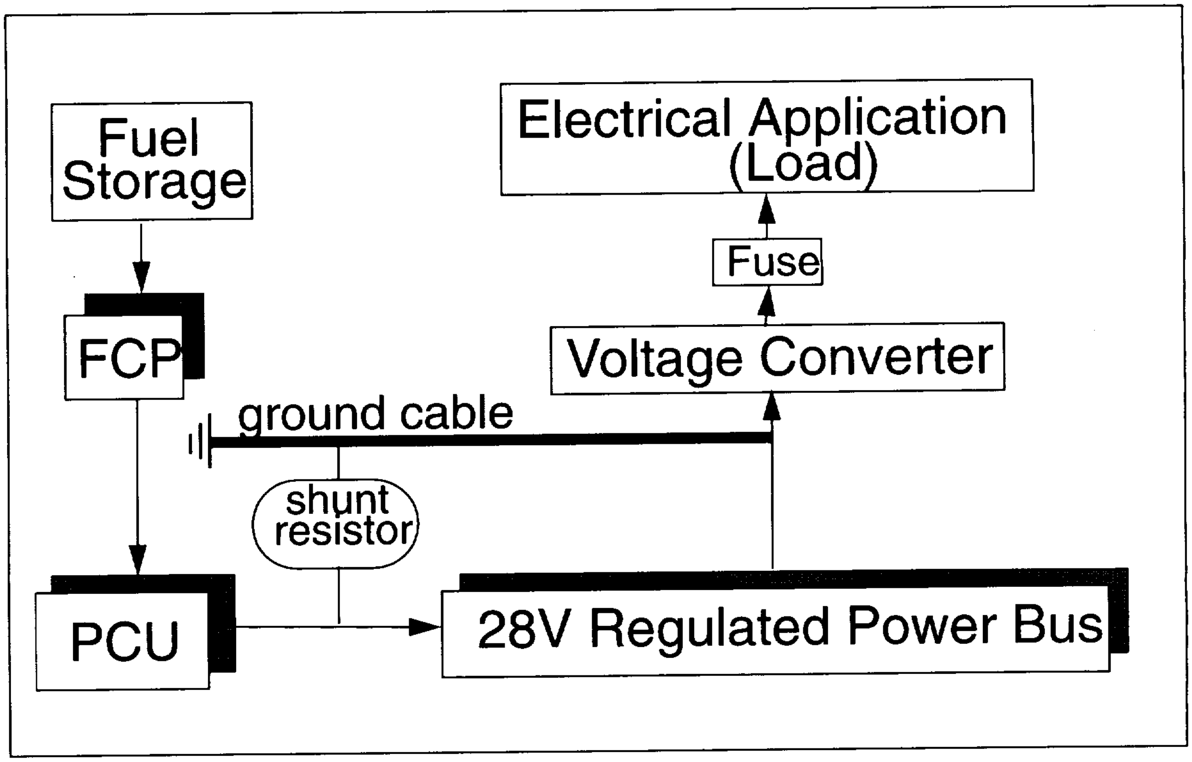

FIGURE 4.2 *Electrical Power System Architecture.*

4.5.3 Fault Detection and Isolation

Faults in the EPS include failed loads (short circuits), arcs, and EPS component failure. A system of resetable fuses is incorporated into the positive power line between the power bus and the loads in order to isolate a failed load and thereby protect the remainder of the EPS. A well-isolated ground cable serves to suppress any potential arcing due to voltage differential. This method is superior to grounding via structure because it eliminates the need for high continuity between structural elements. Finally, in the case of failure of EPS components themselves, all primary elements of the system have redundancy. Telemetry of the EPS is extensive, and all faults are detected and reacted to immediately [5].

TABLE 4.3 *Failure Modes*

Component	Failure Mode	Effect	Prevention or Response	Criticality	
				Crewed Config.	Teleop.
Fuel Cell (FCP)	Primary FCP failure	Loss of FCP redundancy	Switch to backup FCP	1	2
	Backup FCP failure	Loss of vehicle power	None		

TABLE 4.3 *Failure Modes*

Component	Failure Mode	Effect	Prevention or Response	Criticality: Crewed Config.	Criticality: Teleop.
FCP fueling system	Fuel line blockage	Loss of vehicle power	High reliability	1	2
Power Control Unit (PCU)	Primary PCU failure	Loss of PCU redundancy	Switch to backup PCU	2	2
	Backup PCU failure	Loss of voltage regulation and telemetry	High reliability Abort mission		
Power Bus	Primary bus failure	Loss of bus redundancy	Switch to backup power bus	1	2
	Backup bus failure	Loss of vehicle power	None		

4.6 Power Demands

The power required by Freebird's systems varies immensely throughout an average mission. The actual mission scenario and its duration are the primary drivers of the total required fuel for FCP usage. Basically, any Freebird mission can be split into several distinct regions of operation: stand-by, orbital alignment and separation, cruise, service, re-alignment and cruise, and docking.

4.6.1 Stand-by

While docked near ISSA, Freebird will not draw any electrical power from it except in emergency situations. This is mandated so as to not encumber the design and implementation of ISSA with any unexpected requirements.

4.6.2 Mission Power Allocation

The two primary missions that Freebird accomplishes are service and tending flights to LEO and to GEO. While the LEO missions require much less propulsion and time, the fact that they are manned means a great deal of electrical power is required for ECLSS activities. However, the most power-intensive functions of Freebird will be performed separately so as to avoid overdriving and possibly damaging the FCPs.

4.7 Summary Budgets

The most massive part of the power system is the fuel cells and their associated fuel tanks, being about 210kg. Trades were made between cost and efficiency of power supply and storage components. The mass and power budget is shown in Table 4.4.

TABLE 4.4 *Summary Budgets*

Component	#	Mass [kg]	Avg. Power [Watts]	Peak Power [Watts]	Hardware Cost [k$]
Fuel Cell Powerplant	2	154.2	0.0	0.0	4900
Power Conditioner	2	50.0	0.0	0.0	1590
EPS Wire Harness	2	100.0	0.0	0.0	920
Fuel Cell Hydrogen Tank	1	34.0	0.0	210	330
FCP Hydrogen	1	14.6	0.0	0.0	variable
Fuel Cell Oxygen Tank	1	31.8	0.0	1000	308
FCP Oxygen	1	122.3	0.0	0.0	variable
Total	---	507.0	0.0	---	8098

The telemetry of this system is outlined in Table 4.5

TABLE 4.5 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital
Fuel Cells (x2)	LOX Tank Density	1	1	A
	LOX Temperature	1	1	A
	LH2 Tank Density	1	1	A
	LOX Temperature	1	1	A
	Heater Status	1	1	D
	Coolant Pump Status	1	1	D
	Coolant Temperature	1	1	A
	H2 Pump Status	1	1	D
	Power Load	1	1	A

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Chapter 5

Propulsion

This section will detail the propulsion system requirements, interfaces, option trades and selections, propellant storage issues, and the reaction control system.

5.1 Propulsion System Requirements

The requirements for this system were determined from the overall vehicle requirements.

- The main engines must provide a worst case ΔV of 10,386 m/s.
- The main engines must be throttleable and refuelable.
- The vehicle must not exceed 3g of acceleration.
- The system must last over Freebird's 10 year lifetime.
- The main engines must be maintainable and replaceable.
- The reaction control system (RCS) must provide maneuverability and attitude control for Freebird.
- The system must meet NASA crewed and uncrewed safety and reliability requirements.

5.2 Propulsion Interfaces

The primary components of the propulsion system are the main engines, propellant storage, and RCS. The interfaces, both between components and subsystems, are shown in Figure 5.1, the functional block diagram.

5.2.1 Propulsion Trade-offs

The primary propulsion system is responsible for providing the thrust for orbit transfers and inclination changes. Several different types of engines, using a variety of fuels were investigated. The mission requirements necessitated an engine with restart capabilities, high specific impulse and a thrust level on the order of

100kN. These engine capabilities assure a minimum trip time and maximum engine lifetime. Solid propulsion was not used due to its inability to be restarted, and electric propulsion was not used because of its low thrust levels.

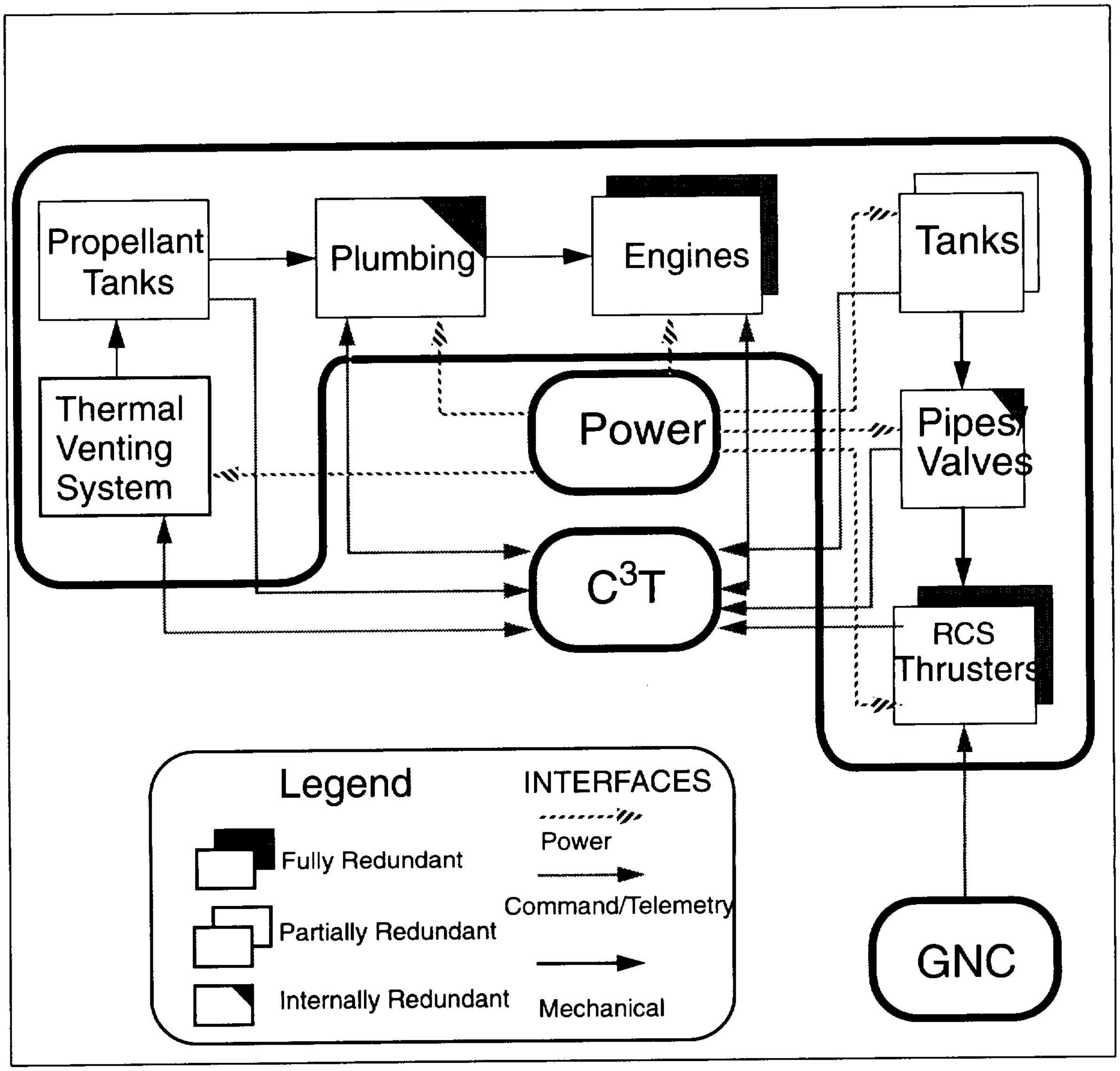

FIGURE 5.1 *Propulsion System Functional Block Diagram.*

5.2.2 Hypergolic Fuel

Hypergolic fuel consists of two inert liquids which ignite spontaneously when combined. It is safe to handle and can be stored in space for long periods of time. The one common type of hypergolic fuel is nitrogen tetroxide and Monomethyl hydrazine which has a specific impulse of approximately 340 seconds [1].

5.2.3 Cryogenic Fuel

Cryogenic fuel refers to liquid hydrogen (LH2) and its oxidizer, liquid oxygen (LOX). This type of fuel has the benefits of a very high specific impulse (~450 Isp), but is dangerous to handle and rapidly boils off in space without insulation. Because of its short lifetime, cryogenic fuel is primarily used on launch vehicles. However, with insulation and a sophisticated venting system (see "Propellant Storage" on page 91), it may be stored for several months. Because this length of time is acceptable for Project Freebird and because it has a significantly higher specific impulse than hypergolic fuel, cryogenic fuel was chosen for the main propellant.

5.2.4 Engine Selection

Most cryogenic engines are high thrust boosters, and are typically started only once. The KUD-7.5 Russian engines are qualified for six starts, the second highest number of starts of all cryogenic engines, and have a thrust level within the necessary specifications. These engines could be used for one or two missions. However, since Freebird will need to make many missions between engine replacements, they are not satisfactory to use in the Freebird design.

The RL10A-4 engines by Pratt and Whitney are qualified for 20 starts and 3000 seconds and have a thrust level of 92kN. With further testing, Pratt and Whitney believes these engines can be qualified for 100 starts and 5.25 hours. This qualification will extend the engine's lifetime to approximately two years. Thus, with the current rated thrust and the new lifetime qualifications, these engines were chosen for the Freebird design. [2,3]

5.2.5 Engine Placement

The design driver for engine number and placement is minimum burn time given a 3g envelope; this maximizes engine lifetime and minimizes trip time. The propulsion pack contains four engines, arranged in a box configuration. This design provides a 0.5-1.5g initial acceleration, depending on the mission, which is well within the 3g limit. This acceleration will increase, however, as fuel is burned, but two of the engines may be turned off once the acceleration reaches the limit. The highest acceleration with 2 engines is 1.9g.

The four engine design also allows redundancy in design. If one of the engines were to fail, then its opposing engine could be shut off and the vehicle could operate with the two remaining engines, increasing the mission time but insuring completion

5.3 Propellant Storage

Freebird's main propellant storage tanks are capable of storing LH_2 and LOX in orbit for extended durations with a loss rate of only 2-5% per month. The system also provides necessary pressurization and stratification controls to provide continuous propellant flow to the turbopumps of the RL10A-4 engines during firing.

5.3.1 Tank Geometry

The main propellant tank layout for Freebird includes two spherical tanks, one for LH2 and one for LOX. The larger LH2 tank is located directly in front of the LOX tank to maximize safety and storability and minimize volume requirements. The main factors in tank geometry determination were launch vehicle dimensional and volumetric constraints and the remote manipulation system (RMS) zones of operation.

A propellant mass ratio of 5 for the base unit requires the main tanks to contain 20 000 kg of propellant. The RL10A-4 engines function on an oxidizer-to-fuel ratio of 5.5. This defines the propellant volume demands as seen in Table 5.1 and Figure 5.2.

TABLE 5.1 *Main Propellant Tank Parameters.*

Parameter	LH_2	LOX
Geometry	Spherical	Spherical
Propellant Mass [kg]	3076.9	16923.1
Density[kg/m^3]	75.8	1226.3
Volume [m^3]	40.59	13.80
Radius [m].	2.132	1.49

In addition to the main tanks, add-on propellant tanks are used to extend the range of Freebird to polar and geosynchronous orbit (GEO) missions. Two pairs of LH2 and LOX tanks with fuel line interfaces attach to opposite sides of the propulsion unit. The Extended Mission Propellant Tanks (EMPTs) are launched into Freebird's stand-by orbit a few days to a week before the intended mission. Freebird then retrieves and attaches the tanks at the beginning of the extended mission. As a result, the extended tanks are not insulated for long term, in-orbit storage of more than a week or two. Additionally, the EMPTs are not refuelable, and are jettisoned after propellant is consumed and mission orbit is achieved. The parameters for the extended mission tanks are generated the same as for the main tanks with the goal of a mass fraction of 10. See Table 5.2.

TABLE 5.2 *Extended Mission Propellant Tank Parameters.*

Parameter	LH_2	LOX
Geometry	Spherical	Spherical
Propellant Mass [kg]	2307.7	12692.3
Density[kg/m^3]	75.8	1226.3
Volume [m^3]	30.44	10.35
Radius [m]	1.937	1.352

5.3.2 Tank Construction

Based on size and shape requirements, the construction material and design must be chosen that will withstand propellant pressure levels, inertial loads, and the demands of cryogenic propellants. Maximum pressure loads using a factor of safety (FOS) of 1.4 are determined using

$$P_{total} = 1.4(P_{pressurized} + P_{inertial}) \quad \textbf{(Equation 5.1)}$$

$$P_{inertial} = 12\rho g r \quad \textbf{(Equation 5.2)}$$

where P_{total} is the total pressure load, $P_{pressurize}$ is the pressure load due to the gas, $P_{inertial}$ is the pressure loads induced by acceleration, ρ is density, g is the acceleration due to gravity, and r is the radius of the tank. The pressure for LH2 and LOX are listed in Table 5.3.

With the pressure loads known, the thickness and mass of the tanks can be easily calculated once a construction material is chosen. Materials were compared on the basis of tensile strength, density, and fracture strength. As seen below in Table 5.4, T50/E composite was chosen. In addition to the above criteria, suitability for containing cryogenic liquids was a concern. Al-6061, AISI-301, and composite liner are acceptable. Due to the strength and mass qualities, composite materials[9] with a 1 mil titanium liner was chosen. Although slightly

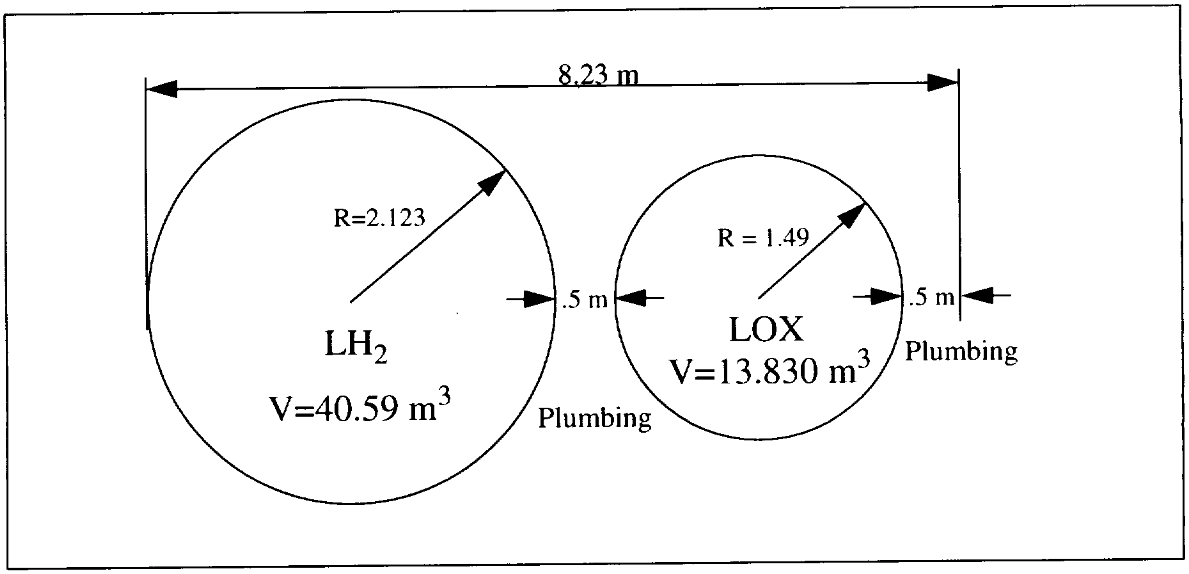

FIGURE 5.2 *Main Propellant Tank Dimensions.*

TABLE 5.3 *Main Tank Pressure Load.*

Parameter	LH_2	LOX
$P_{pressurized}$ [Pa]	187900	187900
$P_{inertial}$ [Pa]	38000	510100
P_{total} [Pa]	316200	689000

heavier than other potential liners, titanium was chosen for its resistance to oxidation[10]. With a material chosen, thickness and tank mass can be calculated with

$$t = rP_{total}/2\sigma \quad \text{(Equation 5.3)}$$

$$m_{tank} = 1.5(\rho/\sigma)P_{total}V_{tank} \quad \text{(Equation 5.4)}$$

where t is the tank wall thickness, σ is the critical stress of the material, m_{tank} is the mass of the tank, and V_{tank} is the volume of the tank. Table 5.5 lists the resulting thicknesses and tank masses.

TABLE 5.4 *Tank Wall Construction Material Options - from Mil-Std Handbook and Lockheed.*

Parameter	7075-T6	6061	17-7PH	AISI-301	T50/E
Description	Aluminum	Aluminum	Steel	Stainless Steel	Composite
ρ [kg/m^3]	2796	2713	7640	7917	1578
σ [MPa]	496.4	241.3	1151.4	979.1	303.4

TABLE 5.4 *Tank Wall Construction Material Options - from Mil-Std Handbook and Lockheed.*

Parameter	7075-T6	6061	17-7PH	AISI-301	T50/E
ρ/σ [kg/Pam^3]	5.63×10^{-6}	1.12×10^{-5}	6.63×10^{-6}	8.09×10^{-6}	5.20×10^{-6}
Fracture Strength	low	high	low	high	very high

5.3.3 Tank Pressure Control

Pressure control of the main tanks is vital to Freebird's propulsion systems. Excess pressure can result in tank failure and rupture, and lack of pressure can restrict propellant flow during firing. Excess pressure is released via a venting system. While firing, a small amount of propellant is heated and used to pressurize the tanks.

TABLE 5.5 *Main Tanks Wall Thickness and Mass.*

Parameter	LH_2	LOX
t [m]	.0011	.0017
m_{tank} [kg]	107	79

Pressurization

A small amount of regeneratively heated H_2 and O_2 from the RL10 engines is returned via a regulated system into the propellant tanks. This maintains operating pressures throughout firing.

Venting

LH2 and LOX are stored at approximately 20K and 70K, respectively. Since ambient temperatures are significantly higher, heat energy enters the propellant tanks. As the cryogenic propellants warm up, more propellant becomes gaseous and tank pressure rises. The excess gaseous propellant must be vented to control pressure and prevent tank failure.

However, Freebird was designed to remain in orbit, fueled and ready to go, for extended periods of time on the order of months. In order to accomplish this, Freebird uses a Thermal Venting System (TVS)[4] which minimizes the amount of propellant vented and maximizes the amount of heat energy expelled.

The TVS works by cycling some of propellant through a heat exchanger and injecting it into the tank with a spray bar. As Figure 5.3 shows the warm propellant is pumped out of the tank and into the heat exchanger located inside of the spray bar. Some of the warm propellant is expanded, thereby cooled, and the rest flows up the core of the heat exchanger. As both propellants travel up the heat exchanger, heat is transferred and concentrated in the expanded flow. After the warm propellant is cooled, it enters the spray bar and is injected radially back into the tank. The expanded and heated gas is retained until the maximum desired pressure level is reached and then vented.

Advantages of this system in addition to venting:

- The spray bar injection system thermally destratifies the bulk propellant preventing hotspots and making propellant level easily determined.
- The pumping system doubles as the refueling pump system reducing weight demands.

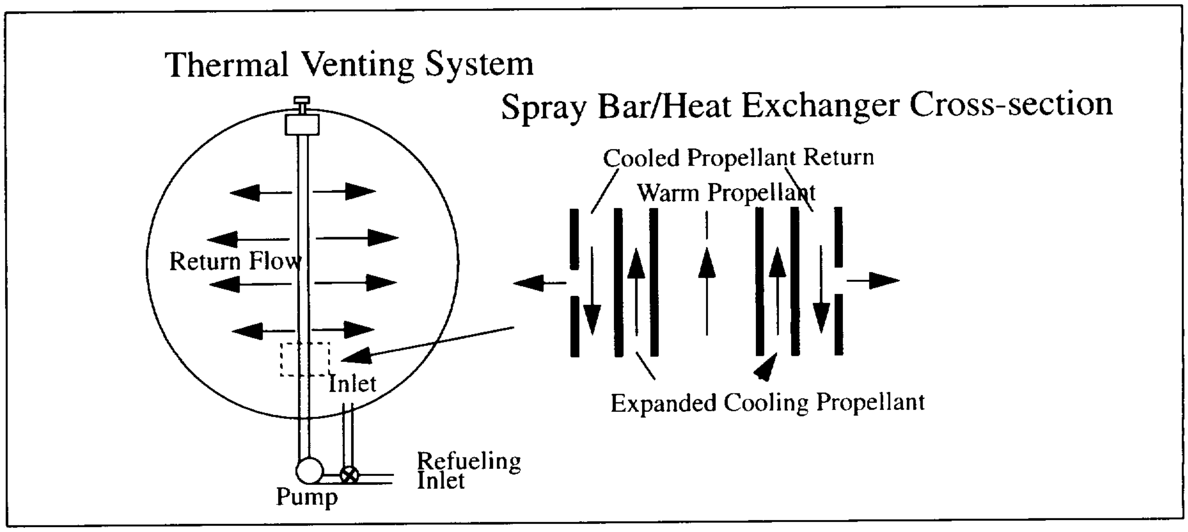

FIGURE 5.3 *Thermal Venting System (TVS) Operational Diagram.*

5.3.4 Propellant Feed System

The RL10 engines use turbopumps which require a low constant pressure, gas free flow of propellant. One of the complications of on orbitstorage and operation of liquid propellant systems is the lack of gravitational force to separate the liquid and gas ullage in the propellant tanks. The presence of gaseous propellant in the turbopumps during the operation can both damage the turbopumps and cause the engines to sputter. Since both results seriously endanger a mission, it is absolutely necessary to guarantee uninterrupted, gas free propellant flow.

The primary method of ensuring continuous flow is the use of Propellant Management Devices(PMD)[6],[7] shown in Figure 5.4. In zero-g the affects of surface tension in a liquid become dominant. By taking advantage of capillary forces created by the surface tension, a combination of vanes, galleries, and screens can be used to retain the liquids and funnel the propellant into the feed lines (see Figure 5.5). Since this method conveys only fluids, it separates the fluid from ullage gas and prevents gas from entering the feed lines. In addition to the PMDs, the acceleration forces will force the propellant to the bottom of the tanks providing continuous flow throughout burntime.

PMDs have been commonly constructed from titanium, stainless steel, and aluminum elements. Employing recent developments in titanium screen manufacturing [5], Freebird will use a completely titanium PMD system making it resistant to corrosion by cryogenic propellants and compatible with the titanium liner.

The PMD will also serve the secondary purpose of reducing sloshing. The normal damping effects that result from gravity are severely reduced, and as a result it becomes necessary to induce damping. Careful design of the PMD can improve liquid damping and reduce the affects of sloshing.

5.3.5 Plumbing

A complicated system of pipes and valves transfer the propellant from the tanks to the engines' turbopumps. In addition to the components of the TVS, fuel lines for the main and extended mission tanks, refueling lines, and emergency pressure relief valves are all necessary.

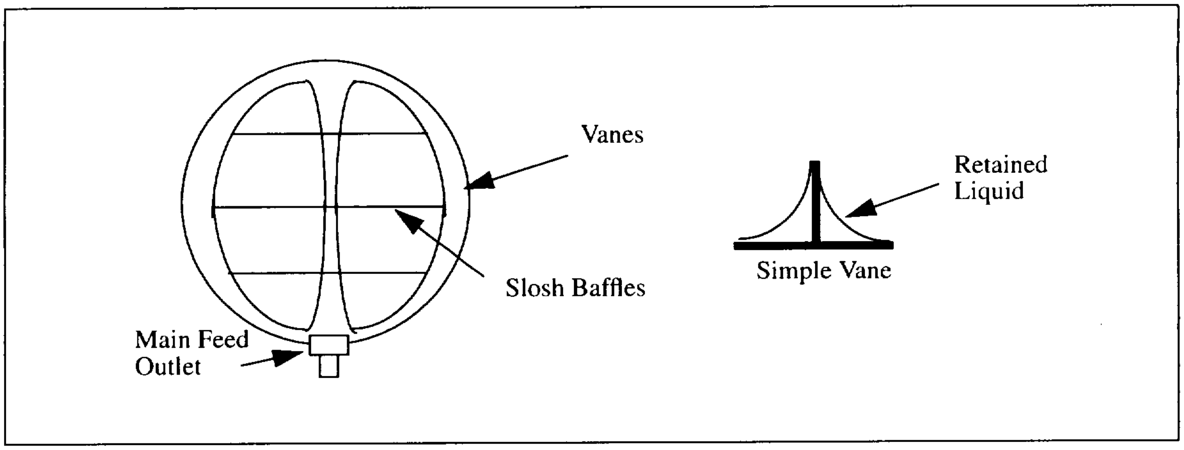

FIGURE 5.4 *Typical Propellant Management Device (PMD) Design.*

Although not a major volume consideration, these elements add mass and thermal considerations. Mass is minimized using low weight/high strength materials like stainless steel and limiting total length of lines. However, every penetration through the insulation and tank wall allows heat to pass into the cryogenic propellants and results in an increase in boil-off rate rates.

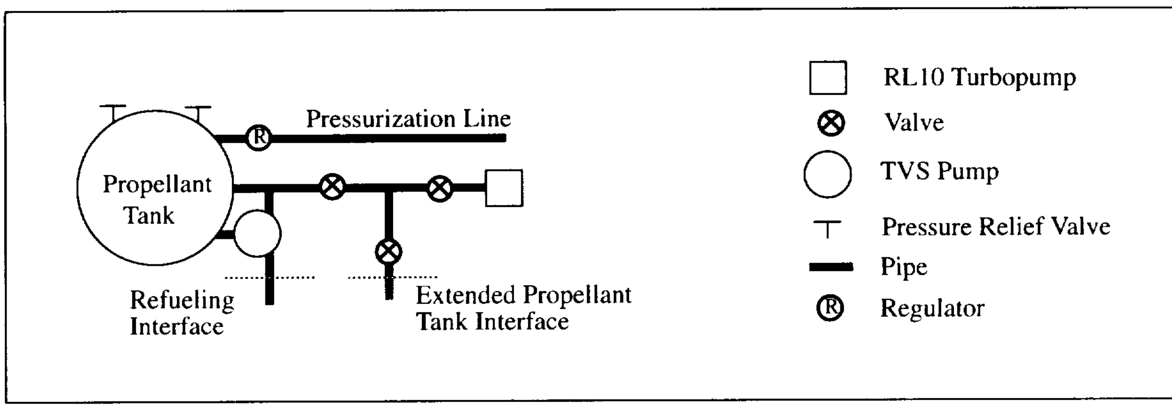

FIGURE 5.5 *Plumbing Block Diagram.*

The major factor in reliability and lifetime of the plumbing system is valves. Since the interruption of propellant flow can be mission critical, the valves controlling the propellant flow must be reliable. As a result, all essential valves will be redundant to improve reliability.

5.3.6 Refueling and Extended Mission Propellant Tanks Interfaces

Since Freebird is a multi-mission, reusable spacecraft, the propellant storage has an interface to allow refueling in orbit. After rendezvousing with the launched fuel, Freebird can initiate a propellant transfer by grabbing the free floating propellant tanks and connecting the fill lines to the refueling interface[8]. See Figure 5.6.

The refueling tanks are single use, relatively uninsulated stainless steel tanks with an inflatable bladder for pressurization. These tanks will most likely be launched on a Proton Launch Vehicle.

On opposite sides of the propulsion module, two Extended Mission Propellant Tank interfaces can be found. These interfaces based on the same connecting technology as the refueling lines will integrate the additional LH_2 and LOX tanks with the original propellant feed system. In addition, the interface will include separation springs to allow ejection of the extended mission tanks after they have been emptied.

5.3.7 Attitude Control and Maneuverability

Attitude control and maneuverability for Freebird is supplied by a system of reaction control thrusters located on the propulsion module. The thrusters will be replaced with the propulsion module. The configuration of these thrusters can be seen in Section 7.7 on page 143. The system consists of two sizes of thrusters, two orbital maneuvering thrusters and twenty-six reaction control thrusters.

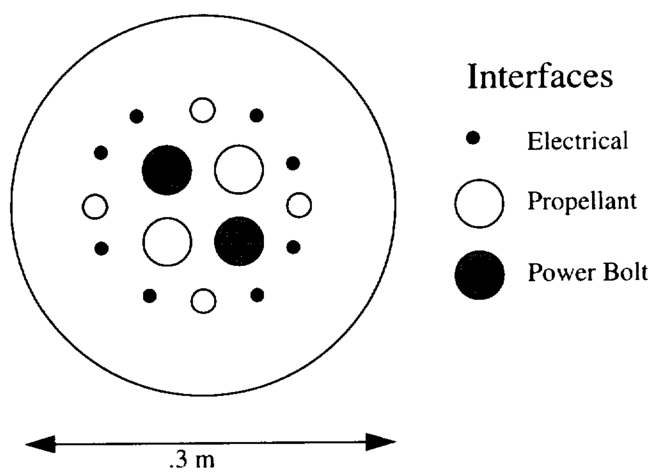

FIGURE 5.6 *Propulsion Module Interface Design.*

5.3.8 RCS Thrusters

When choosing thrusters the important considerations are thrust, Isp, lifetime, mass, safety and contamination. There are three main types of thrusters used for reaction control. First are cold gas thrusters. These are safe, reliable, and cheap. However they are heavy, and the thrust provided is extremely small. Monopropellant thrusters are also simple, reliable and low-cost. These are also fairly low thrust, low Isp, and heavier than the third option, bipropellant thrusters. The bipropellant thrusters most commonly used in RCS applications are nitrogen tetroxide and monomethyl hydrazine (N2O4 and MMH). These fuels are storable and the thrusters provide good performance, but they are somewhat complicated.

It has been demonstrated that plume loads are a more important concern than exhaust contamination. Therefore, contamination need not be an issue in choosing a system. However, this stipulates that the thrusters on the order of magnitude required for Freebird cannot be fired within about 20m of an asset [12].

Because of Freebird's size, the demands put on the orbital maneuvering thrusters require a bipropellant system. The specifications of the thrusters chosen are shown in Table 5.6.

TABLE 5.6 *Reaction Control Thrusters*

	Orbital Maneuvering Thrusters [2]	Reaction Control Thrusters [11]
Thrust [N]	3870	490
Isp [s]	281	311
Mass [kg]	10.25	3.7
Approximate size [cm x cm]	51.8 x 26.7	61.0 x 30.5
Burn time [s]	15,000	50,000

5.3.9 RCS Fuel Storage

The RCS system operates off two sets of tanks, one half of the thrusters running off each tank in normal usage. In case of a problem with one of the tanks, the thrusters are connected to both tanks. This ensures that all thrusters can be used, albeit in a limited manner because of the limited fuel.

The RCS tanks are made of stainless steel because of the highly reactive nature of the propellants. They are pressurized via a regulated cold gas system. This gas must be replenished when the RCS is fueled.

The rendezvous requirements on Freebird require a large amount of RCS fuel. This stipulates that for a mission to GEO, RCS fuel must be provided on the extended mission fuel tanks.

5.4 Failure Modes

The failure modes for the propulsion components are defined in Table 5.7

TABLE 5.7 *Failure Modes*

Component	Failure Mode	Effect	Prevention and/or Response	Criticality	
				Crewed Config.	Teleop.
LOX Propellant Tank	Rupture	Loss of Primary oxidizer	High reliability, backup relief valve	1	2
LH2 Propellant Tank	Rupture	Loss of primary propellant	High reliability, backup relief valve	1	2
Valves	Disrupt fluid flow	Loss of fuel feed	Redundancy	1	2
PMD	Structural failure	Unpredictable engine starts	High reliability, Return to ISSA	2	2

TABLE 5.7 *Failure Modes*

Component	Failure Mode	Effect	Prevention and/or Response	Criticality	
				Crewed Config.	Teleop.
TVS	Pump failure	Loss of thermal control over propellants	Redundant Pump	1	2
Interfaces	Structural failure	Refueling capability lost	High reliability	2	2
RL10A-4 Engine	Failure of one pair of engines	Decreased thrust	Redundancy	1	2
	Failure of all four engines	Loss of main propulsion capability	High reliability		
RCS Tanks	Rupture	Loss of fuel -- maneuverability limited	Return to ISSA	1	1
	Rupture of both sets	Loss of maneuverability	Return to ISSA		
RCS valves	Disrupt fluid flow	Loss of RCS fuel feed	Redundancy, return to ISSA	1	2
Orbital Maneuvering Thrusters	Mechanical failure	Reduced maneuverability	Utilize RCS thrusters, may necessitate return to ISSA	2	2
RCS Thrusters	Mechanical failure	Reduced maneuverability	Redundancy	2	2

5.5 Summary Budgets

The budgets for this system are itemized by component in Table 5.8

TABLE 5.8 *Summary Budgets*

Component	#	Mass [kg]	Avg. Power [Watts]	Peak Power [Watts]	Hardware Cost [$K]
TVS	2	60.0	N/A	N/A	2,540
Valves and Pipes	1	230.0	N/A	N/A	3,347
Totals	---	2064.6			29,525

TABLE 5.8 *Summary Budgets*

Component	#	Mass [kg]	Avg. Power [Watts]	Peak Power [Watts]	Hardware Cost [$K]
LH_2 Propellant tank	1	107.0	N/A	N/A	1,698
LOX propellant tank	1	79.0	N/A	N/A	1,254
PMD	2	40.0	N/A	N/A	582
Interfaces	2	60.0	N/A	N/A	2,328
RL10A-4 Engines	4	672.0	N/A	N/A	12,000
Orbital Maneuvering Thrusters	2	20.4	N/A	N/A	758
RCS Thrusters	26	96.2	N/A	N/A	3,563
RCS Tanks, valves and pipes	1	150.0	N/A	N/A	1,455
RCS Fuel	1	550.0	N/A	N/A	---
Totals	---	2064.6			29,525

The telemetry data required by the propulsion system is shown in Table 5.9

TABLE 5.9 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/Digital	Command
Main Engines (x4)	Temperature	2	100	A	--
	Pressure	2	100	A	--
	Status	1	100	D	On/Off
	Valve Status	4	100	D	Open/Close
Main Engine Tanks	LOX Tank Pressure	2	10	A	--
	LOX Tank Temp.	2	10	A	--
	LH2 Tank Pressure	2	10	A	--
	LH2 Tank Temp.	2	10	A	--

TABLE 5.9 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/Digital	Command
RCS Engines (x28)	Temperature	2	100	A	--
	Pressure	2	100	A	--
	Status	1	100	D	On/Off
	Valve Status	2	100	D	Open/Close
RCS Fuel Tanks (x2)	MMH Pressure	2	10	A	--
	MMH Temperature	2	10	A	--
	N2O4 Pressure	2	10	A	--
	N2O4 Temperature	2	10	A	--

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Chapter 6

Thermal Control System

The Thermal subsystem is charged with maintaining all equipment and crew members under desired temperature ranges during all mission phases throughout the lifetime of Freebird.

6.1 Thermal Control System Requirements

The thermal system must provide thermal control in order to maintain all components within their thermal operating ranges in all environmental conditions, and maintain the crew in a comfortable temperature range.

Electronics and structure are important in each module. In addition, the crew module must provide a habitable environment for the crew. Also, the smart box must provide thermal control for the fuel cells. Finally, the propulsion pack must provide temperature control for the Reaction Control System (RCS) propellant and insulation for the cryogenic tanks.

To meet the thermal control requirements, the following options were analyzed and their best solution determined.

- Thermal sources and sinks
- Operating ranges of vehicle components
- Thermal control components and their configuration
- Thermal paths

6.2 Thermal Control System Interfaces

Figure 6.1 represents the functional block diagram of the thermal subsystem. The thermal system interfaces with the Command, Communication, Control and Telemetry (C^3T) system by providing data and receiving instructions. The thermometers send their readouts to C^3T, which in turn sends control signals to the relevant components of the thermal subsystem, as shown in Figure 6.1. Also, several thermal components requiring

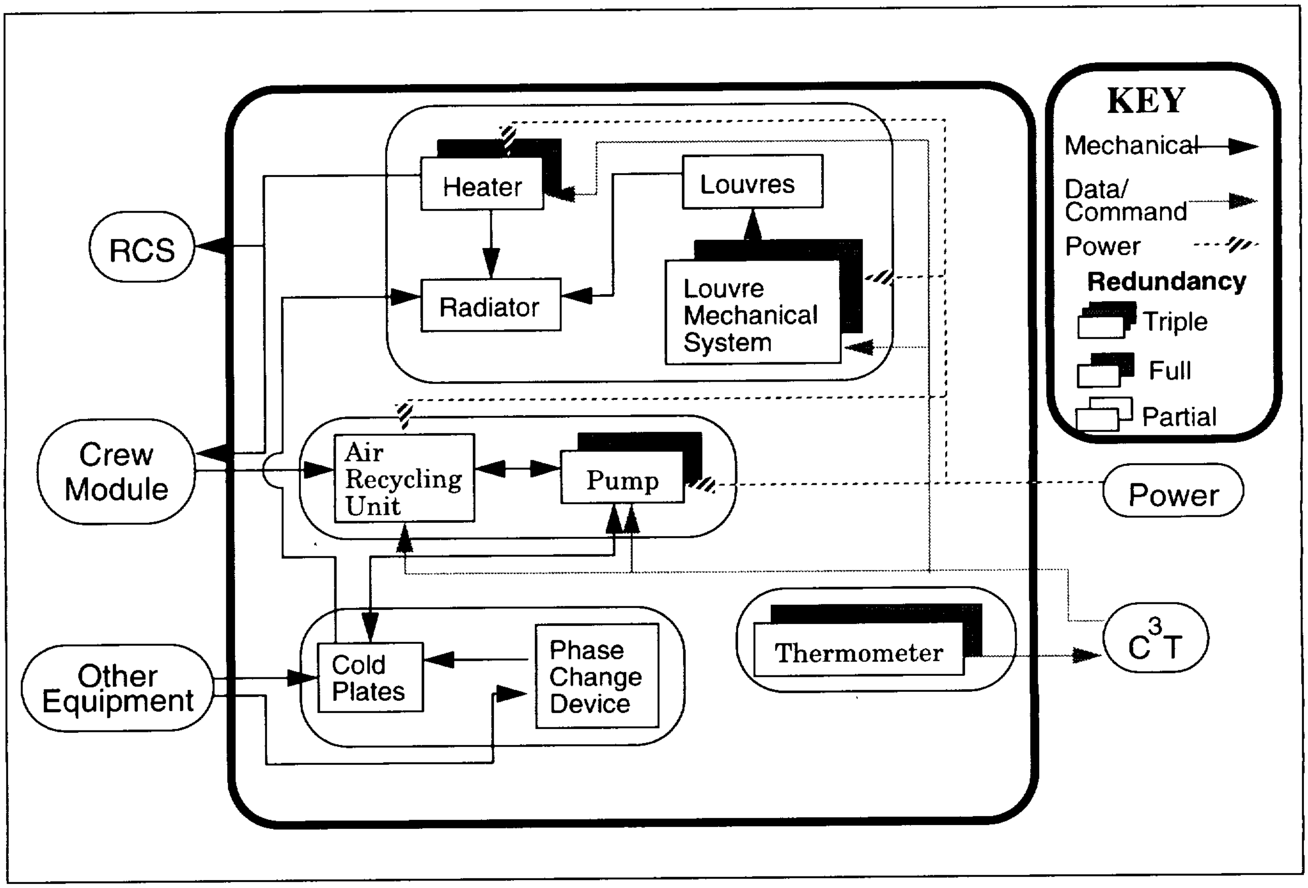

FIGURE 6.1 *Thermal Control System Functional Block Diagram.*

power during some phases of a mission depend on that system to provide it. All other interfaces are physical. That is, the thermal system is responsible for maintaining the other systems within their operating temperature ranges. Therefore, all other interactions are in the form of heat dissipation. The thermal system design was affected by other systems' requirements during the design phase.

6.3 Environment

Freebird's thermal control system works to combine heat fluxes from two separate thermal environments: the external heat flux from the sun and the earth, and the internal heat flux from hardware and crew. The thermal control system described in Section 6.4 on page 107 works to combine these fluxes and to use or reject the heat as efficiently as possible.

6.3.1 External Environment

The external environment provides the spacecraft with three major sources of thermal radiation: direct solar flux, the Earth's albedo, and the Earth's infrared (IR) emissions (see Section 1.4.2 on page 29 for a more detailed description).

- Direct solar flux is the single largest source of external heat flux to Freebird. As long as Freebird is not in Earth's shadow, some part of the vehicle will be exposed to this flux. The amount of direct solar flux absorbed by Freebird is determined by Equation 6.1.

$$Q_{s_a} = G_s A \alpha \qquad \textbf{(Equation 6.1)}$$

where G_s = solar constant = 1358 +/- 5 W/m^2, A = exposed surface area, and α = solar absorptivity =.248.

- Like the direct solar flux, the albedo has no effect on Freebird while it is in shadow. This is the smallest external source of thermal flux. The amount of solar flux reflected from the Earth and absorbed by Freebird is determined by Equation 6.2.

$$Q_{A_a} = G_s a A \alpha K_a \sin^2\rho \qquad \textbf{(Equation 6.2)}$$

where a = albedo = 30% +/- 5% of G_s, K_a = a factor which accounts for the reflection of collimated incoming solar energy off a spherical Earth and is defined in Equation 6.3, ρ = angular radius of Earth, $\sin\rho = R_E/(H+R_E)$, R_E = radius of Earth = 6378 km, and H = altitude of orbit.

$$K_a = 0.664 + 0.521\rho - 0.203\rho^2 \qquad \textbf{(Equation 6.3)}$$

- The flux due to IR radiation absorbed by Freebird is determined by Equation 6.4.

$$q_{I_a} = q_I \sin^2\rho A \varepsilon \qquad \textbf{(Equation 6.4)}$$

where q_I = Earth's IR emission = 237 W/m^2 and ε = IR emissivity =.924.

Table 6.1 gives the values of the external heat fluxes radiated at the positions of the orbits and the amounts actually absorbed by Freebird, based on the solar absorptivity and IR emissivity of the white epoxy surface coating (See Section 6.4.1 on page 107).

TABLE 6.1 *External Heat Fluxes.*

		Direct Solar Flux [W/m^2]		Earth's Albedo Flux [W/m^2]		Earth Infrared [W/m^2]	
		available	absorbed	available	absorbed	available	absorbed
LEO	Earthside	0	0	360	89.5	210	194
(390km)	Sunside	1358	337	0	0	0	0
GEO	Earthside	0	0	5.19	1.28	4.10	3.79
	Sunside	1358	337	0	0	0	0

From the values in Table 6.1, the worst case hot and cold scenarios are determined. The side of Freebird facing away from Earth while the vehicle is in shadow could reach 4 K. However, this will not occur due to the transient effects of orbiting. The side facing the sun out of shadow could get as warm as 283 K, which is cooler than the room temperature maintained inside the crew module. The greatest temperature differential will probably be in geosynchronous orbit (GEO) outside of Earth's shadow, where the side facing the Earth would reach a steady state temperature of 99 K.It is proposed that Freebird periodically rotate to keep the temperatures more equal.

6.3.2 Internal Environment

Freebird's internal environment deals with heat flux originating from crew members and various hardware, including electronics and propellant. The amount of heat flux due to hardware will vary between each of the three craft modules: propulsion pack, "smartbox", and crew module. This is due to the fact that each module requires the implementation of different components.

Heat flux resulting from insulation leaks must also be taken into account for each module. The insulation being used on Freebird is Multi-Layer Insulation (MLI) composed of Aluminized Mylar and Kapton layers (see Section 6.4.1 on page 107). The primary mode of heat transfer through MLI is radiation. Thus, the thermal performance of MLI is described by effective emissivity rather than conductivity. The amount of insulation leakage for each of the three modules can be approximated from the following equation:

$$q = \sigma\varepsilon_{eff}(T_i^4 - T_o^4) \qquad \textbf{(Equation 6.5)}$$

where q is the amount of heat transfer through the MLI blanket per square meter, σ is the Stefan-Boltzmann constant, ε_{eff} is the effective blanket emissivity, T_i is the absolute temperature of the inside surface in Kelvin, and T_o is the absolute temperature of the outside surface in Kelvin.

The value of the effective emissivity, ε_{eff}, is directly related to surface area. Therefore, the effectiveness of MLI is highly dependent upon the size of the insulated object. Because heat transfer is a function of the inside and outside temperature differential, the efficiency of MLI will also vary depending on the orientation of the spacecraft (i.e., how much of the surface area is in direct sunlight as opposed to shade). Other factors having an effect on quantity of insulation leakage are edges, seams, joints, and penetrations such as electrical wires and structural supports.

One of the predominant sources of heat flux in the internal environment of Freebird's crew module is its crew members. Each of the three crew members emits heat into the internal environment at all times. The quantity of this heat flux will vary somewhat depending upon each crew members' level of activity. The amount of heat flux will range from a minimum of 85 W when a crew member is at a low level of activity, to a maximum of 250 W when he or she is at a high level of activity. This heat flux variation is summarized in Table 6.2. Heat contributed to the internal environment from crew members will be greatest when Freebird is involved in a rescue mission. In this worst case scenario, heat will be dissipated by eight crew members for a probable maximum total of 1140 Watts.

TABLE 6.2 *Human Heat Flux Variation. [2]*

	Activity	Power [Watts]
Minimum	Sleep	85
Maximum	Exercise	250
Average	Light Work	120

Table 6.3 summarizes the heat flux in the crew module due to hardware, crew, and insulation leakage. The total average amount of power radiated by hardware and three crew members in this module is 526 Watts. The average heat flux due to insulation leakage from the inside of the crew module towards space, is approximately

83W as determined by the application of Equation 6.5, with an effective emissivity value based on module dimensions equal to 0.0042.

TABLE 6.3 *Cabin Module Heat Flux*

	Maximum Heat Flux [Watts]	Average Heat Flux [Watts]	Minimum Heat Flux [Watts]
Hardware	56	36	36
Crew	1140	490	120
Insulation	-83	-83	-83
Total	1113	443	73

The amount of heat flux in the "smartbox" is determined solely by its hardware and insulation leakage. Table 6.4 summarizes these values for the base module. The total average heat flux from hardware is 200 Watts. The amount of heat that is rejected due to insulation leakage through the MLI blanket has been determined from Equation 6.5. Based on structural surface area, the effective emissivity of the base module is approximately equal to 0.0034. Thus, the average amount of insulation leakage from the base module towards space is 30 Watts.

TABLE 6.4 *Base Module Heat Flux Contributors.*

	Maximum Heat Flux [Watts]	Average Heat Flux [Watts]	Minimum Heat Flux [Watts]
Hardware	798	200	190
Insulation	-30	-30	-30
Total	768	170	160

No equipment in Freebird's propulsion module requires heat dissipation. The propulsion system provides its own cooling. However, the thermal system does provide thermal protection from the external environment in the form of surface coating and insulation for the entire propulsion pack, including the cryogenic tanks.

6.4 Thermal Control System

The thermal control system is charged with maintaining the spacecraft, its components, and its crew within a desired temperature range. This is accomplished through the use of a passive system, an active system, or a combination of both. Passive components include items such as surface coating, insulation, cold plates, and heat pipes. Active components include heaters, pumped cooling systems, air recycling units, louvres, and venting systems. Table 6.5 shows acceptable temperature ranges for the different spacecraft components and the crew.

6.4.1 Passive Thermal Systems

The "smartbox" and propulsion pack will consist mostly of a passive thermal system. Part of the thermal control system for the crew module and the cryogenic tanks will also be passive.

TABLE 6.5 *Temperature Ranges for Spacecraft Components and Crew[3].*

Components	Temperature range, °C
Electronics	0 to 40
Propellant, RCS	7 to 35
Solar Arrays	-100 to +100
Structures	-45 to +65
Crew	18 to 22

Surface Coating

Surface coating is an essential component of passive thermal control. Using an appropriate coating minimizes the heat rejection needed. In all instances, surface coatings are picked so that their equilibrium temperature under exposure to the sun is minimized. Other factors include minimizing weight and cost, and maximizing durability. Table 6.6 lists different common surface materials, along with their steady state temperature when exposed to the sun in an orbit around Earth. The external heat flux value used is 1358W/m^2. Even though a material's absorptivity and emissivity are the same at a single frequency, the solar absorptivity (α_s) and infrared emissivity (ε_{ir}) can be substantially different since the radiation frequencies are not the same in this case. Therefore, the lower a material's α_s to ε_{ir} ratio is, the lower the equilibrium temperature becomes.

TABLE 6.6 *Radiation Properties of Various Materials[3].*

Material	Solar Absorptivity	Infrared Emissivity	Absorptivity to Emissivity Ratio	Equilibrium Temperature (°K)
Aluminum	.2	.034	5.88	613
Steel	.357	.135	2.64	503
Titanium	.766	.513	1.49	435
Gold	.299	.023	13.00	747
Black Paint	.975	.874	1.12	404
White Epoxy	.248	.924	.27	283
Aluminized Teflon	.163	.8	.20	264
OSR	.077	.79	.10	220

The three materials considered were White Epoxy, Aluminized Teflon, and Optical Solar Reflectors (OSR) due to their low steady state temperatures. Aluminized Teflon was discarded due to its mass and OSR was found to be fragile and costly. Therefore, White Epoxy was chosen as the coating material for the spacecraft. The equilibrium temperature of White Epoxy is close to the room temperature that will be maintained inside the crew module. Therefore, the small temperature differential combined with the Multi-Layer Insulation (MLI) will result in minimal heat transfer across the spacecraft walls (See Section 6.3.2 on page 106). It is important to note that this analysis assumes steady state exposure to the maximum external heat flux. During times in an orbit when the Sun is eclipsed by the Earth, the surface temperature of the spacecraft can be significantly less than the room temperature found on the inside wall. Insulation is very effective, but some heat leakage out of the vehicle will nonetheless be present. In order to compensate, the radiators will provide for less heat rejection by use of active devices called louvres. This design to be an easier design to cold-bias the spacecraft in this manner, than to provide for extra heat rejection methods during periods in the orbit when the vehicle is exposed to the sun.

Multi-Layer Insulation

Multi-layer blankets are another essential component of a passive thermal system. As previously discussed, during nocturnal periods of the orbit. The spacecraft's insulation will minimize the heat loss from the inside of the vehicle to the colder outside wall. Also, when the spacecraft is exposed to the Sun there will undoubtedly be non-uniform temperature surface areas on the outside wall. Insulation will prevent the same temperature differentials from existing on the inner walls. If insulation were not present, an astronaut could lean against the inside wall and be burned Only a few feet away the inside wall temperature would have been equal to the room temperature. Similar problems can cause electronic equipment mounted on the inside wall to fail.

Two commonly used MLI's are Aluminized Mylar layers and Kapton layers. In both cases, the layers are separated by a thin net, usually made of nylon bridal veil. This net prevents thermal conduction, leaving thermal radiation as the only method to transfer heat through the insulation. The Aluminized Mylar is a stronger material than Kapton, but is more dense. Since the structural strength needed for the vehicle has been achieved through other means, the Kapton MLI was chosen due to its lower density, resulting in a lower total mass. The particular Kapton MLI being used on all modules is available from the Sheldahl Corporation, and a diagram of it is provided on Figure 6.2. The Teflon coated glass cloth layer is placed towards the outside of the spacecraft.

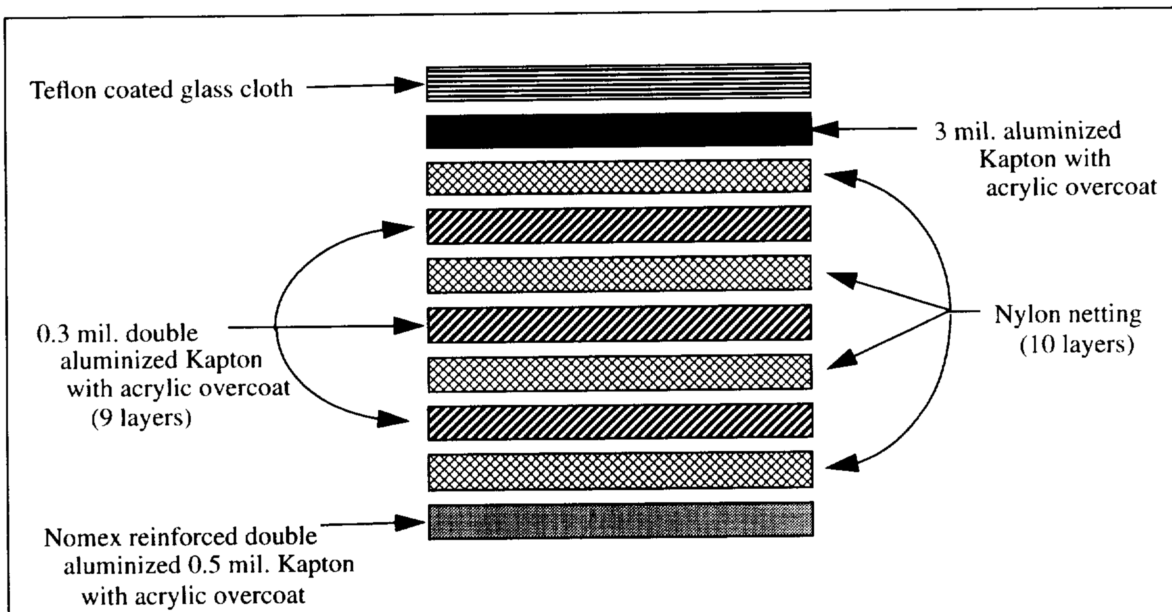

FIGURE 6.2 *Kapton Multi-Layer Insulation from the Sheldahl Corporation[6].*

The same MLI is being used for the crew module, "smartbox", propulsion pack, and cryogenic tanks. Since MLI also has advantages in non-thermal related fields, the thickness of the MLI was determined in conjunction with the structures system based upon both the amount needed for debris shielding and thermal insulation. For this design, the thickness required by the structures system exceeds that required by the thermal system, and therefore the limiting thickness is not discussed here. The exceptions to this are the cryogenic tanks, which require more extensive thermal insulation, since the propellant temperatures must be kept low. No active cooling system will be provided in addition to the venting system. The goal of this design has been to prevent venting of more than 5% of the liquid propellant during on-orbit storage of up to ninety days. Figure 6.3 shows a plot of percentage boiloff as a function of the MLI thickness. The experimental data shown can be extrapolated to reach the conclusion that at least 10 cm of MLI will be needed for the cryogenic tanks.

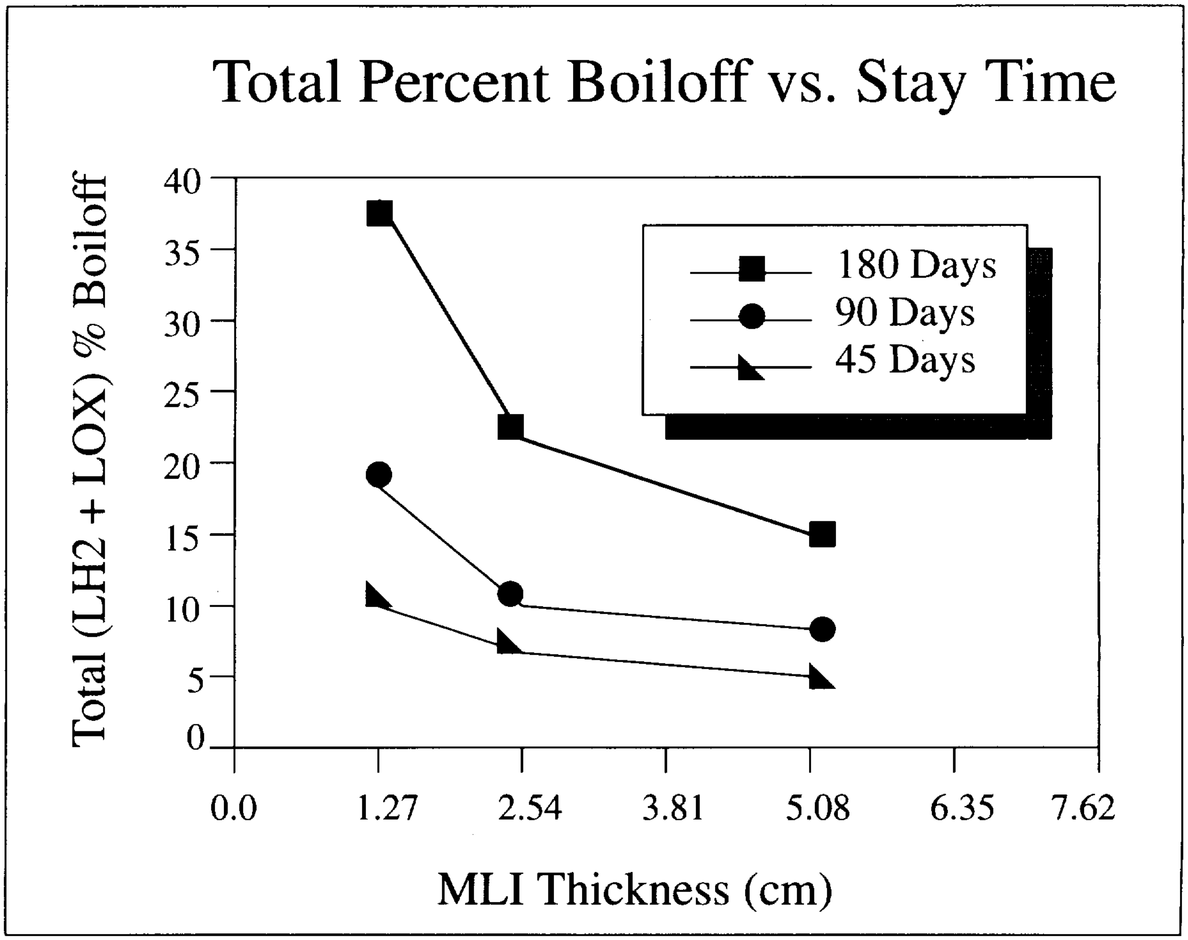

FIGURE 6.3 *Total Percent Boiloff Versus Stay Time[5].*

Another use for MLI is to isolate the fuel cells due to their sensitive thermal nature. While most electrical equipment can have its heat passively dissipated, fuel cells require more careful monitoring, and therefore are cold-biased and insulated.

Phase Change Devices

Heat rejection from electrical components is achieved by the use of phase change devices, cold plates, heat pipes, and radiators. This applies to all modules, except for the crew module, where an active thermal system will also be implemented.

A phase change device is used for equipment that has short bursts of high power. Phase change devices absorb thermal energy through a solid/liquid phase change, and then slowly solidify again, transferring their heat onto a cold plate. Phase change devices are not used for all electrical components, but rather those having erratic power behaviors, in an attempt to keep a constant heat flux flowing towards the radiator. Therefore phase change devices are usually required for only a few components.

Cold Plates

A cold plate can be manufactured for use with either passive or active thermal systems. In the passive case, the cold plate is a highly heat conductive mounting plate for electrical components such that they easily dissipate their heat onto the plate. From there, the heat travels through heat pipes to the radiator, where it is radiated into space. In the case of electrical devices with erratic heat output, the phase change devices serve as the mounting platform and are attached to the cold plate at the opposite end (See Figure 6.4). From there, the heat pipes carry the heat to the radiator as before.

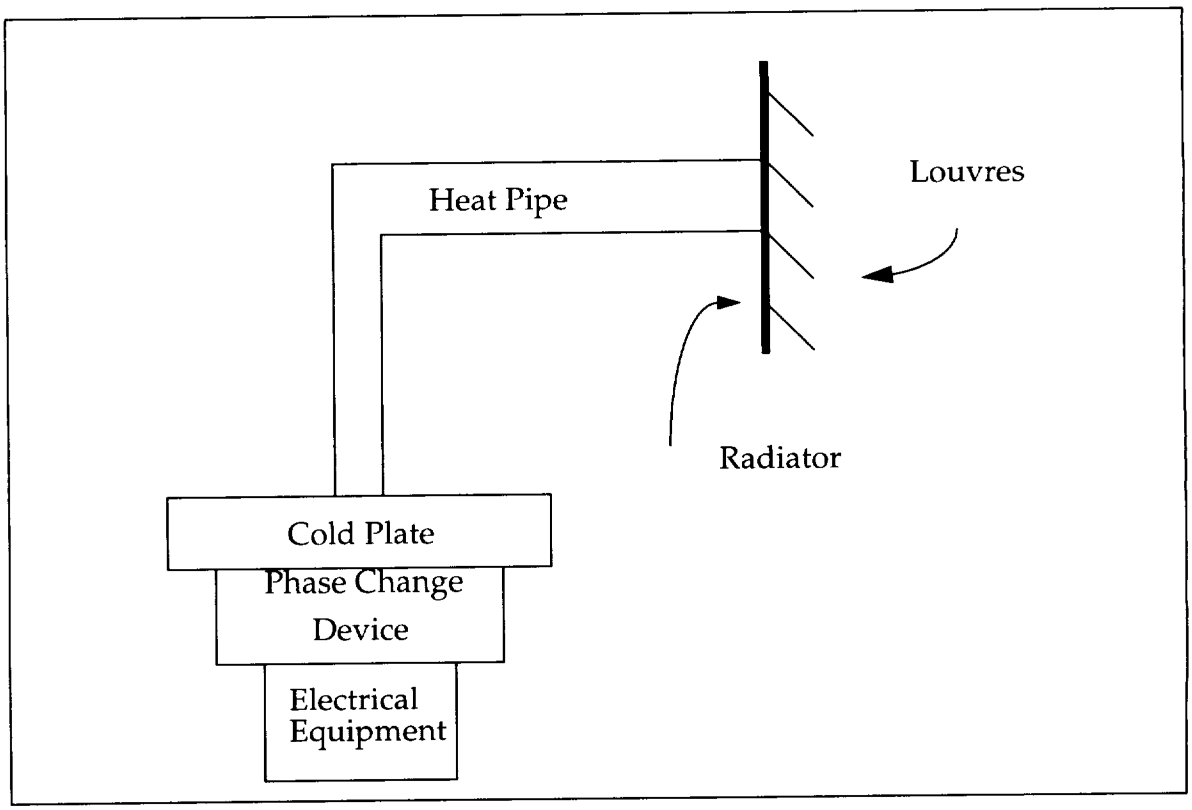

FIGURE 6.4 *Passive Thermal System Configuration for Heat Dissipation of an Electrical Component[3].*

In the active thermal control case the cold plate functions in the same manner as in the passive case, but is configured with fluid passages. Fluid is circulated through the cold plate and transports the waste heat to the radiator. This active system does not make use of heat pipes (See Section 6.4.2 on page 112).

Heat Pipes

Heat pipes are devices with very high thermal conductivities. In the longitudinal direction, they are made up of an evaporator section and a condenser section. A further adiabatic section can be included to separate the evaporator and condenser section, if the length of the heat pipe must be increased. The cross-section of the heat pipe consists of a container wall, a wick structure, and the vapor space (see Figure 6.5).

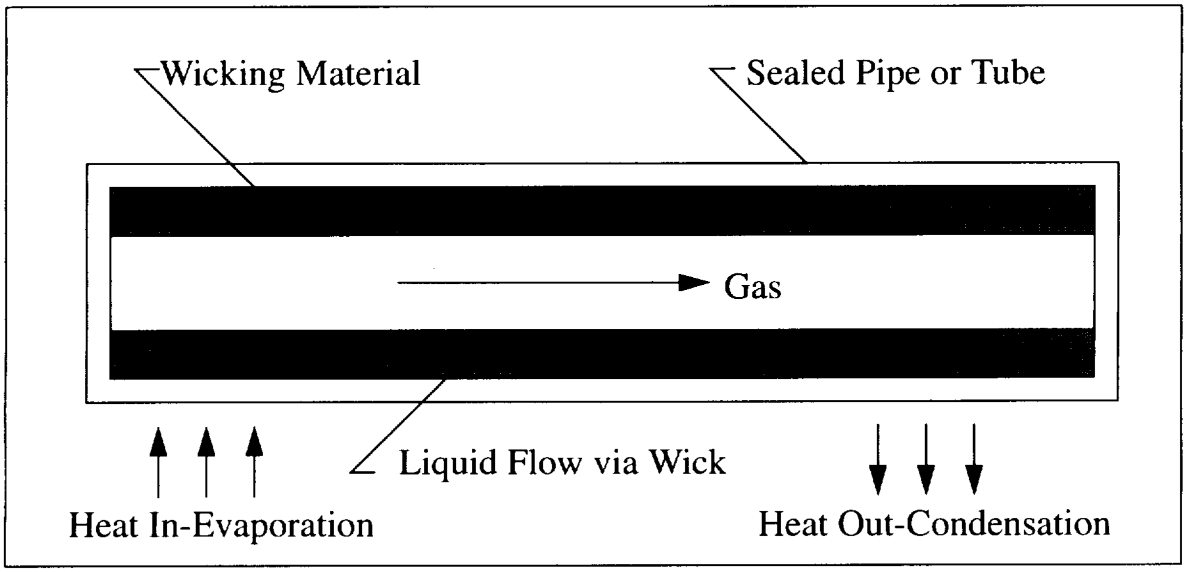

FIGURE 6.5 *The Main Regions of a Heat Pipe[4].*

The substance inside the heat pipe is transformed into a gas in the evaporator. This gas, carrying the newly acquired heat, flows down the middle of the heat pipe to the opposite end, where the heat is transferred from the pipe to the radiator. This condensation stage occurs in the wicking material. Capillary forces then draw the fluid from the condenser end to the evaporator end of the heat pipe, and the process begins again.

The heat pipes being used are available from Hughes Aircraft Corporation, Space and Missile Group. They have diameters of 1.27 cm and can transfer heat at 5080 Watt*cm (e.g., the heat pipes can transfer 508 Watts over a distance of 10 cm). Therefore, multiplying the heat dissipation needed by a component times the length of a heat pipe needed to reach the radiator helps determine the number of heat pipes needed per component. Table 6.7 lists the number of phase change devices, cold plates, and centimeters of heat pipe needed for the crew module and base unit.

TABLE 6.7 *Heat Transfer Devices*

Module	# of Phase Change Devices	# of Cold Plates	Length of Piping (Cm)
Crew Module	none	7	0
Base Unit	1	11	1,800

6.4.2 Active Thermal Systems

Active Thermal Systems are used in situations that require precise temperature control (small temperature variations). A prime example of this is the crew module. Humans do not take well to temperature variations of more than a few degrees centigrade. However, most electrical equipment can withstand variations of up to 40°C. Also, humans have greater heat dissipation than most equipment. As stated in Section 6.3.2 on page 106, a nominal crew of three will put out 490 Watts, whereas all the equipment within the crew pod will only put out 36 Watts under average conditions. The number of heat pipes necessary to carry so much heat flux can be extensive. An active system can carry such heat loads with much greater efficiency. These two factors were the drivers for using an active system for the crew module where more power dissipation is required. This active system consists of a pumped coolant heat transfer device and an air recycling unit.

Other active components include heaters, louvres, and thermal venting systems. Heaters are found in all modules, and are used in RCS tanks and line heating, and radiator heating, all of which have stricter thermal operating ranges than the average electrical component. Louvres are mechanical devices used to maximize or minimize the efficiency of the radiator as appropriate. The thermal venting system is used for cryogenic tanks, keeping the pressure at appropriate levels for the propellant to remain in its liquid state (see Section 5.3.3 on page 94).

Heaters

The radiators will each need a thermostat and a heater. The radiator heater must provide 45 Watts in a worst case scenario in the crew module (See Section 6.4.3 on page 116). There will be two such heaters for each crew module radiator, one of which is for redundancy. The base unit will have a single heater on each radiator, but they will not be used during nominal missions. The louvres, discussed later in this section, are sufficient to maintain the radiators within their operating temperatures at all times for the base unit. A heater is nonetheless placed on each base unit radiator as a redundancy procedure for two reasons. First if Freebird must operate at very low powers, so that the minimum heat dissipated is even less than shown on Table 6.4 on page 107, the louvres may no longer be sufficient to maintain the radiators within their operational range, and heaters will be needed. Another possible emergency, such as a failure of the louvre mechanical system, might cause the louvres to become stuck in the open position, and therefore heaters would again be needed. Second, while Freebird is in the Vehicle Storage Facility (VSF) and is on stand-by mode, the heat dissipated will again be less than that shown on Table 6.4 on page 107. Heaters will receive power directly from the VSF to prevent Freebird from reaching low temperatures.

During mission operation, the thermometers will trigger C^3T to command the heaters to turn on and off so that the radiator is maintained at the appropriate temperature.

Pumped Coolant Heat Transfer Mechanism

Three conduction mechanisms have been considered to carry heat from equipment and the crew cabin to the radiators. The first are heat pipes. As discussed previously, this is a passive thermal component that is limited in two ways. Heat pipes cannot be used to accurately maintain a desired temperature within a small variation range, and they are limited as to the amount of heat rejection they can put out due to the restrictions on pipe length. The longer the heat pipe is, the smaller is the amount of heat that can be dissipated. In many instances, the equipment that requires heat dissipation cannot be placed near a radiator. The amount of heat piping then required increases rapidly due to having longer heat pipes that carry less power. Active pumped systems have no such constraints on length.

There are two active pumped systems that were considered. The first is a single phase active heat transport system. This system requires a pump, which is relatively unreliable as compared to a passive system using heat pipes. Unfortunately, this is unavoidable, considering that all active systems will require pumps. In an active system, the piping of the pumped fluids runs through the mounted cold plates to increase the performance of the cooling system. Another disadvantage in a single phase fluid active system is that a large temperature differential is required to accomplish a substantial heat transfer between the equipment and the pumped fluid. Since the radiator temperature needs to be even lower than the pumped fluid temperature, a large radiator will be needed for such a system. The heat flux is the same, but from Equation 6.6 it becomes evident that for a constant heat flux, as the radiator temperature goes down, the radiator area must increase. Considering the competition for surface area space among the subsystems, reducing radiator area by using another thermal control system becomes very desirable.

The two phase heat transport system acts much like a heat pipe, but since it is an actively pumped system, it does not have all of the restrictions in pipe length or broad temperature control ranges found in heat pipes. Also, since the pumped fluid has two phases, no large temperature differential is needed. The result is a near isothermal (thus allowing a smaller radiator), lower mass flow, and lower specific weight thermal control system. This system incorporates the best of both previously discussed options. The only disadvantages are the requirement of having a pump, which is unavoidable, and the fact that a two phase heat transport system is a relatively new technology. However, the technology necessary is not complicated, nor does it incorporate any new principles. This is the heat transport device that has been chosen for Freebird.

The two phase heat transport system, much like the single phase one, runs its piping directly through the cold plates onto which heat dissipating equipment is mounted. Also, piping is run through a heat exchanger that collects waste heat from the air recycling unit (The air recycling unit is discussed later in this section). At each such junction, the pipes are selected such that the mass flow through them is the necessary amount to change phase due to the particular heat dissipated by the component in the path. Not every component has a constant output of heat. However, the mass flow though the pipes must be sufficient to handle the worst case (highest heat dissipation needed) for any component on its path. If a component is extremely erratic in its heat output pattern, it is attached to a phase change device that smooths out the power dissipation as a function of time (Section 6.4.1 on page 107). After acquiring heat from the cabin and equipment, the fluid has been evaporated. It travels to a condenser, where the fluid is forced into another phase change, returning it to a liquid by removing the waste heat. The fluid then passes through a subcooler which ensures that the fluid has been completely condensed. Both the condenser and the subcooler are connected to the radiator, which radiates the waste heat into space. The coolant fluid being used is Freon. The two phase heat transport process is illustrated in Figure 6.6.

Cabin Thermal System [7]

The cabin thermal control system consists of an air recycling unit, whose components include fans, ducts, a heat exchanger, and an atmosphere control subsystem. There are two fans which circulate the air through the ducts. One is always in stand-by mode for redundancy purposes.

The recirculated cabin air goes through a heat exchanger where its heat is dissipated. From the heat exchanger, waste heat is transferred to the active thermal control system via the pumped fluid, which is then transferred to the radiator.

The heat exchanger is a dual purpose device. In addition to cooling the recycled air, humidity is reduced. The resulting condensed moisture is transported by diverting a portion of the air flow through holes in the condenser (which acts like a vacuum cleaner) to a rotary water separator where it is removed. The cooled air is returned to the cabin and the condensed water is stored in a waste water tank.

The atmosphere control subsystem does not regulate the temperature ranges of any component on board Freebird, and therefore is not a thermal system. Nonetheless, the subsystem is part of the air recycling unit, and therefore is included in this section. It includes a debris trap that removes objects, such as pieces of thread, bits of paper, spilled liquid, or food waste. It also includes lithium hydroxide canisters that absorb carbon dioxide, and charcoal canisters that remove odors. The entire air flow passes through the debris trap, but only 10% is deflected through the canisters.

Louvres

Louvres are positioned above the radiator surface. They can vary in angle from normal to parallel with respect to the radiator surface. When parallel, the louvres are said to be closed. The louvres resemble Venetian blinds, and when closed, they greatly lower the effective emissivity of the radiators. They are used in nighttime conditions, when full radiator emissions are not necessary. When the spacecraft is hidden from the Sun behind the Earth, the only external heat flux is due to the Earth's infrared emissions. Under these conditions, it becomes necessary to warm the radiators with heaters in order to prevent the radiator temperature from dropping below

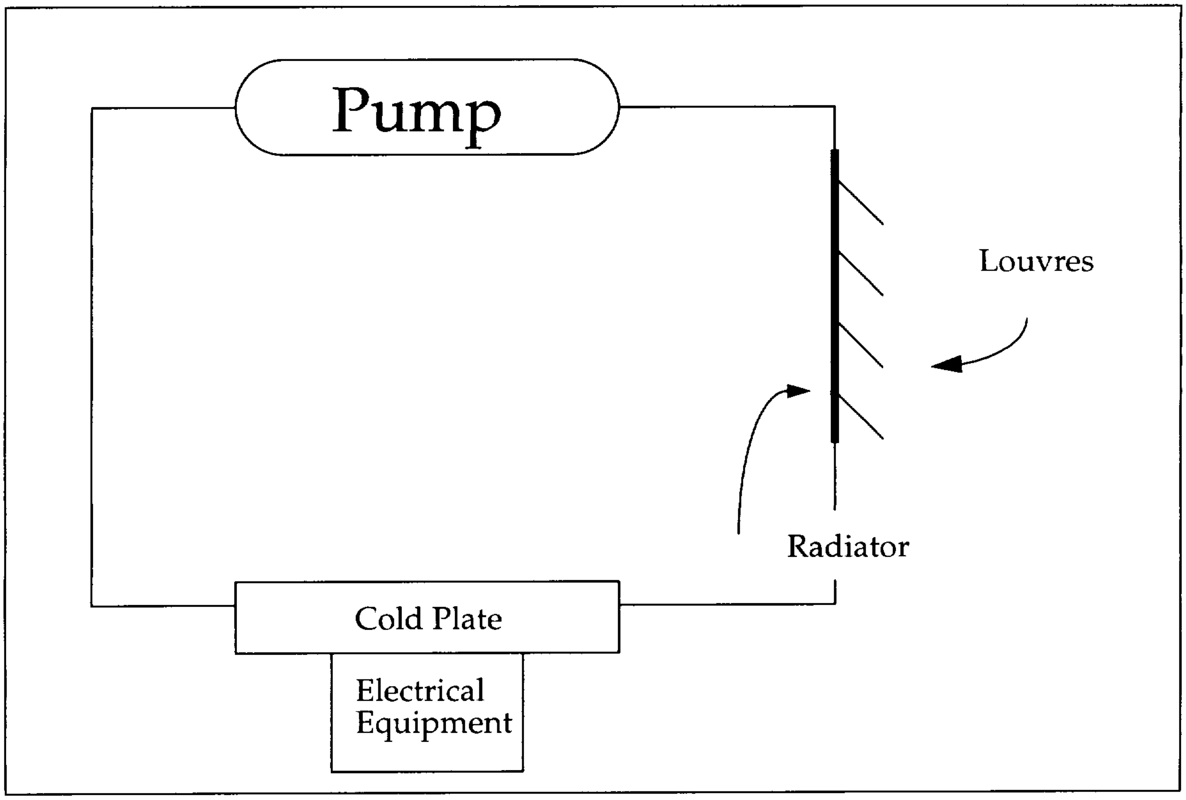

FIGURE 6.6 *Two Phase Heat Transport System [3].*

the lower limit levels. With the louvres closed, the radiator will emit less heat, and therefore, the temperature will not drop as much. Louvres can greatly reduce the thermal power consumption by lowering the power output needed from radiator heaters. The spacecraft has a set of louvres over each radiator. The louvre configuration can be seen on Figure 6.6.

In order for louvres to be effective, they need to have low emissivities. This way, when the spacecraft finds itself in the shade, the temperature of the louvres will not decrease dramatically. On the other hand, while the spacecraft is in full view of the sun, if the louvres are accidently exposed to by the sun, the resulting temperatures can be very large. These large louvre temperatures can cause structural damage to the louvres themselves through thermal expansion, and also reduce the emissivity of the radiator when it needs it most, in the day side of the orbit.

A solution to this problem that has been used in the past involves coating the louvres with white epoxy. The result is that the temperature of the louvres does not exceed the equilibrium temperature of white epoxy under solar heat flux conditions (Table 6.6 on page 108). However, when the spacecraft finds itself in the shadow side of the orbit, and the white epoxy is used, the emissivity that results by closing the louvres is not much lower than that if no louvres were present at all.

The proposed solution for Freebird is to use an innovative concept for a louvre. The idea is to design a double louvre with two thin plates of Aluminum on top of each other, sandwiching Alumized Kapton MLI in the middle. In addition, one of the sides of the louvres would be coated with white epoxy. The concept is that the white epoxy would prevent large temperatures from occurring during daylight portions of the orbit. However, when

Freebird reaches the nocturnal portions of the orbit, the closed louvres would not retain the large emissivity due to the protective multi-layer insulation.

Aluminum louvres have shown a reduction in thermal emissivity of up to two thirds of the radiator's nominal capacity. When coated with white epoxy, the louvres achieve a reduction in thermal emissivity of only 20%. Using a MLI double plated louvre is estimated to reduce the emissivity by 56%. Since the nominal emissivity of Aluminized Teflon, the radiator material of Freebird, is 0.8, the emissivity with the louvres closed will become 0.35.

The louvres are moved by mechanical motors. Two such motors will be present at each radiator for redundancy purposes. The motor on stand-by mode will be slightly more powerful than the nominal operations one. The purpose for this is to prevent the first motor from failing while at the same time inhibiting the second one from taking over the function.

6.4.3 Radiators

The radiators are the only means for the spacecraft to dissipate heat. Three considerations go into material selection for a radiator. The radiators must have high thermal conductivity, such that heat will be quickly transferred from the heat pipes. The radiators must also have low absorptivities, such that the heat flux from the Sun or Earth to the radiators will be minimized. Finally, radiators must have high emissivities, such that heat can be quickly radiated into space. Some materials, such as white epoxy, have low absorptivities and high emissivities, but do not conduct heat well. Other materials, such as most metals, conduct heat very well, but have very poor emissivity to absorptivity ratios. Aluminized Teflon is a popular and good material selection candidate. It meets the above three constraints and therefore has been chosen as the radiator material. Radiation properties for Aluminized Teflon can be found on Table 6.6 on page 108.

The crew module and base unit will each have two radiators. The total radiator area must be such that it is capable of dissipating the maximum amount of heat required. Table 6.8 shows a step by step analysis detailing how the radiator size was determined for the crew module. Table 6.9 does the same for the base unit. The following steps were followed:

TABLE 6.8 *Crew Module Radiator Sizing*

No.	Item	Value
1	Maximum Power Dissipation	1113 Watts
2	Minimum Power Dissipation	73 Watts
3	Radiator Temperature	285 K
4	Combined Radiator Area	9.12 m^2
5	ε of Radiators with Louvres Closed	0.35
6	Low End Radiator Temperature	282 K
7	Heater Power to Radiators	45 Watts

1. The maximum power dissipation which the radiator must provide is shown (See Section 6.3.2 on page 106).
2. The minimum power dissipation needed is shown (See Section 6.3.2 on page 106).
3. The radiator temperature is the temperature at which the radiator must be maintained such that it can take heat from the components that require heat dissipation while maintaining the equipment within its operating range (See Table 6.5 on page 108). This temperature must be lower than the equipment tem-

TABLE 6.9 *Base Unit Radiator Sizing*

No.	Item	Value
1	Maximum Power Dissipation	768 Watts
2	Minimum Power Dissipation	160 Watts
3	Radiator Temperature	285 K
4	Combined Radiator Area	6.30 m^2
5	ε of Radiators with Louvres Closed	0.35
6	Low End Radiator Temperature	292 K
7	Heater Power to Radiators	None

perature in order for heat to be transferred to the radiator. On the other hand, if the radiator temperature is too low, the equipment runs the risk of going below its minimum temperature limit. Therefore, the radiator temperature was picked such that it is 5°C above the minimum acceptable temperature of the component having the highest minimum temperature. Therefore, the radiators will be maintained at 12°C since the highest minimum temperature is that required by the RCS propellant, 7°C. The crew is not considered amongst the components for which the radiator must be at least 5°C above the minimum operating temperature, because the crew has its own temperature control through the use of the air recycling unit. This unit can be simply shut off, and therefore the crew does not at any time run the risk of having the cabin drop temperature to undesirable levels.

4. The radiator area is obtained based on worst case hot conditions. It assumes that the radiator is facing Earth in LEO with its louvres fully opened. The external heat flux to the radiator is the sum of the Earth's albedo and the Earth's infrared emissions. The internal heat flux is the maximum power dissipation that the radiator must provide for, obtained from Part 1. Using the radiator temperature (T), the above heat fluxes (both internal and external), and Table 6.6 on page 108 to obtain values for the solar absorptivity (α_s) and infra-red emissivity (ε_{ir}) of Aluminized Teflon, solving Equation 6.6 gives a value for the radiator area (A_r) for a given internal heat flux to be dissipated (Q). In this case, Q is the maximum heat flux that the radiator will need to dissipate. The first two terms of the equation on the left hand side can be traced back to Equation 6.4 and Equation 6.2 of Section 6.3.1 on page 104. σ is the Stefan-Boltzmann constant, 5.67 X 10^{-8} $Wm^{-2}K^{-4}$.

$$q_I \sin^2\rho A_r \varepsilon_{ir} + G_s a A_r \alpha_s \sin^2\rho + Q = \sigma\varepsilon_{ir} A_r T^4 \qquad \textbf{(Equation 6.6)}$$

5. The radiator's emissivity changes depending on the louvres' position. When minimum heat dissipation is required and the spacecraft is in nighttime conditions, the radiator temperature can drop substantially. Therefore, louvres are used to minimize the radiator's emissivity and raise its temperature. The radiator's emissivity with the louvres closed is 0.35 (See Section 6.4.2 on page 112).
6. Solving Equation 6.6 with Q as the minimum power dissipation value and G_s equal to zero gives the resulting temperature that would occur if no heater was used to maintain the radiator at its minimum acceptable temperature during nighttime conditions. The emissivity used was that calculated in Part 5.
7. Heater power input to radiator needed to raise the temperature from the value found in Part 6 to the one determined in Part 3 is determined.

6.5 Failure Modes

TABLE 6.10 *Failure Modes*

Component	Failure Mode	Effect	Prevention or Response	Criticality	
				Crewed Config.	Teleop.
Coolant Pump	Mechanical failure	Complete failure of crew module thermal system	Full redundancy	1	N/A
Air recycling unit	Mechanical failure	Loss of heat rejection from crew cabin	Partial redundancy	1	N/A
Heaters	Mechanical failure or power loss	Loss of temperature control on RCS propellant	Full redundancy	1	2
		Failure to maintain radiator at minimum allowable temperature	Full redundancy	1	2
Heat Pipes	Excessive deformation	Possible failure of a critical component	Full heat dissipation redundancy for all critical components	1	2

The thermal control system failure modes are listed in Table 6.10. All critical components are listed with their major failure modes, the effect of failure, the risk associated with each failure, and the measures taken to prevent such a failure. For a crewed configuration, all system failures are vehicle and crew critical (criticality rating of 1). In response, all critical components are fully redundant, except for the air recycling unit, which has internal redundancies on its less reliable parts. In teleoperated configuration failure of the coolant pump and air recycling unit are not relevant since the crew module isn't part of a teleoperated mission. All other failures in this configuration are vehicle critical (criticality rating of 2), so full redundancy is required for teleoperation as well.

6.6 Summary Budgets

TABLE 6.11 *Summary Budgets - Crew Module*

Component	#	Mass [kg]	Avg. Power [Watts]	Peak Power [Watts]	Cost [$]
Radiator	2	0	0	0	0
Insulation	40%	2.2	0	0	42,000
Cold plates	7	3.75	0	0	54,000
Coolant pump	2	4	35	50	78,000
Coolant piping	100%	10	0	0	194,000
Air recycling unit	1	35	50	75	679,000
Radiator heaters	4	.4	0	45	5,000
Louvres	20	2	0	0	39,000
Louvre mechanical system	4	8	0	10	155,000
Thermometers	12	0	0	0	1,000
Thermal System Total		65.35	85	180	1,247,000

TABLE 6.12 *Summary Budgets - Base Unit*

Component	#	Mass [kg]	Avg. Power [Watts]	Peak Power [Watts]	Cost [$]
Radiator	2	0	0	0	0
Insulation	60%	3.3	0	0	63,000
Cold plates	11	5.75	0	0	83,000
Phase change devices	1	2	0	0	29,000
Heat pipes	100%	15	0	0	218,000
Radiator heaters	2	.2	0	0	3,000
RCS propellant heaters	64	6.4	32	48	85,000
Louvres	20	2	0	0	39,000
Louvre mechanical system	4	8	0	10	155,000
Thermometers	18	0	0	0	1,500
Thermal System Total		42.65	32	58	676,500

6.6.1 Mass Budget

The thermal system mass budget is itemized in Table 6.11 for the crew module and in Table 6.12 for the base unit. The total system mass is broken down according to components. The mass budget drivers are the air recycling unit for the crew module, and the heat pipes for the base unit. The radiators would be expected to be a mass driver, but the actual radiator mass is accounted for in the structures system budget. The only mass associated with the radiators accounted for by the thermal system is the mass of the surface coating on them, which is negligible. The total mass of the thermal system of all modules is 108 kg.

Power Budget

The thermal system power budget is itemized in Table 6.11 for the crew module and in Table 6.12 for the base unit. The thermal system power requirements are broken down by component, and average and peak power requirements are given for each. At average and peak power the coolant pump and air recycling unit are the power drivers, accounting for 100% of average power and 69% of peak power consumption. Aside from occasional use of the louvre mechanical system, the RCS propellant heaters are the only power consuming components in the base unit, since all others are either passive, or have only been placed there for redundancy purposes. An example of such a redundancy are the radiator heaters in the base unit, which are back-ups for the louvre mechanical system. The louvre mechanical system is used intermittently for small adjustments and requires little power even at peak power.

Cost Budget

The thermal system cost budget is itemized in Table 6.11 for the crew module and in Table 6.12 for the base unit. It details the quantity of each component used by the system and the total cost of each set of components. These costs are hardware costs only. The major cost drivers for the crew module are the louvre mechanical system and air recycling unit, which together account for 54% of the total module cost. The major cost drivers for the base unit are the louvre mechanical systems and the heat pipes, which account for 55% of the base unit cost. Also, important cost consideration must be given to the louvres which will involve some research costs due to their new design presented earlier. Research costs are not reflected in the numbers given on the tables. Finally, it is interesting to note that the radiator cost is also negligeable. Just as in the mass budget section, the cost of the radiator is included in the structures system's budget. The extent of the radiator's cost as relevant to the thermal system is simply the cost of the surface coating (Aluminized Teflon). Due to relatively small radiator areas, the cost of the coating becomes insignificant. The total cost of the thermal system of all modules is $1,923,500.

Telemetry Budget

The thermal system telemetry budget is shown in Table 6.13. The larger amount of telemetry data is in the form of temperature readings. The only commands provided are to louvre mechanical systems to orient the louvre angles, and to heaters to turn them on and off. The air recycling unit pump has a commanded rate of mass flow that it allows to pass through. This is determined by the crew cabin astronauts who set the thermostat at whichever value they wish.

TABLE 6.13 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital	Command
Smart Box Thermometers	Temperature	18	1	A	
Smart Box Louvres	Louvre Angle	4	1	A	Commanded Angle
Smart Box Radiator Heaters	Status	2	1	D	On/Off
RCS Heaters	Status	64	1	D	On/Off
Thermometers	Temperature	12	1	A	
Louvres	Louvre Angle	4	1	A	Commanded Angle
Radiator Heaters	Status	4	1	D	On/Off
Thermostat	Temperature	1	1	A	

TABLE 6.13 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital	Command
Air Recycling Unit Pump	Mass Flow	2	1	A	Commanded Rate
	Status	1	1	D	
Freon Coolant	Temperature	2	1	A	
Active Coolant System Pump	Status	2	1	D	

Freebird operates in an environment of great temperature extremes. To effectively maintain operating temperatures of all Freebird systems, the thermal control system utilizes several key technologies. White epoxy surface coating is used on all exposed vehicle surfaces to maintain an appropriate surface temperature. Multi-layer Aluminum Kapton insulation is used to insulate all vehicle modules. An active thermal system is used to maintain the crew module operating environment. Passive thermal systems and components are used in the other modules. A louvre system is used to regulate the radiators' emissivities.

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Chapter 7

Guidance, Navigation, and Control System

The following is an explanation of the terminology used for Project Freebird with respect to the Guidance, Navigation, and Control (GNC) system. Attitude determination and navigation have been treated as two separate components of GNC, although the two are highly coupled. *Attitude determination* means, just as its name implies, the determination of the vehicle's attitude and *navigation* is used to mean orbit determination. *Guidance* is used to mean orbit control and *control* is used as a shortened form of attitude control system.

In further detail, the attitude determination and control system (ADCS) stabilizes the vehicle and orients it in the desired directions during the mission despite disturbance torques acting on it. In order to do this the vehicle must determine its attitude using sensors and control it using actuators. Navigation refers to the collection of data and its reduction to determine the current orbital position and velocity of the vehicle. Lastly, guidance refers to the action of adjusting the vehicle's orbit to meet some predetermined conditions. [1]

7.1 GNC Requirements

The GNC system requirements were determined from the overall system requirements and accuracies were specified in order to quantify these requirements. These requirements and accuracies provided the basis for this design study and were used to make informed trade studies and appropriate hardware choices for the system.

For Freebird to achieve its overall mission objectives, the GNC system must:

- Determine absolute position to within 1km in LEO and 50km in GEO
- Determine relative position to within 0.005m
- Control relative position to within 0.01m
- Determine velocity to within 0.003m/s
- Control velocity to within 0.01m/s
- Determine attitude to within 0.1°
- Control attitude to within 0.5°
- Provide a system reliability of 0.999

The absolute position requirements are given as the accuracies achievable by ground station tracking. With full autonomous capability, Freebird should be able to determine its position with an accuracy at least as good as ground tracking or better. The relative position accuracies come from NASA docking requirements with ISSA, and are thus used to dictate the accuracies required for Freebird to rendezvous and dock with any asset. The vehicle velocity accuracies are derived from the Space Shuttle maneuverability numbers. Lastly, the attitude accuracies are dictated by the requirements for pointing purposes, such as antenna and thermal pointing maneuvers.

7.2 GNC Interfaces

The following is a discussion of the internal and external interfaces of the GNC system.

7.2.1 Functional Block Diagram

The internal interfaces are shown in Figure 7.1. The Dual Cone Scanners (DCS) gather information on attitude and altitude of the vehicle and the Miniature Inertial Measurement Unit (MIMU) gathers information on the rotation and translation of the vehicle. This information is utilized by the Microcosm Autonomous Navigation System (MANS) to determine the orbital position and velocity of the vehicle. This information is passed to the control algorithms, which then compare the current state vector to the desired state vector given by C^3T and computes the necessary control steps to achieve the desired one. This information is then passed on to the propulsion system, either the Reaction Control System (RCS) or the main engines, through the C^3T command subsystem.

Information is also obtained by components during the rendezvous, docking, and proximity operations phases. These components include a Rendezvous Radar, a Laser Navigation Sensor (LNS), and a star tracker which gather navigation information to be utilized by the Microcosm Autonomous Planning System (MAPS) and the control software. The same process as described above is carried out to issue control commands.

The reasons for the choices of components listed above and how they work will be discussed in further detail in the following sections.

7.2.2 External Interfaces

Since the GNC system is critical for every mission, it comprises part of the "smartbox". Due to the nature of the system, there are many interfaces with other systems. Figure 7.1 also diagrams these external interfaces.

One of the more important interfaces is with the C^3T system. C^3T handles all the commands of the vehicle and therefore either originates or relays commands for the GNC system. For example, for a piloted mission, the pilot becomes part of the control loop and issues the command to do a maneuver. This command routes through C^3T and is then passed on to the GNC system. This is an example of an indirect interface with the ECLSS system which provides the human interface for the pilot. Another instance would be with the Thermal system. It may need to rotate the vehicle for thermal purposes and thus sends the command to C^3T which is then passed on to GNC as a command to maneuver. Other issues involved with C^3T involve antenna pointing and rendezvous maneuvers.

The interface with the Power and Propulsion system is twofold. The power portion does just that, provide power to the GNC system. The propulsion system implements the control needed for the vehicle, as determined by GNC, using both RCS and the main engines.

The direct interface with the Thermal system involves only the physical dissipation of heat produced by the GNC system.

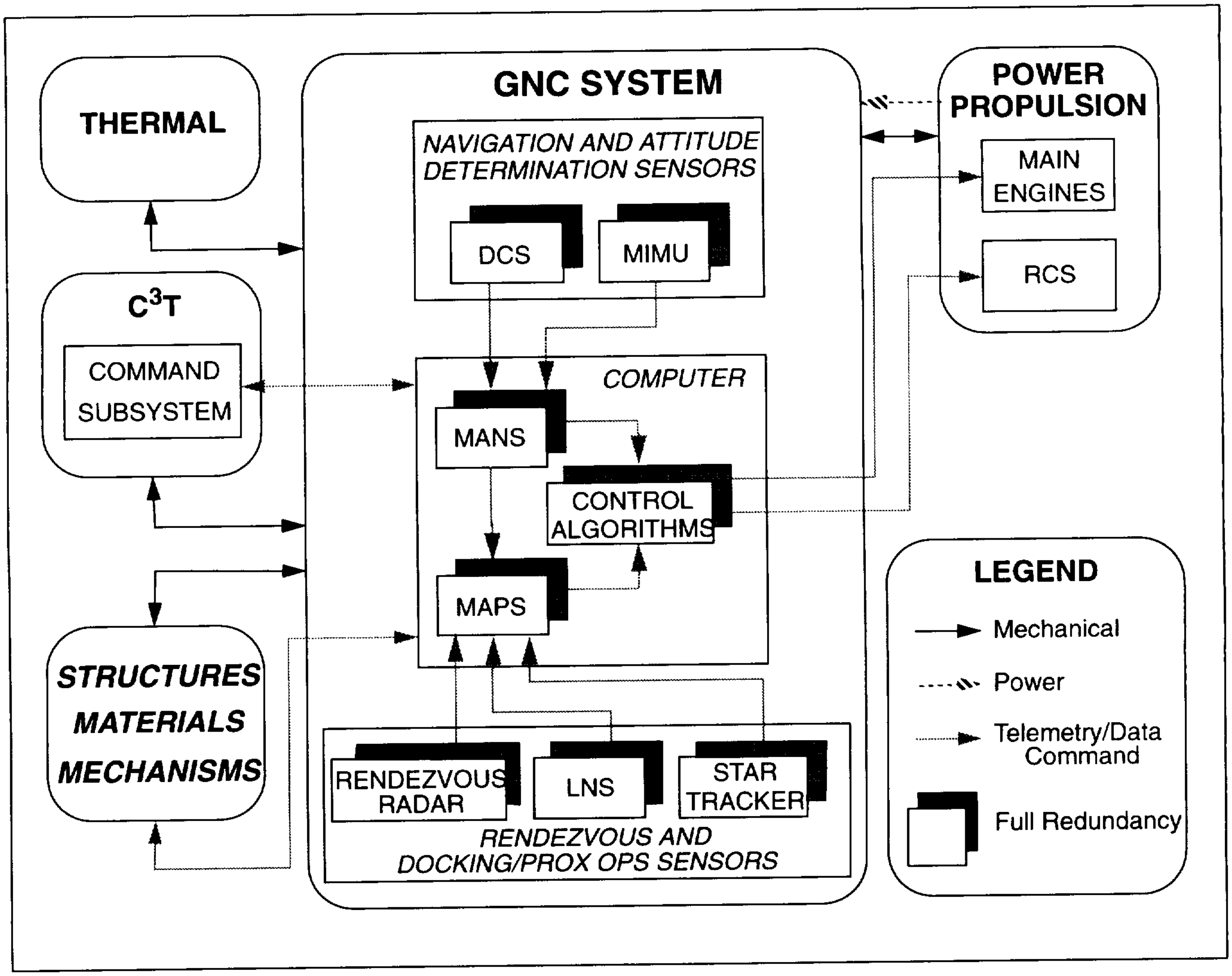

FIGURE 7.1 *Guidance, Navigation, and Control System Functional Block Diagram.*

The interface with the Structures system involves the physical aspect of having the system attached to the structure of the vehicle itself. There is also an interface with the mechanisms involving the Remote Manipulator System (RMS) and the Common Berthing Adaptor (CBA). These interfaces are somewhat indirect in that in order for the implementation of these mechanisms to occur the orientation of the vehicle must be known and communicated to the Structures system.

7.3 Design Considerations

Several preliminary trade studies were made before approaching the design of the GNC system. It was determined that the following guidelines would be followed for the design phase of the system:

- use components available over entire mission envelope (including GEO)
- maximize autonomy

- minimize contamination and thruster plume loads
- be reconfigurable / adaptable
- maximize degrees of freedom

Component Availability Over Entire Mission Envelope

The issue of availability of components over the entire mission envelope was first and foremost to be addressed. Since one of the mission objectives of the vehicle involves operating in GEO, it was decided that usability in GEO was a major selection criterion for all GNC components in order to simplify the system and to avoid carrying deadweight. This for example, eliminated the use of the Global Positioning System (GPS) as a navigation tool, due its unavailability in GEO. [1]

Level of Autonomy

The desired level of autonomy of the GNC subsystem was another major design driver. Currently, it is common to have an autonomous ADCS. However, navigation and guidance are typically ground based with few available autonomous navigation or guidance systems. Autonomous navigation refers to the ability to determine position and velocity on-board the spacecraft, independent of ground stations. Navigation, attitude determination and attitude control are computation intensive and make up much of the ground support processing work load. Thus if the spacecraft had the capability to be autonomous, it would reduce the manpower and work load of the ground station operators, which reduces the operations costs considerably. Also, considering the vehicle will carry out missions both with crew and remotely for a lifetime of ten years, it would be ideal to have the capabilities of an autonomous vehicle. Therefore it was determined to maximize autonomy, while maintaining the capability to communicate with ground stations. [2]

Another aspect of autonomy deals with the level of reliance on humans in the loop. The issue of automatic or manual control is a very important one. With manual control, the pilot becomes part of the control loop, where the human eye is the primary "sensor" and the human brain is the primary "processor." The advantages of using manual control involve a shorter development time for secondary sensors and processors and certification, the flexibility and excellence of the human brain to react to anomalous situations, and the simplicity of this type of system. The disadvantage of a manual system is that it limits the general capabilities of the GNC system. While the human brain is flexible, it is not as accurate and not as timely as an autonomous system. Another limitation exists in the timing of maneuvers, i.e. a human pilot often requires daylight to do maneuvers and thus limits many aspects of the missions.

An autonomous control system has the advantages of being able to accurately predict orbital motion, being fully capable at any point during the mission without any down time, and being able to operate without humans on board. However, the development time involved with this type of system may be longer than a manual system, as well as possibly weighing, costing, and being larger than the manual system. Again since a mission scenario for the vehicle is to operate without a crew, it is preferable to have as fully autonomous a system as possible in order to reduce necessary teleoperating. However, for the manned missions, the capability exists to include the pilot in the control loop and override the system if necessary. [3]

Contamination and Thruster Plume Load Issues

The issues of contamination and plume loads from the control thrusters and main engines was also of primary importance to the design of the system. The effects of plume impingement loading and contamination of sensitive equipment and hardware on satellites and ISSA, and impingement on astronauts both out on EVA and in the crew module were looked at. It was discovered that the driving factor for proximity operations was the plume load issue and not the contamination, although the contamination still had an effect when dealing with the astronauts on EVA. Since the plume load issue was a major one it was decided to minimize the contamination and loads due to the thrusters. This decision affected the placement of the reaction control thrusters, which is discussed in Section 7.7.2 on page 143, as well as the proximity operations strategies discussed further in Section 7.4.5 on page 131.

Reconfigurability/Adaptability

Freebird is quite an extensive vehicle in terms of missions and functions. Therefore, the vehicle system level decision to use a modular design poses many challenges with respect to the GNC system. A certain level of reliability is of course a primary requirement and in order to meet this requirement, a subsystem should be reconfigurable in the event of a failure. A robust GNC system would adapt to failures by reconfiguring the control algorithms to account for the failed systems by optimizing the command functions for the remaining actuators. In this report, reconfigurable involves reacting to changes within the control system, whereas adaptable means reacting to changes outside the control system, such as mass configurations. An adaptable system estimates vehicle parameters from data given by sensors, as opposed to a system which is given these parameters from an outside database. These parameters include inertial properties, such as center of mass location and moments of inertia. This adds to the robustness of the system and allows for a level of autonomy when the vehicle configuration is changed, such as when carrying a satellite or adding additional fuel tanks. It was decided that Freebird be both reconfigurable and adaptable due to the varied nature of its missions and configuration.

Degrees of Freedom

A further effect of the robust nature of Freebird and its wide range of missions is that it must have the ability to maneuver in an accurate and flexible manner. Rendezvous and docking with an object, as well as proximity operations require a system with as many degrees of freedom (DOF) as possible to allow for ease of these maneuvers. Therefore, it was decided that Freebird would have a 6-DOF control system. This allows for yaw, pitch, and roll rotation and x, y, and z translation.

7.4 Control Modes

In designing the vehicle control system, the first step was to define the different control modes that will be carried out during the lifetime of Freebird. Considerations that went into designing control modes were the mission requirements, mission profile, and type of insertion for the launch vehicle. From these considerations, requirements and constraints for the GNC system were also defined. For Project Freebird, the following non-mutually exclusive control modes were defined.

7.4.1 Acquisition

Definition

Acquisition refers to the initial determination of the vehicle's attitude and position, stabilization of the vehicle, calibration and initialization of sensors, and testing of GNC systems. Acquisition will be done during all vehicle start-ups and after system resets.

Strategy

Upon entering acquisition mode, the spacecraft will inhibit thruster firing while the sensors are calibrated and initialized so that position and attitude may be determined. Once position and attitude are known, the vehicle will be stabilized using the reaction control system.

7.4.2 Stationkeeping

Definition

The default mode when Freebird is not maneuvering or docked will be stationkeeping. Stationkeeping is used to maintain Freebird at a desired relative position, attitude, and attitude rate with respect to a target. [4]

Strategy

Freebird will primarily employ V-bar stationkeeping, i.e., maintain a position on the velocity vector of the target as illustrated in Figure 7.2, to minimize fuel use. V-bar stationkeeping uses the minimum amount of fuel since it is the only completely stable position, i.e., no thrusting is required to maintain the position. [4] Therefore, other positions will be used only if required for a specific operation.

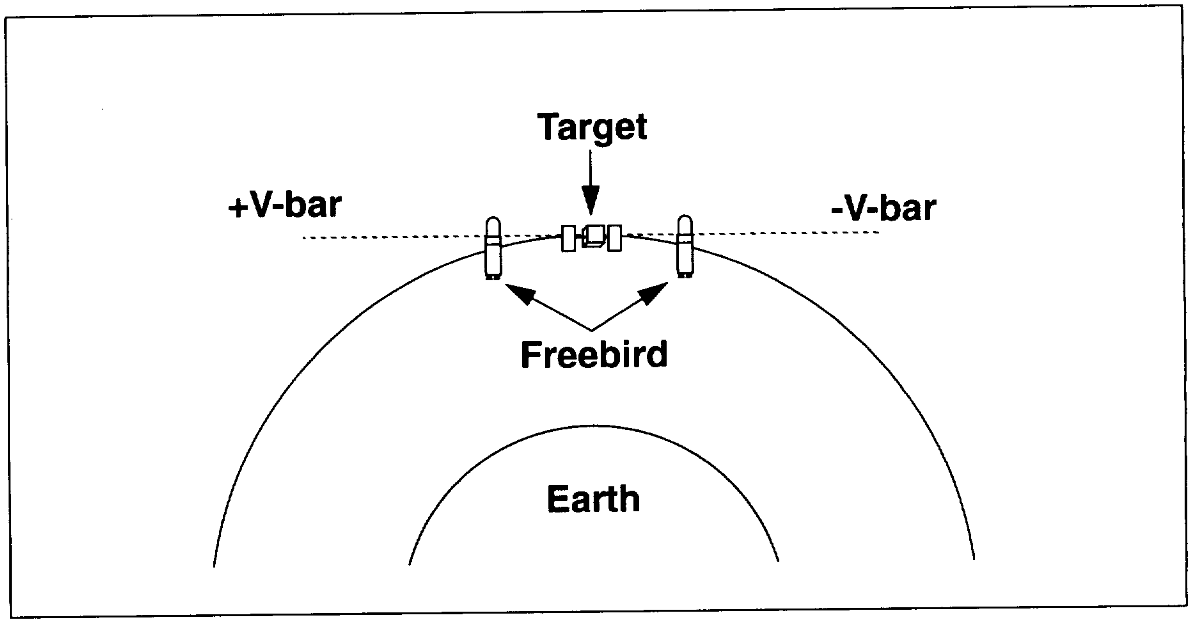

FIGURE 7.2 *V-bar Stationkeeping.*

7.4.3 Slew Maneuvers

Definition

Slew maneuvers reorient the vehicle attitude. This could be required during any phase of a mission, such as thermal pointing during an orbital transfer.

Strategy

Freebird will use its RCS system to perform slew maneuvers.

7.4.4 Orbital Transfer/Rendezvous

Definition

Orbital transfer and rendezvous encompass all the operations that take Freebird from its initial position to within a few hundred meters of its destination.

LEO Strategy

Freebird's LEO rendezvous maneuvers are modelled after the strategy currently used by the Space Shuttle. Initially, Freebird will make an inclination change with its main engines. Once Freebird is in the same inclination as the target, it will commence with primary phasing. This requires insertion into an elliptical orbit with either the perigee or apogee at the same altitude as the target spacecraft. This allows Freebird to adjust its phase to match that of the target vehicle.

The primary phasing should place Freebird approximately 75km behind the target vehicle before the next phase of rendezvous is initiated. Figure 7.3 is a relative motion plot of a Freebird rendezvous from this point on. It shows Freebird in a lower energy orbit, approaching the target from below and behind. If Freebird is coming from a higher energy orbit (i.e. ISSA is located in a higher altitude orbit than the target vehicle), the rendezvous is reversed, and Freebird approaches the target from above and ahead. This can be imagined by turning Figure 7.3 upside down. Once Freebird reaches apogee (perigee if Freebird is approaching from above) of its primary phasing, it initiates its secondary phasing maneuver, slowing down its rate of angular approach. Unadjusted, it should now take three orbits for Freebird to pass the target vehicle.

The purpose of the secondary phasing is to allow Freebird to collect data on the position of the target vehicle. For a target spacecraft in LEO, ground tracking is accurate only to a few kilometers. Freebird uses both star-tracker and radar data to improve this accuracy. The secondary phasing of Freebird should be timed so that the target will be illuminated by the sun as Freebird drops from apogee to perigee. During this time, Freebird's onboard startracker will track the target vehicle, providing accurate angular data from Freebird to the target. As Freebird approaches the target, radar data also becomes available. Beyond about 40km, star tracker data is still far more accurate, but as Freebird gets closer, radar resolution improves. Freebird radar also provides data on target range and range rate. Corrective burns may be made as the target positional data is improved.

Following two secondary phasing orbits, knowledge of target position should be accurate enough for terminal phase initiation. After one orbit of the terminal phasing, Freebird should be within several hundred meters in front of the target vehicle. The target should now be within range of Freebird's Laser Navigation Sensor (LNS). At this point, Freebird's orbit is circularized, and Freebird moves into its proximity operations mode.

GEO Strategy

The major challenge posed by a rendezvous to a spacecraft in GEO is the low positional accuracy of both Freebird and the target vehicle. Ground station tracking provides positional data on the target vehicle only to within 50km in GEO[1]. The Freebird navigation sensors are slightly better, providing orbital position data of Freebird to within 35km in GEO. Another problem posed in GEO is the lengthy 24hour orbit. This rules out the rendezvous technique used in LEO, which requires at least three orbits.

Freebird will use a Hohmann transfer to maneuver to GEO altitude. The apogee of this transfer orbit needs to be approximately at the position of the target vehicle at the time of Freebird arrival. This necessitates phasing maneuvers in LEO, to align the perigee of the transfer orbit. Once Freebird is at GEO altitude, it will need to circularize its orbit, as well as perform an inclination change from 51.6° to an equatorial orbit. This means that Freebird must be crossing the equator at the time of perigee thruster burn, constraining the possible transfer orbits.

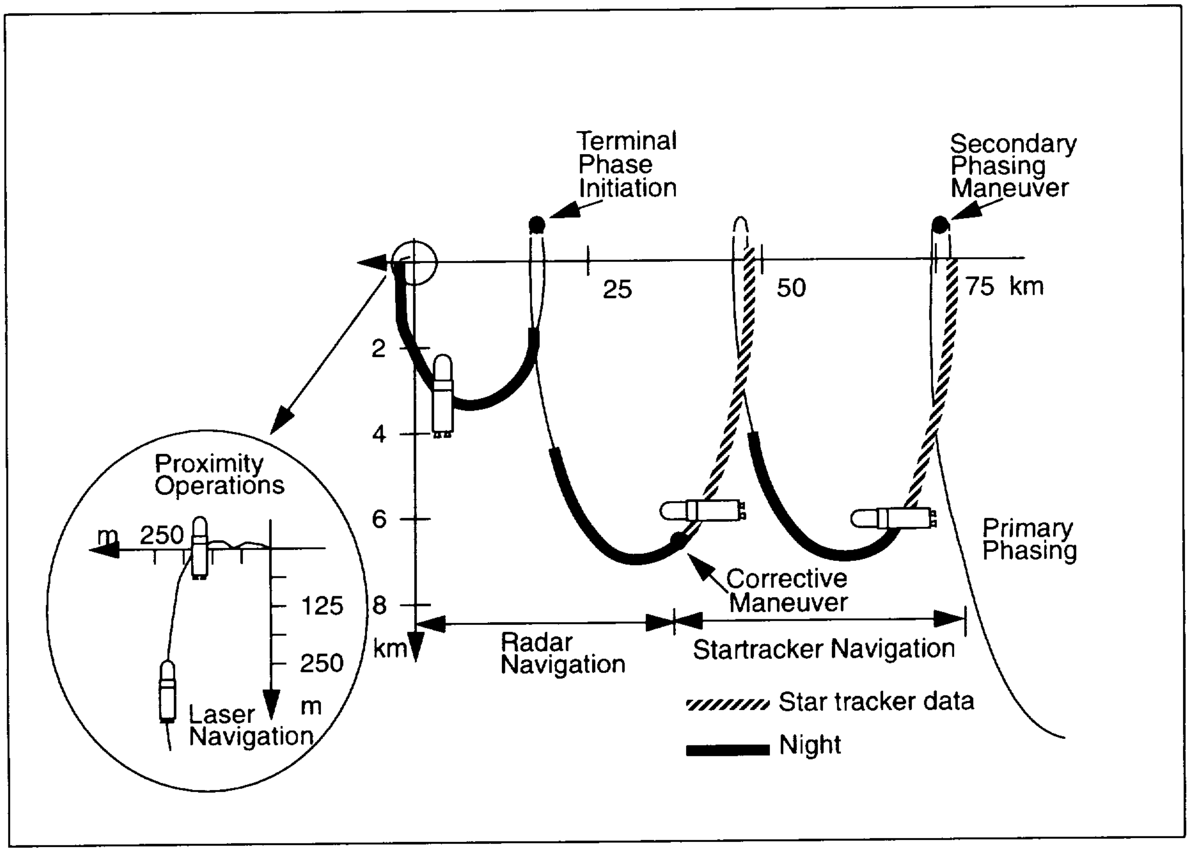

FIGURE 7.3 *Low Earth Orbit Rendezvous.*

The possible transfer orbits are further constrained due to lighting considerations. Because Freebird will not be able to acquire positional data of the target vehicle from multiple orbits, as it does in a LEO rendezvous, it will need to acquire this data as it approaches the target during orbital transfer. Freebird will be able to collect this data using its startrackers from up to 600km away if the lighting is right. Ideally, Freebird would be directly between the sun and target, as illustrated in Figure 7.4. This will certainly almost never be the case, meaning Freebird will need to be closer than 600km to the target before it will be able to detect the target. As Freebird is provided with more accurate positional data on the target, it will be able to make corrective burns.

At some times during the year, proper illumination of the target may be impossible during approach. This problem is still a critical issue for Freebird. The GEO rendezvous strategy is not fully defined yet, and as the Freebird mission is further developed, and a more rigorous analysis on rendezvous in GEO will have to be performed. GEO rendezvous may eventually require that Freebird make at least one complete orbit at GEO altitude, or it may require the consumption of more RCS fuel. This problem will have to wait until adequate research on rendezvous in GEO is made.

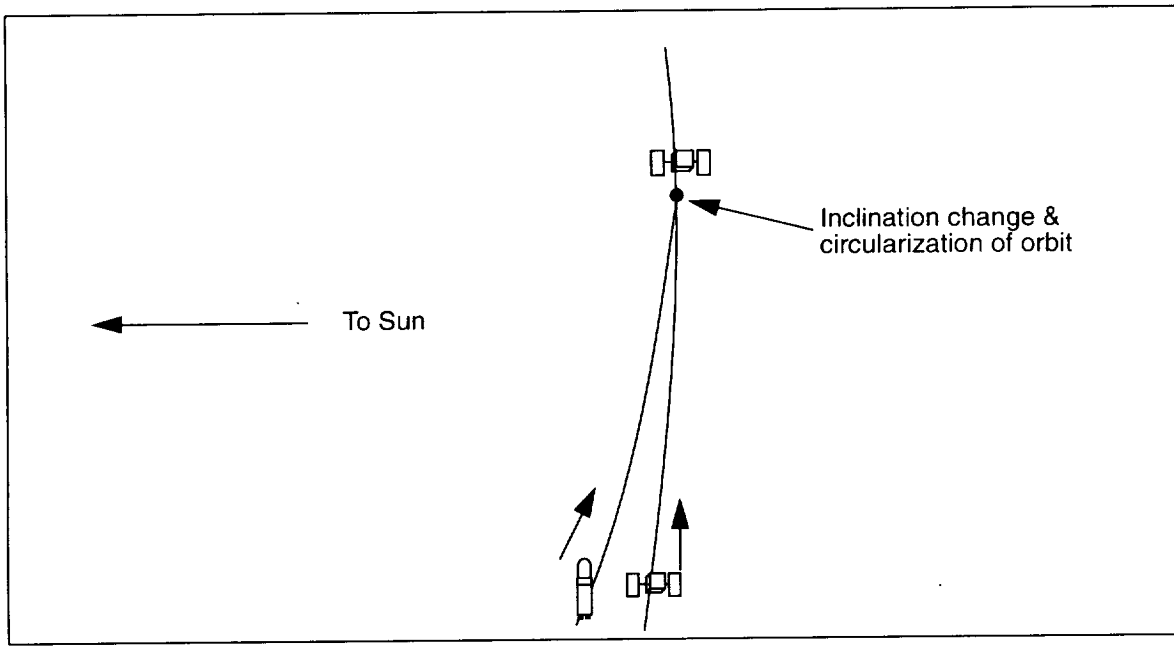

FIGURE 7.4 *Geosynchronous Orbit Rendezvous.*

7.4.5 Proximity Operations

Definition

Any maneuvers taking place within 1km of an asset are considered proximity operations (PROX OPS). This may include transition to specified offset positions, stationkeeping, approach, docking, flyarounds, and finally, after all mission objectives are satisfied, the separation of Freebird to a stand-off position. [4]

Strategy

The primary considerations for designing PROX OPS strategies are plume impingement, RCS fuel conservation, braking techniques, and maneuver targeting. [4] The general maneuverability requirements of Freebird enable most proximity strategies to be used without requiring any additional hardware. Therefore, exact strategies for most PROX OPS are beyond the scope of this project. However, the fact that Freebird will be flying a number of unmanned missions is an important system design driver which requires a high level of autonomy, including autonomous docking. Since this requires specific Freebird hardware and target configurations, autonomous docking will be discussed.

In addition to general considerations, ISSA has specific requirements for all spacecraft operating in its vicinity. The ISSA docking requirements specify that vehicles must control their attitude to within 3°, relative position to within 1cm, and relative velocity to within 1cm/s. In addition, primary RCS thrusters may not be fired in the direction of ISSA anywhere within 30m.

The on-board rendezvous radar provides navigation performance for rendezvous, but cannot support the final proximity operations phase at ranges less than 30m from the target because of hardware limitations. Historically, most docking has been done manually. In fact, both rendezvous and docking operations have required crew and ground involvement in the operation of integrated guidance, navigation, and control. Freebird, as mentioned before, will need autonomous or teleoperated docking capability for uncrewed missions. Traditionally, manual intervention has been required to mode the guidance, navigation, and flight control subsytems;

evaluate the navigation filter operation and measurement incorporation; initiate on-board maneuver targeting programs; and execute the maneuvers. During the final phase of rendezvous and docking, the pilot operations take precedence over nearly all of the on-board guidance, navigation, and targeting functions. The approach is flown manually using visual cues and raw radar data to determine the relative attitude and relative position of the target and the chaser spacecraft. The Freebird top-level requirement to perform uncrewed missions make mandatory on-board manual intervention into GNC system operations impractical.

To provide a high-accuracy sensor that can be used at close ranges, Johnson Space Center initiated the development of a PROX OPS relative laser navigation sensor (LNS). This sensor can accurately provide the relative range, range-rate, angles, and angle rates to the target at ranges typical of the end of a rendezvous mission phase all the way to docking. This sensor will be used both to enhance current manual proximity operations and to provide the improved navigation required for future uncrewed rendezvous and docking missions. In order for LNS to provide autonomous rendezvous capability, the target must have an isosceles triangular arrangements of reflectors on its surface, as shown in Figure 7.5. The LNS can use these reflectors to determine the target vehicle's relative attitude. It is reasonable to assume that some assets will not have the reflectors necessary for automatic rendezvous. In this case, the uncrewed mission will have to be teleoperated, and the docking camera will be responsible for determining the target vehicle's attitude. The docking cameras will also serve as a backup for the LNS.

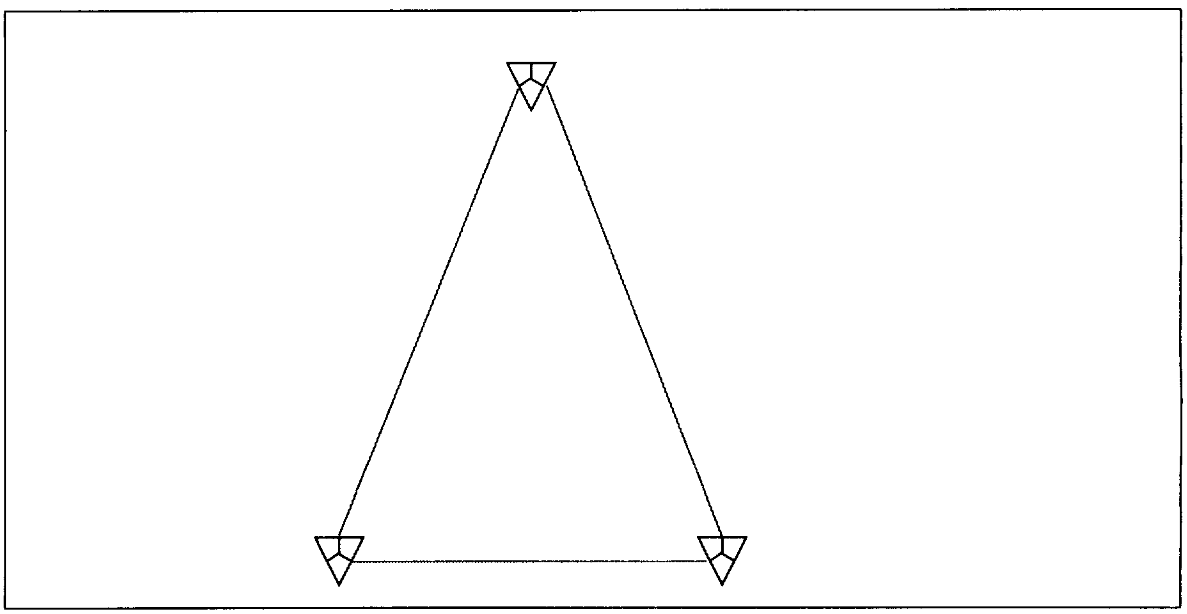

FIGURE 7.5 *Cube Corner Reflector Configuration for Laser Navigation Sensor Target Attitude Determination.*

When Freebird cannot dock with an asset, grappling techniques will be used. The closed circuit television (CCT) on the wrist of the RMS will give a visual image of the pin orientation of the grappling fixture on the target (see Figure 7.6). A central raised pin can be compared in distance to the other pins to judge the relative attitude of the grappling fixture. This image can then be analyzed by a human operator, as on the Space Shuttle, or via computer image processing for autonomous grappling, to line up the RMS with the grappling fixture.

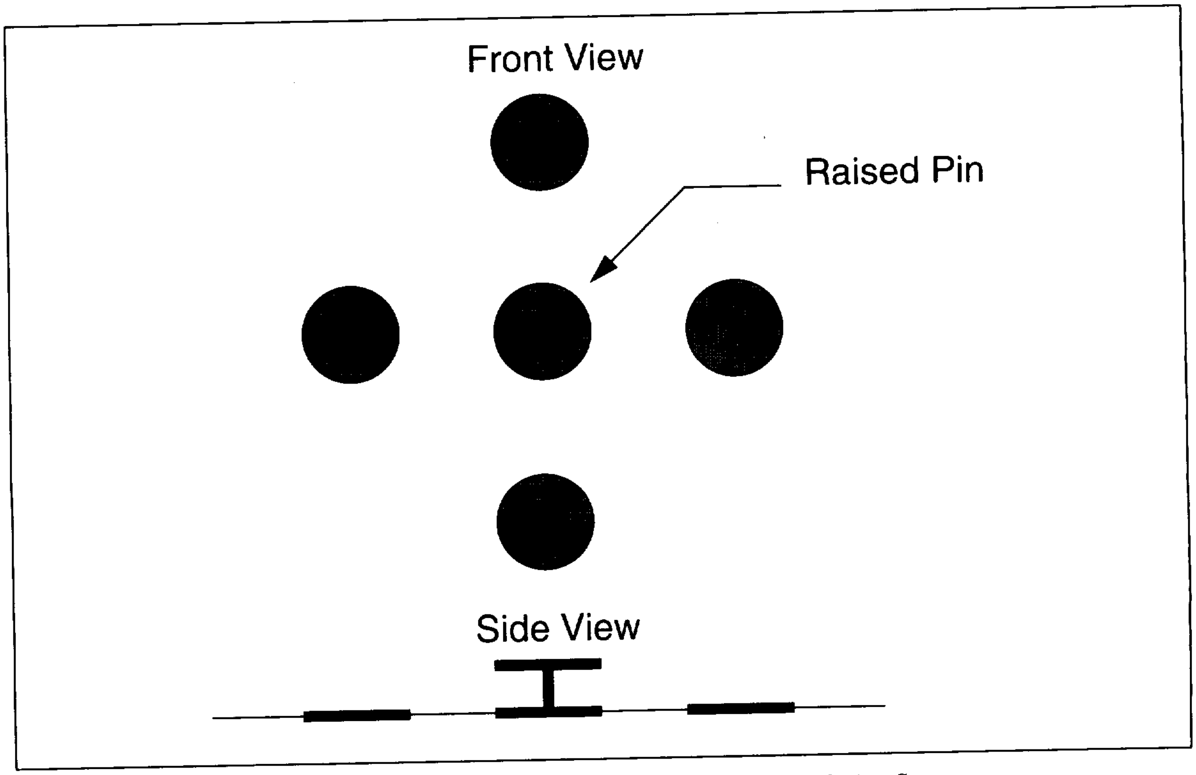

FIGURE 7.6 *Pin Configuration for Closed Loop Vision System on Remote Manipulation System.*

7.5 Attitude Determination and Navigation Hardware

The trade studies and eventual decisions on Freebird attitude determination and navigation are discussed in the following section.

7.5.1 Viable Attitude Determination Hardware

The sensors available to determine Freebird attitude are presented in Table 7.1. They consist of:

- Inertial-Measurement Units (IMUs)

  IMUs consist of gyros and accelerometers, and provide accurate measurements of rotation and translation. The drawback is that they are subject to gyro drift, and must be periodically updated.

- Sun/Moon sensors

  Sun/Moon sensors are accurate and lightweight. However, they do not provide full three-axis attitude determination. Frequent obstruction of the Sun or the Moon, especially in low-Earth orbit, can lead to data loss.

- Star Sensors

  Star sensors are high accuracy, but are expensive and bulky. They require an extensive star map data base for attitude determination.

- Earth sensors

  Earth sensors, also known as horizon sensors, detect the Earth's horizon using infrared devices. They only provide direction to nadir, and not full three-axis attitude determination. Their accuracy is relatively low, but can be improved using multiple Earth sensors. The Dual Cone Scanner (DCS), a type of Earth sensor, also provides altitude information by measuring the apparent diameter of the Earth.

- Microcosm Autonomous Planning System (MAPS)

  The MAPS software operates in conjunction with MANS (discussed in Section 7.5.2) to provide direction to any specified ground station or spacecraft. MAPS is not fully developed, and has never been flight tested.

TABLE 7.1 *Viable Attitude Determination Hardware*

Hardware	Advantages	Disadvantages
IMU	Completely internal High accuracy	Gyro drift (0.003°/hr to 1°/hr)
Sun/Moon sensors	Accurate (0.005° to 3°)	Sun/Moon not always in field of view
Star sensors	High accuracy (0.0003° to 0.01°)	Expensive Requires extensive star map data
Earth sensors	Inexpensive	Low accuracy (0.1° to 1°)
MAPS	Provides directions to and orbits of assets	Not fully developed Requires MANS

7.5.2 Viable Navigation Options

The options available to provide navigation information are listed in Table 7.2. They consist of:

- Ground-station tracking

  This is the standard method used for spacecraft navigation, and would not be difficult to implement. However, it requires costly operational ground support, and its accuracies range from a few kilometers in LEO to 50km in GEO.

- Microcosm Autonomous Navigation System (MANS)

  MANS uses information provided by Earth sensors and Sun/Moon sensors to determine orbital position and attitude, providing 3σ accuracies of 100m to 1.5km. It can be used in LEO and GEO, and can also use data from gyros, accelerometers, star sensors, and GPS. It has only recently been flight tested.

- Space Sextant

  The Space Sextant, developed by Martin Marietta, was flight tested in the late 1970s, but is no longer being marketed. It measures the angle between a star and the Moon to provide navigation data, and requires heavy telescopic instruments. It can be used in LEO and GEO.

- Inertial-Measurement Units (IMU)

  The rotational and translational data provided by the IMU can be numerically integrated to provide accurate navigation data. However, drift problems mean that the navigation data requires frequent updating. The Space Shuttle currently uses IMUs, updated using a star tracker.

TABLE 7.2 *Viable Navigation Hardware*

Options	Advantages	Disadvantages
Ground-station tracking	Most widely used method of navigation	Operations costs higher than autonomous navigation Low accuracy (3σ accuracy about 50km in GEO)
MANS	Accurate Autonomous	Only recently flight tested
Space Sextant	Accurate Autonomous	Heavy compared with other systems
IMU	Accurate translational information	Needs to be updated by another system due to drift

7.5.3 Viable Rendezvous Hardware

The available rendezvous sensors are listed in Table 7.2. They consist of:

- Star Tracker

  A Star tracker provides very accurate angular data from chaser to target vehicles. It can operate at ranges much further than radar, provided that the target is properly illuminated. There is a minimum range at which a star tracker can operate, about where the target begins to consume more than a few pixels on the star tracker charge coupled device (CCD).

- Rendezvous Radar

  Radar provides target vehicle angular, range and range rate data. However, the accuracy decreases significantly as the range increases.

- Laser Navigation Sensor (LNS)

  The LNS is currently being developed at Draper Laboratories for autonomous rendezvous and capture of satellites. As well as angular, range and range rate information, it can give attitude information on the target vehicle, which makes it attractive for automatic rendezvous. It is much more accurate than radar, but because of its limited range, it is limited to proximity operations.

- Docking cameras

  Docking cameras can be used to provide a low resolution angular, range, and range rate data for proximity operations. They are useful for manual rendezvous as well as teleoperated rendezvous.

TABLE 7.3 *Viable Rendezvous Hardware*

Options	Advantages	Disadvantages
Star Tracker	High accuracy (0.0003° to 0.01°) Usable at large ranges	Requires sunlit target vehicle No range or range rate data Not useful at close ranges
Rendezvous Radar	Determines range and range rate up to 50km	Resolution poorer than star tracker and LNS No attitude information for target vehicle
LNS	High accuracy Provides attitude information for target vehicle	Needs further development
Docking Cameras	Can be used for teleoperated rendezvous	Useful only at close ranges Poor resolution compared to other sensors

7.5.4 Selected Freebird Attitude Determination and Navigation Hardware

As was discussed in Section 7.3 on page 125, an autonomous navigation system was shown to be preferable over the standard ground tracking approach, due to the lower operational costs. Leaving aside those systems which do not function in GEO, the options for Freebird's navigation system are limited to MANS and Martin Marietta's Space Sextant. MANS was ultimately chosen for the Freebird design, because it is the cheapest, lightest, most accurate, and most technologically advanced option currently available. In addition, the MAPS software was also chosen because of its compatibility with the MANS software, and usefulness for the Freebird mission. Various other sensors were chosen based on their applicability to the Freebird mission and compatibility with MANS. The particular rationales behind the types and number of sensors chosen for Freebird are discussed below. Freebird's rendezvous strategy was modelled after the Space Shuttle's current strategy, so similar sensors were selected. A star tracker was selected for long range target detection, and a radar was selected for closer ranges. A Laser Navigation Sensor was selected for proximity operations, especially useful for automatic rendezvous and docking. Docking cameras were also selected for close ranges, which are useful for manual docking and necessary for teleoperated docking.

Microcosm Autonomous Navigation System (MANS)

Microcosm currently offers an accurate, low cost navigation software package that is applicable to all of the required orbits for Freebird. The Air Force Phillip's Laboratory's Technology for Autonomous Operational Survivability (TAOS) mission, launched March 13, 1994, is currently flying the first flight test of the MANS software, used in conjunction with two Dual Cone Scanners with Sun/Moon fans. [5] The Freebird navigation system was modelled after the navigation system on the TAOS mission, using the MANS software as well as four Dual Cone Scanners with Sun/Moon fans.

Freebird simultaneously runs three MANS programs for redundancy. Each MANS acquires data provided by Freebird's Dual Cone Scanners. With information on altitude, direction to nadir, and directions to the Sun and Moon, MANS is able to determine an orbit for Freebird. [6] With subsequent data coming in, MANS uses a Kalman filter to better extrapolate Freebird's orbit. In simulations of MANS using two DCSs with both the Sun and Moon present, the 3σ positional accuracy was determined to be 150m at LEO. GEO accuracies are 35km, due to increasing errors in altitude determination with increasing altitude.

Of course, the Sun and Moon will not always lie within the field of view of the DCSs. The Sun/Moon fans will lose coverage when the Sun or Moon are: 1) eclipsed by the Earth, 2) blocked by Freebird, 3) outside the fan limits, or 4) too close to the Earth horizon. The most significant of these will be the time which Freebird spends in the Earth's shadow, which represents about 40% of the time in LEO. When only the Sun or the Moon is within the DCS field of view, the positional accuracy of MANS drops down to 1km. Without data from either the Sun or the Moon, MANS is not able to model any changes in Freebird's orbit, so the accuracy becomes significantly poorer with increasing time. For this reason, the MANS on Freebird also uses the data provided by the two IMUs. The drift rate of the IMU is low enough to keep MANS updated and accurate for whatever length of time Sun and Moon data are not available. The IMU will also be able to immediately update MANS whenever the RCS or OMS thrusters are fired and Freebird's orbit is changed.

Microcosm Autonomous Planning System (MAPS)

MAPS is an extension of MANS. It is a software system which uses the navigation and attitude information provided by MANS to provide the direction to any ground station or spacecraft. [2] It requires a database containing the locations of ground stations, as well as the orbits of other spacecraft that Freebird will need knowledge on. MAPS will be useful in determining antennae pointing directions. It will be especially useful in determining the orbits of target spacecraft, which is necessary for rendezvous targeting. MAPS has yet to be developed, so at this point, its functionality is rather flexible. For the Freebird mission, MAPS will be expanded to accept rendezvous sensor data, in order to improve the accuracy of its target orbit determination. MAPS will also provide data on the time and duration of eclipses.

Dual Cone Scanner with Sun and Moon Fans

The rational for selecting the Dual Cone Scanner (DCS) with Sun and Moon Fans is explained in the MANS section. It is the same sensor used aboard the current TAOS mission.

The Dual Cone Scanner (DCS), manufactured by Barnes Engineering Division of the EDO Corporation, is a modified version of the Conical Earth Sensor (CES). [7] The CES is a standard scanning Earth sensor which provides accurate nadir determination. By using two conical scans of the Earth rather than one, the DCS has the advantage over the CES of added redundancy, increased accuracy, and the ability to determine spacecraft altitude by measuring Earth radius. In addition, the DCS does not experience the sun-on-horizon errors common to the CES, because it utilizes two scanning cones set at different angles. For the Freebird mission, the DCS has been further enhanced by the addition of two fan sensors. Alone, DCS attitude determination is limited to pitch and roll. With the addition of Sun and Moon sensors, the DCS provides full attitude determination. Figure 7.7 illustrates the mechanical layout of the modified DCS.

The DCS consists of two optical scanning paths which rotate around a central axis set at different angles with respect to the rotation axis. The fields of view of these scanning paths are illustrated in Figure 7.9. The incoming light from the two paths is directed onto a pyroelectric detector. The angles chosen for the optical scanning paths are 60° and 72°, optimized for Earth width sensing during orbital transfer from LEO to GEO. The two scanning paths provide two Earth scans, or four horizon crossings, per rotation. Roll and pitch can be determined by measuring the changes sensed in the Earth pulse lengths. Apparent Earth radius, which can be converted to spacecraft altitude, can be determined by comparing the Earth pulse lengths provided by each scanning path.

Sun and Moon attitude are determined by two fans which rotate around the spin axis, and direct visual light onto a silicon photodiode. These fans are also shown in Figure 7.9. The fans are tilted 15° with respect to the spin axis, and are separated from each other by 5° at the equator. They each have a 70° field of view.

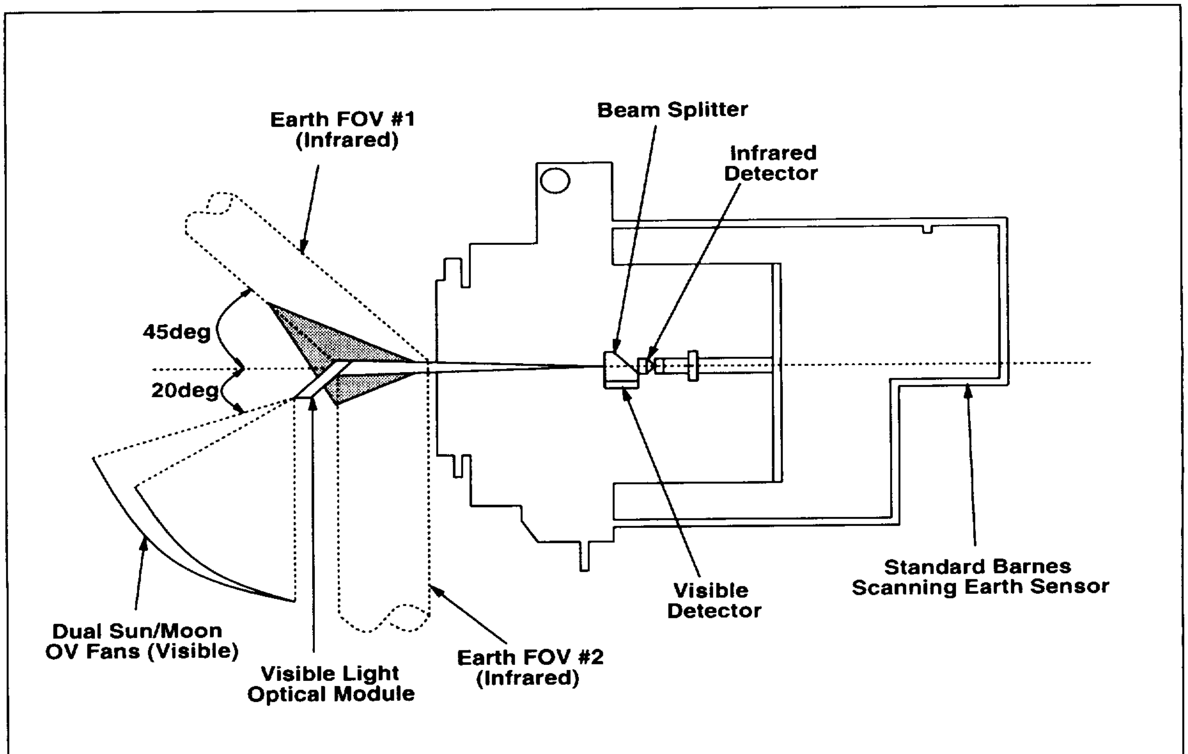

FIGURE 7.7 *Mechanical Layout of the Dual Cone Scanner with Sun and Moon Fans [6].*

Freebird has two pairs of Dual Cone Scanners mounted around the perimeter of the Smart Box. The DCS mounting configuration is illustrated in Figure 7.8. Each DCS is oriented 120° from the other DCS it is paired with. Nadir should be pointed directly between the DCS pair for optimal Earth horizon sensing in both LEO and GEO. The second DCS pair exists for redundancy, and is placed 180° from the first pair. Use of two DCSs at the same time provides increased accuracy over the use of only one sensor. For LEO, the 1 second time average accuracies (3σ) achieved by a DCS pair are 0.05° for attitude and 300m for altitude. These accuracies are achieved for almost 100% of spacecraft orientations. In GEO, altitude and attitude can be determined for 63% of spacecraft orientations, and the 3σ accuracies are 0.02° for attitude and 35km for altitude.

The specifications of the DCS are listed in Figure 7.4.

TABLE 7.4 *DCS Specifications[8].*

Parameter	Characteristic
Size [cm]	10.4 dia × 20.1 long
Mass [kg]	4.54
Power [W]	10
Operating Temperature [°C]	-20 to +60

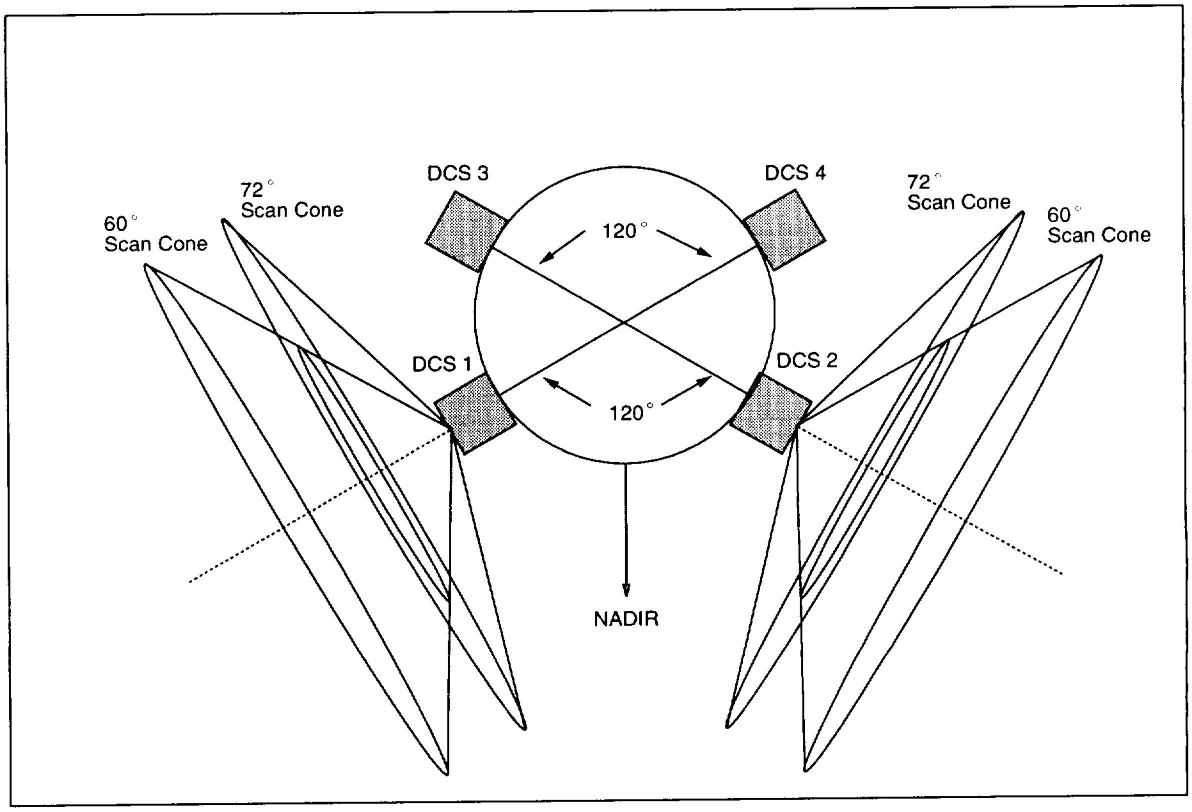

FIGURE 7.8 *Dual Cone Scanner Mounting Configuration.*

Miniature Inertial Measurement Unit (MIMU)

An Inertial Measurement Unit was selected for the Freebird mission for the reasons given in the MANS section. Freebird uses Honeywell's MIMU for accurate translational and rotational data. The MIMU was selected because it is a relatively lightweight, inexpensive, and accurate IMU. The MIMU consists of three Honeywell GG1320 Ring Laser Gyros (RLGs), which measure changes in angular rotation, and three Allied Signal QA2000 Q-Flex accelerometers, which measure changes in linear motion[9]. The MIMU also includes Honeywell's radiation hardened RH-1750 processor for internal processing. Two MIMUs are used in Freebird for redundancy. Specifications for the MIMU are listed in Table 7.5.

CT-611 Space Shuttle Star Tracker

Ball Aerospace's CT-611 Space Shuttle Star Tracker has been selected for the Freebird design. Its specifications are listed in Table 7.6. The star tracker will be used strictly for orbital rendezvous, as described in Section 7.4.4 on page 129. The star tracker will acquire and track a sun illuminated target of stellar magnitude as dim as 3.9, meaning that a properly sunlit spacecraft may be detected up to 600km away. The CT-611 has

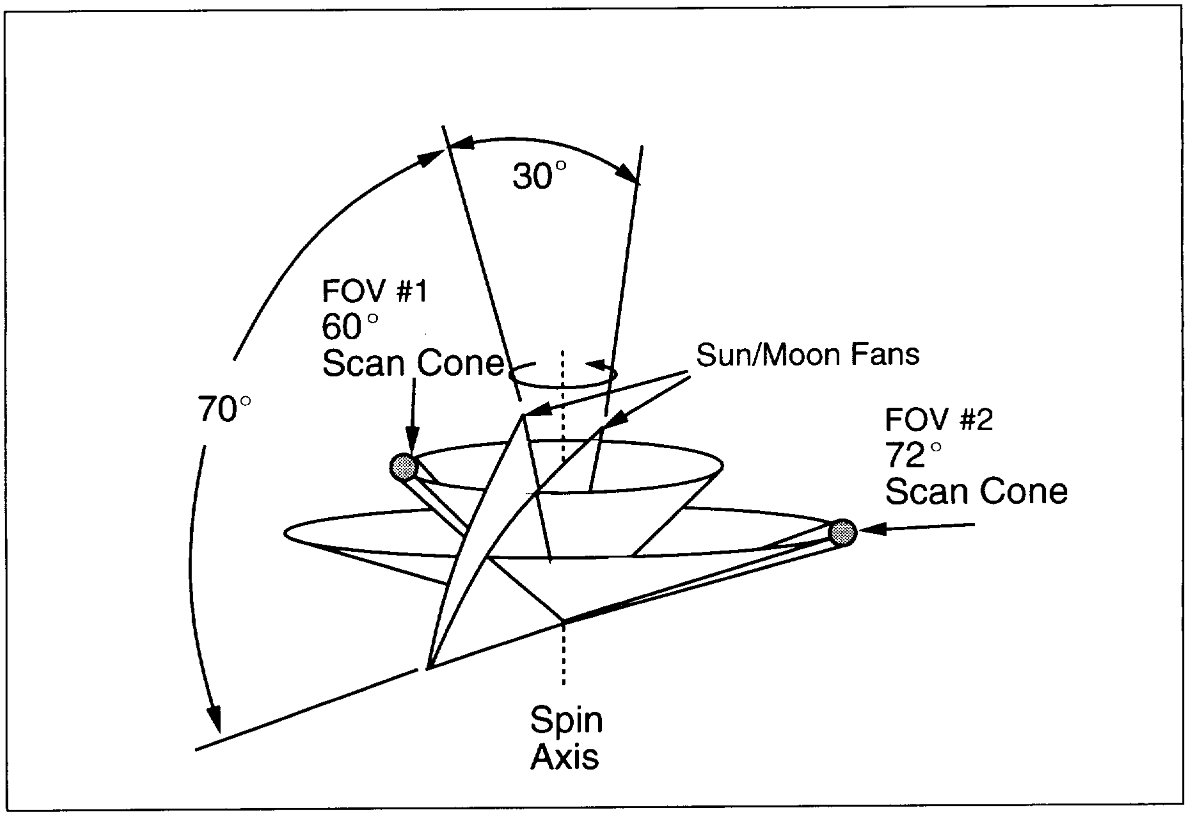

FIGURE 7.9 *Earth, Sun, Moon Fields of View.*

TABLE 7.5 *MIMU Specifications [9]*

Parameter	Characteristic
Size [cm^3]	2260
Mass [kg]	1.92
Power [W]	18
Maximum Angular Acceleration [rad/s^2]	1,000
Angular Drift Rate [rad/hr]	0.01
Maximum Acceleration [g]	25
Operating Temperature [°C]	-30 to 70
Radiation [Krad]	100

the capability of identifying the target within its 10° x 10° field of view. Freebird's target search technique will be similar to the method used aboard the Space Shuttle. [4] Initially, navigation will send the coordinates of the target to the star tracker, and the star tracker will search these coordinates for the target within a 1° x 1° square for up to four seconds. If the target is not found due to navigation errors, the star tracker will search its entire field of view for 20s before aborting. The Freebird design calls for two star trackers for redundancy. They are each located on the smart box 180° apart. Star tracker locations are illustrated on Figure 3.16 on page 75.

Each star tracker will be mounted on a Ball Aerospace LC1060 Control Gimbal. Specifications are listed in Table 7.7.

TABLE 7.6 *CT-611 Space Shuttle Star Tracker Specifications [10]*

Parameter	Characteristic
Field of View [°]	10 x 10
Sensitivity Range [M]	-7 to +3.9
Accuracy [arc s]	15
Update Rate [Hz]	10
Dimensions [cm]	21.2 x 16.8 x 28.58
Mass [kg]	6.79
Power [W]	12.5
Operating Temperature [°C]	+10 to +60
Radiation [Krad]	100

TABLE 7.7 *LC Control Gimbal Specifications [11]*

Parameter	Characteristic
Yoke Diameter [cm]	0.17
Weight [kg]	9.7
Elevation Travel Range [°]	60
Azimuth Travel Range [°]	180
Accuracy [μrad]	96
Operating Temperature [°C]	-13 to +35

Rendezvous Radar

A rendezvous radar is needed to supplement the star tracker's angular data with range and range rate data in addition to providing angular data when the star tracker cannot be used, such as when the target is too big in the field of view, too bright, or too dim. The large position uncertainties in GEO make it desirable to have a long range radar, i.e., able to pick up a 1m^2 target at a range of 100km, for performing rendezvous in GEO. Weight considerations indicate that it would be more efficient for the rendezvous radar to use Freebird's communications parabolic antennas rather than have its own radar dish. The Space Shuttle currently uses this method in its Integrated Radar And Communications System. The drawback of using the same antennas is the loss of the use of one of the antenna for communications during radar operation. Communications currently uses S-band, which is inadequate for the type of range needed from the rendezvous radar. The rendezvous radar would require its own transponder operating in at least Ka band. Preliminary calculations indicate that a long range space based radar is feasible, although such systems have yet to be built since space missions have not previously required this kind of range for rendezvous radar. Rough calculations of the rendezvous radar parameters required for Freebird are listed in Table 7.8. These parameters were calculated using equations in *Hughes Aircrafts' Introduction to Airborne Radar* [12].

TABLE 7.8 *Rendezvous Radar Parameters*

Parameter	Characteristic
Band	Ka
Frequency [GHz]	40
Bandwidth [kHz]	15
Transmitter Power Output [W]	150
Antenna Diameter [m]	1.2
Range [km]	100

Laser Navigation Sensor (LNS)

The Laser Navigation Sensor is part of an effort by NASA to replace its current reliance on manual docking and rendezvous. [13] LNS provides accurate range and range rate data from 1km all the way to docking. LNS also provides target attitude information when Freebird is within 55m of the target and the target vehicle has been outfitted with three cube corner reflectors. The reflectors must be mounted on the satellite at the corners of an isosceles triangle a few feet in size, with the long edge of the triangle around 1.5 feet longer than the other two legs. LNS tracks the set of reflectors on the satellite, and by comparing their measured orientations to the known configuration of the reflectors, LNS can calculate the attitude of the target spacecraft. Current specifications of the LNS are listed in Table 7.9.

Freebird has two LNSs for redundancy. The LNS layout is illustrated on Figure 7.5 on page 132.

TABLE 7.9 *Laser Navigation Sensor Specifications [3]*

Parameter	Range of Operations	Laser Accuracies (1σ)
Range (R) [km]	0 to 1	0.001 R
Range Rate [m/s]	0.9	0.001 R
Attitude [°]	28	0.3
Attitude Rate [°/s]	1	0.0002

Docking cameras

Teleoperated docking and satellite capture will require a video downlink from a docking camera aboard Freebird. When Freebird is within 1km of the target, ground controllers will be able to use docking camera data to estimate range to about 3%. This is accomplished by overlaying a calibrated ranging ruler on the camera data, which is a technique used by the Space Shuttle through its Closed Circuit Television. [4] In addition to teleoperated missions, the docking cameras will be useful on crewed missions, as well as serve as a backup for the LNS. Freebird has two docking cameras for redundancy. Camera mounting is illustrated on Figure 3.16 on page 75.

7.6 Guidance

The guidance system is responsible for controlling the position of the vehicle center of mass. The main engines will only be used for major orbital transfers, due to the limited number of starts available. Therefore, the RCS system will have two 4000N secondary thrusters for small to medium size orbital transfers in addition to a set of smaller primary thrusters for small perturbation translational control. In order to meet the docking velocity control requirement of 1cm/s, the primary RCS system must have a minimum impulse of no greater than 50Ns in any one direction (assuming a minimum vehicle mass of 5000kg). A complete discussion of the RCS system design will be delayed until the attitude control requirements for the RCS system are discussed in Section 7.7.2.

7.7 Attitude Control

The attitude control system is responsible for stabilizing and controlling the attitude of Freebird. When the main engines are in use, they will be gimballed to correct for thrust imbalances and to control the attitude of Freebird. At all other times, the main attitude control system will provide 3-DOF rotational control.

7.7.1 Hardware Trade Study

The large vehicle mass combined with the maneuverability requirements of Freebird restricted the choice of attitude control actuators to those that are capable of high torque levels, namely control moment gyros (CMG's) and thrusters. Their advantages and disadvantages are summarized inTable 7.10. The main advantage of using CMG's would be realized by using one CMG to replace the four thrusters, and their fuel, that would otherwise need to be added to the translational thrusters to provide control about the roll axis. Since the thrusters necessary for translational control are also adequate for pitch and yaw control, the main additional savings from using CMG's for control of theses axes would be fuel savings and extended thruster life. In addition, the use of CMG's instead of thrusters reduces the contamination due to thruster plumes. The main disadvantages of CMG's are their large mass, 150kg each or 600kg for a complete set of four, and their relatively limited control authority, about 4000Nm of torque each.

TABLE 7.10 *Attitude Control Hardware*

Hardware	Advantages	Disadvantages
Control moment gyro (CMG)	No contamination Completely internal Immune to vehicle configuration No fuel consumption	~4000Nm maximum torque each Minimum of 1 per axis plus a total of 1 backup ~150kg mass each
Thruster	High torque levels Pitch and yaw controllable by translation thrusters	Roll control requires 4 additional thrusters Consumes fuel Plume impingement on vehicle and assets Location complicated by modular vehicle configuration Less controllability

7.7.2 Selected Control Hardware

The hardware for the attitude control system was chosen primarily on a minimum mass basis. Since the bulk of the RCS fuel will be used for small to medium orbital maneuvers during rendezvous, i.e., attitude control requires relatively small amounts of fuel (~50kg between refuelings), using CMG's would result in a net mass

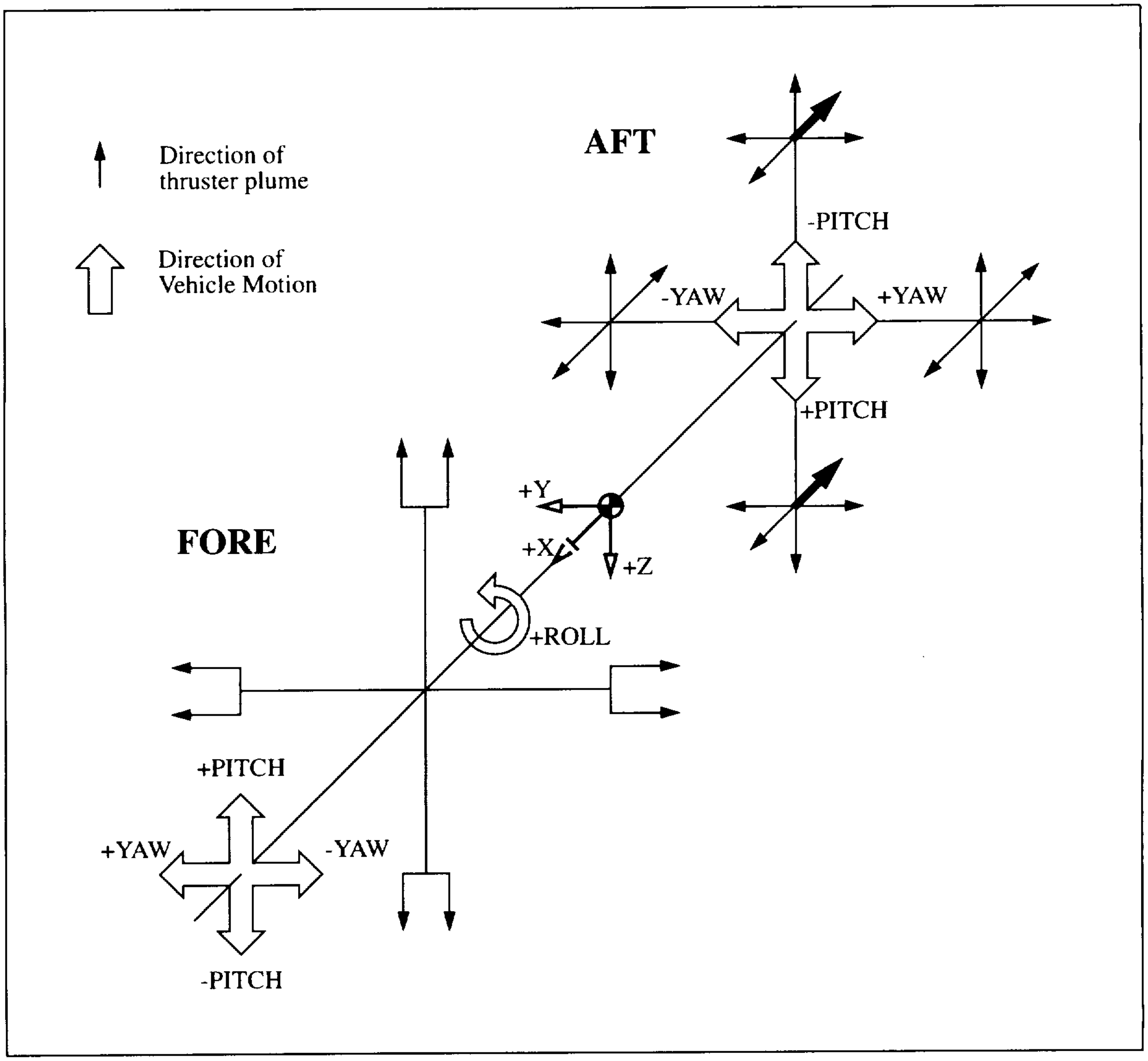

FIGURE 7.10 *Approximate Reaction Control System Thruster Orientations and Locations.*

increase. Therefore Freebird will use only RCS for its attitude control. There will be twenty-six primary 490N RCS thrusters and, as previously mentioned, there will be two secondary 4000N RCS thrusters. See Figure 7.10 for the approximate location and orientation of the RCS thrusters. Note that twenty of the twenty-eight thrusters are located in the aft section of Freebird. This uneven distribution of thrusters was the result of efforts to minimize thruster plumes around the forward section of the vehicle which comprises the "smartbox", crew module, and payload. Each of the four thruster sets in the forward end of the propulsion pack consist of singly redundant pairs. The y-axis and z-axis thrusters in the aft section are arrayed in sets of three: one on the roll axis and one to either side as in Figure 7.11. This configuration enables roll control by the off axis pairs and also allows firing of y-axis and z-axis thrusters in single fore-and-aft pairs for precision control or double pairs for high authority control. The x-axis thrusters are redundant in four pairs symmetrically arrayed about the x-axis for several reasons. First, the x-axis thrusters will be more heavily used than the y-axis and z-axis thrusters. The x-axis thrusters will do most of the orbital correction burns, although they won't be used for attitude control. The higher work load on the x-axis thrusters makes it necessary to use multiple thrusters to spread out the burn time so that the x-axis thrusters don't wear out faster than the other thrusters. Second, having redundant pairs of thrusters allows for efficient torqueless translational control even if one of the thrusters fails. In addition, the retro-rockets thrusting in the +x direction will be angled off axis by 45° and thus will only have about 70% of their maximum thrust aligned along the x-axis, increasing the work load for a given desired change in velocity. Lastly, it will be desirable at times to lock out the use of one of the pairs of retro-rockets if an RMS is in one of the plume cones for that pair. Thus having redundant pairs allows the control system to avoid damaging either RMS while not sacrificing controllability of the vehicle. See Chapter 5 on page 89 for more information on the RCS thruster system.

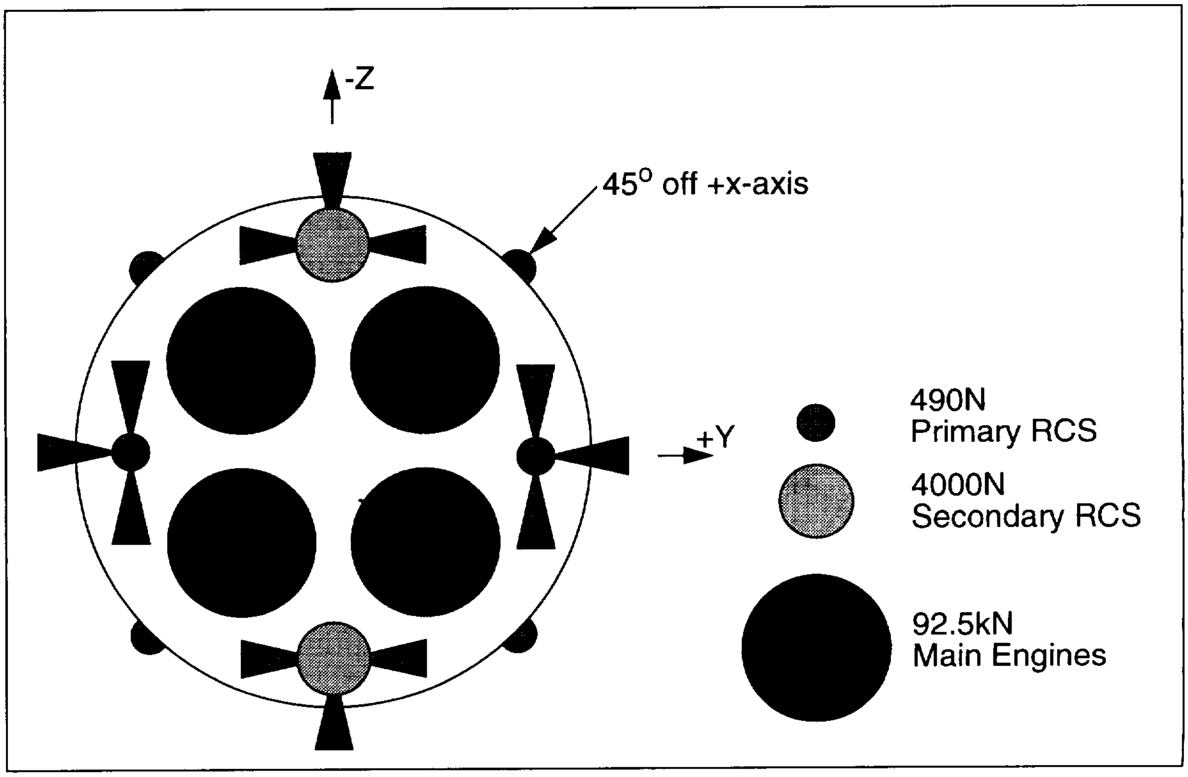

FIGURE 7.11 *Aft End Thruster and Main Engines Layout.*

7.8 Computational Requirements

All GNC software will run on the main computer system supplied by C^3T. The MANS software system requires a throughput of 1 MIPS and all other GNC software requires approximately 10 MIPS. The control system will run at 100Hz to achieve the desired minimum impulses from the RCS thrusters.

7.9 Failure Modes

The failure modes for the GNC system are listed in Table 7.11. It can be seen that there are no failure modes that are life-threatening and only two that are mission critical. This is due to the high level of redundancy and the level of autonomy of the GNC system. This level of redundancy of each component and also the redundancy achievable by the use of other components to perform the failed function leads to a very high reliability. In many cases, operation or information from ground can bring the criticality level down to only a telemetry critical level and thus makes the mission possible.

TABLE 7.11 *Failure Modes*

Component	Failure Mode	Effect	Prevention or Response	Criticality	
				Crewed Config.	Teleop.
DCS	Single failure	Loss of one backup	Switch to backup	3	3
	Double failure	Loss of both backups	Switch to backup	3	3
	Triple failure	Loss of accuracy and altitude data	Use ground tracking for altitude data	3	3
	Complete failure	Loss of high precision updates to MIMUs	Use star tracker and ground tracking data to update MIMUs	3	3
MIMU	Single failure	Loss of accuracy and backup	N/A	4	4
	Double failure	Accuracy degradation	Rely on DCS, star tracker and ground tracking for data	3	3
MANS	Software failure	Loss of accurate navigation data	Rely on ground tracking for navigation	3	3
MAPS	Software failure	Loss of accurate target vehicle data	Rely on ground tracking for data	3	3
Control Software	Software failure	Loss of ability to control vehicle	Uplink backup software from ground	3	3
Star Tracker	Single failure	Loss of backup	Switch to backup	3	3
	Double failure in LEO	Loss of accurate target vehicle data	Rely on ground tracking data	3	3
	Double failure in GEO with active target	Loss of star tracker data	Rely on target to provide data	3	3
	Double failure in GEO with passive target	Loss of accurate target vehicle data	Return to ISSA	2	2

TABLE 7.11 *Failure Modes*

Component	Failure Mode	Effect	Prevention or Response	Criticality Crewed Config.	Teleop.
Rendezvous Radar Transponder	Single failure	Loss of backup	Switch to backup	3	3
	Double failure in LEO	Loss of rendezvous radar data	Rely on ground tracking data to rendezvous	3	3
	Double failure in GEO	Loss of ability to rendezvous with satellite	Return to ISSA	2	2
LNS	Single failure	Loss of backup	Switch to backup	3	3
	Double failure	Loss of LNS data for docking	Dock using docking cameras	3	3
Docking Camera	Single failure	Loss of backup	Switch to backup	3	3
	Double failure	Loss of visual docking ability	Rely on LNS to dock	3	3

A complete failure of all DCS units would be telemetry critical. It would result in the loss of accurate attitude and altitude information which is normally used to provide accurate data to MANS. The MIMUs, which suffer drift, provide rotational and translational information to MANS, and thus when used alone tend to provide inaccurate updates to MANS. In response to the DCS failure, the combination of information from the star tracker and ground tracking can update the data fed to MANS and thus increase the accuracy level acceptably to complete the mission.

If both MIMUs happened to fail this too would be only telemetry critical. The DCS would be able to provide attitude and altitude information, but could not update MANS as to rotational and translational movement of the vehicle. The DCS are also not useful if the sun and moon are out of the field of view. Thus ground tracking and star tracker data must also be used to recover from the failure and complete the mission.

A software failure of either MANS or MAPS would again be telemetry critical. Such a failure would only be likely to occur if there was a bug in the software, assuming that the computers they run on are functional. In this case, ground tracking data could be used for navigation and target vehicle data.

If the control software had a bug in it and failed, the vehicle would no longer be controllable autonomously, but could be controlled from the ground. A more optimal solution would be to have ground control uplink a backup control software package, thus allowing the vehicle to be autonomous once again and complete the mission.

There are several different failure modes possible resulting from the loss of both star trackers. If the loss were to occur in LEO, then ground tracking data would be accurate enough to allow rendezvous with the asset. This information would get the vehicle within range of the asset to be able to switch over to the rendezvous radar and complete the mission. If the failure were to occur in GEO when trying to rendezvous with an active target, the mission could still be completed by using the target vehicle to provide accurate data on it's position, through the use of a radar signal. However, in GEO when trying to rendezvous with a passive target there would not be accurate enough target position information to rendezvous with it and would result in abort of the mission.

Loss of both rendezvous radar transponders, also results in two possible failure modes. Loss in LEO, would not affect the mission, since ground tracking data could provide accurate information within the range of the LNS to take over and perform the necessary proximity operations. Loss of star tracker data in GEO would however result in abort of the mission, once again due to the lack of accurate target position information.

Lastly, a failure in either the LNS or the docking cameras does not affect the mission and docking is still possible since the other can be used. In the case of an LNS failure docking can be performed visually by the crew or by a teleoperator. If the docking cameras were to fail the LNS has the capability to dock autonomously.

7.10 Summary Budgets

A summary of the components comprising the GNC system are listed in Table 7.12 along with mass, power, and cost budgets.

TABLE 7.12 *Summary Budgets*

Component	#	Total Mass [kg]	Avg. Power [Watts]	Peak Power [Watts]	Hardware Cost [$k]
DCS	4	20	20	20	$2 396
MIMU	2	10	40	40	$1 238
MANS	3	0	0	0	$7 800
MAPS	3	0	0	0	$10 400
Star Tracker	2	50	2	15	$6 190
Rendezvous Radar Transponder	2	20	60	60	$2 866
LNS	2	40	40	110	$5 961
Docking Camera	2	10	50	100	$423
Totals	20	150			$37 103

The GNC telemetry requirements are listed in Table 7.13.

TABLE 7.13 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital	Command
DCS (x4)	Vehicle State	1	4	D-16 bit	
	Status	1	1	D	Sensor Utilized
	Rotation Rate	1	4	A	
MIMU (x2)	Vehicle State	1	10	D-16 bit	
Star Tracker	Target Vector	1	10	D-16 bit	

TABLE 7.13 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital	Command
Rendezvous Radar Transponder (x2)	Signal Strength	1	1	A	Coherent Mode
	Loop Stress	1	1	A	Enable Stand-by
	In-Lock	1	1	BiLevel	PRN ON/OFF
	Power out	1	1	A	Subcarrier ON/OFF
	Temperature	1	1	A	
LNS (x2)	Target State	1	10	D-16 bit	
Docking Camera (x2)	Status	1	1	D	Camera Selection

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Chapter 8

Command, Communication, Control and Telemetry

8.1 C^3T Requirements

The Command, Communication, Control and Telemetry (C^3T) system is responsible for processing all commands and communications, as well as monitoring vehicle health with a certain reliability.

- The command subsystem must receive or generate and process all commands.
- The command subsystem must process or store all required data.
- The telemetry subsystem must monitor all health and status signals generated by Freebird.
- The telemetry subsystem must react appropriately to all monitored faults.
- The telemetry subsystem must forward all health and status data to the communications subsystem for downlink.
- The communications subsystem must support communications with external sources, astronauts engaged in Extra Vehicular Activity (EVA) and astronauts inside the vehicle.
- The communications subsystem must be capable of receiving voice and command transmission through uplink, and transmitting voice, telemetry and video through downlink.
- Provide a system reliability of 0.999.

8.2 C^3T Interfaces

The C^3T system must interact both with the other systems on Freebird and systems external to the spacecraft. Figure 8.1 shows the functional block diagram for the overall C^3T system. Each system of the craft will be interfaced with the C^3T system for command and telemetry data. Additionally, Freebird will engage in voice and data transmission to/from ground, and downlink video to groundstations when under teleoperated control.

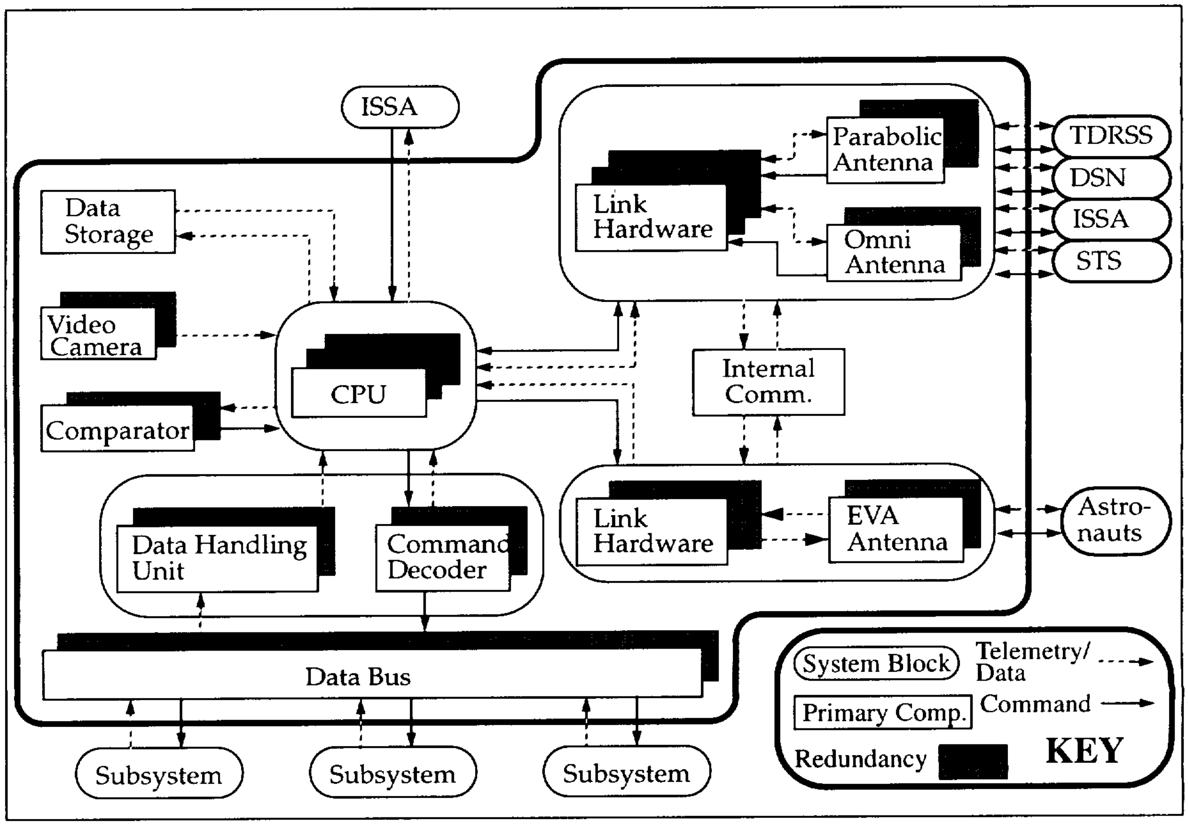

FIGURE 8.1 *Command, Communication, Control and Telemetry Functional Block Diagram.*

8.2.1 Command Subsystem

The Command subsystem receives and processes commands from several sources. It responds by generating new commands which can be sent to other systems for control, or direct some subsystem internal process such as self-testing, re-programming, shutdown, etc. On Freebird, the command system is implemented in the central computers. Two major aspects of the command subsystem are the command decoder and the data handling unit. These are shown apart from the central processing unit (CPU) in the functional block diagram for the sake of clarity. The command decoder receives all incoming command messages and performs all necessary processing on them before sending them to the data bus. The data handling unit is responsible for routing data on the bus and for transferring telemetry data to the central computers.

The command and control subsystem must interface with other systems on Freebird in order to receive commands and control the craft. The interfaces can be grouped into two categories; command-source and command-output. The functional block diagram shows that command-source interfaces are with the communication subsystem and comparators. Interface with the communication subsystem is to receive commands from sources external to Freebird. Another major source of commands is astronauts onboard Freebird during a crewed mission. They interface with the command subsystem through keyboards, joysticks and terminals; all of which are connected to the central computers. Astronauts can also interrupt the command subsystem directly using a hard-wired self-test mechanism.

Command-output interfaces are with all other systems in the craft using a linear data bus. Command signals on the bus are routed to and picked up by the targeted system which interface with the vehicle at the component level.

All command subsystem components have redundancy in the vehicle.

8.2.2 Telemetry Subsystem

The telemetry subsystem reads all health data from the various systems. The subsystem will consist of the hardware necessary to read the many telemetry signals generated by Freebird. Monitoring is done in the following manner: if the signal is analog, it is converted to digital; the CPU then compares the value to its accepted range, and performs any necessary actions; and the telemetry data is sent to the communications subsystem for downlink.

Additionally, if the mission is crewed, important telemetry data will be continuously sent to the crew's monitors. If there is a malfunction in any component, that component's data will also be displayed.

8.2.3 Communications Subsystem

Freebird must communicate with various external sources, as well as providing for internal communications. The communications subsystem interfaces with International Space Station Alpha (ISSA) for docking purposes, as well as a communication link. Rendezvous with the Space Shuttle requires compatible communication capabilities. In Low Earth Orbit (LEO), the vehicle will utilize the Tracking and Data Relay Satellite System (TDRSS) for communication. In geosynchronous orbit (GEO), the vehicle will communicate through the Deep Space Network (DSN). Astronauts engaged in EVA outside Freebird will communicate through a separate system. Internally, the vehicle supports communications between crew members.

The communication bands that Freebird utilizes correspond to those supported by ISSA and TDRSS as well as the Space Shuttle. The primary frequency used for communication is S-band, which corresponds to 2.025-2.120 GHz for uplink and 2.2-2.3 GHz for downlink. DSN operates on S- and X-band frequencies, and Freebird communicates with DSN through S-band while operating in GEO. The EVA suits utilize two UHF frequencies, 296.8 MHz and 259.7 MHz [1].

For internal communications, the astronauts have headsets which allow them to talk with each other and receive information from external sources during maneuvers.

The communications subsystem consists of the transmit/receive system and various links within Freebird. The communications subsystem receives inputs routed through the command and data handling system, or directly from the data storage system. The signal is then routed through one of the redundant transponders, out to the transmit RF switch, and then to one of the antennas. All of the components in the subsystem have been designed with redundant back-ups.

8.3 Command

The command subsystem processes all received or generated commands and data. Data that cannot be processed when received is stored until processor time is available. The subsystem also monitors the validity of the commands received or generated. After verifying a command, the subsystem takes the appropriate actions.

8.3.1 Command Architecture

The central processing unit receives information from, and outputs commands to, a variety of sources. In order to accommodate all of the inputs and outputs, the system could have a variety of architectures. With project Freebird, the vehicle design was a driver in selection of the computer architecture. Several possibilities were examined, including: centralized (star), linear and ring architectures. A centralized structure has lines of communication wired directly to each system. A linear architecture has a common line wired throughout the vehicle, and based on the code placed on a signal, the proper system takes in the command; in the case of telemetry, the computer analyzes the information and then forwards the signal to an external station. A ring architecture resembles a linear architecture in that all systems are connected to the computer along a common line. The ring, however, has all systems connected in series.

The centralized architecture is the simplest setup. In case of a failure of one of the systems, the CPU should not suffer deleterious consequences as in the ring architecture. A failure of a system link to the CPU in a ring architecture breaks the link between the CPU and the subsystems, forcing redundancy in pathways and connections. The centralized system requires more wiring harnesses than the ring or linear setups, and extensive rewiring and modification for the addition of nodes to the vehicle. The linear architecture is very flexible in terms of adding or deleting nodes, while the difficulty of changing the ring system resembles that of altering the centralized. The linear system was chosen for Freebird due to its maturity in space applications as well as its flexibility. The design of Freebird is modular in nature, thus the linear architecture allows ease in the addition or removal of the propulsion pack or crew module.

8.3.2 Command Sources

Commands to the system fall into three categories. While not in flight, highest priority is given to commands to perform a test of the subsystem. Giving a test command highest priority ensures that the subsystem can be tested before it processes data or commands if the subsystem is suspect. Generally, the test command is given for pre-flight checks. Second priority is given to commands from an uplink, which includes manual control by astronauts piloting the vehicle. In the case of an emergency, the uplink from ground would take precedence over the manual override. The on-board computer system receives lowest precedence, as it should generate high-level commands only when operating under autonomous control. Low level commands will automatically executed by the CPU.

Command of the vehicle and system comes from several sources. Under remote control, an uplink signal is the source of instruction. Under autonomous operation, the vehicle responds to outputs from its programming. The vehicle's sensors serve as inputs to the program. The astronauts input commands directly through the pilot's interface while flying the craft.

The uplink signal for command is not a system driver for communication links, as it has a low data rate requirement.

8.3.3 Command Decoding and Validation

The command decoder is software implemented in the central computer. Figure 8.2 gives a simplified functional flow diagram of the command decoding and validation process. Command messages are sent to the central computer from different sources as described in the above section. The command arbitrator forwards the highest priority signal in to the command decoder for processing. Command messages are then validated prior to any further execution. A command message in standard format consists of a synchronization code, spacecraft address bits, command message bits, and error check bits. The validation process involves the reception of the synchronization code, check for an exact match of spacecraft address, and check for an exact match of the command message's length and content using informations in the error check bits.

Validated commands are deciphered by a decoding and decompression program to convert the content of the command message into a format used by the rest of the system. The reasons commands are encoded in the first place are to improve security and to reduce the transferred data size with the use of data compression. The decoding software also determines the specific command output interface channel and its respective type of signal. This process also ensures that the intended affect of the command is valid.

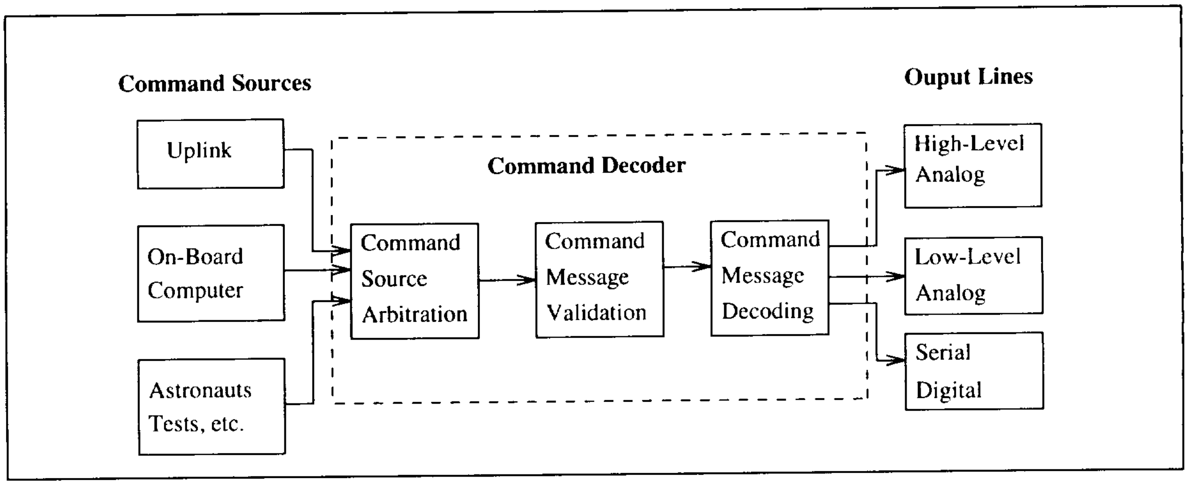

FIGURE 8.2 *Command Decoding and Validation Flow Diagram.*

8.3.4 Command Outputs

The command subsystem outputs three different categories of signals. The subsystem can output a high voltage pulse/controller signal for actuators and servo-mechanisms. A low voltage pulse is also a possible output, a simple digital one or zero. Finally, for communications, the computer can output a stream of data, generally a voice communication signal.

8.3.5 Command Equipment Specifications

The choice of the central processing unit (CPU) was driven mainly by the processing needs of GNC, which requires 11 Million Instructions Per Second (MIPS) throughout for parameter and state identification. Teleoperated control of the manipulator arms, which necessitate real-time video image processing and position calculations, also require large amounts of processor time. The sizing of the CPU was based on the fact that all computational needs on the vehicle are supplied at one central CPU. Estimates for the computational resource required by functions on the vehicle are based on typical numbers [2] scaled up to the size and complexity of Freebird. The total estimate of the required computing power of 15 MIPS was then doubled to allow for future expansion of software and for worse case situation in the estimates. The final CPU size is therefore 30 MIPS. This is well satisfied with Honeywell's Advanced Spaceborne Computer Modules (ASCM).[3] These ASCM features 32-bit RISC multi-processors with 6 MBytes of on-board memory and operation speed of more than 40 MIPS. These modules have a MIL-STD 1553 B serial data bus, which is the data bus standard selected for Freebird. Three of these ASCM modules are available for double redundancy. During normal operation two modules will be in full operation while the third will be in stand-by mode. Each module requires very low nominal and peak power of 1.52 watts and 3.2 watts respectively; and each weights approximately 0.33 kg. They each have a reliability count of 2.5 failures per million hours of operation. All three modules are housed in a single Control Processor Module (CPM) which is 360 cubic inches in volume and nominally draws 25.3 watts of power for its functions and for the ASCM modules inside. The CPM itself weighs 9 kilograms.

8.4 Telemetry

The telemetry subsystem is responsible for monitoring the health and activity of the entire vehicle, which it does by collecting data points from each subsystem. These data points are detailed in the summary tables below, and will take one of three forms: analog, digital, or bi-level. The types of measurements taken will include temperature (cabin, fuel tanks, engines), open/closed (valves and the Common Berthing Adapter), on/off (electrical equipment) and on/off line (various sensors and backup components).

8.4.1 Telemetry Interfaces

The telemetry data monitored on Freebird is shown in Table 8.1. Data is monitored by the central processing units, which then forward the data to the communications subsystem and the crew monitors. Due to the high data rate from the propulsion system during a burn and the Remote Manipulation System (RMS), much of the telemetry data will be processed on-board, with status data downlinked to ground. See the respective sections for the full telemetry data for these systems.

TABLE 8.1 *Summary of Downlinked Telemetry Data*

Component	Telemetry	# of lines	Freq [Hz]	Analog/ Digital	Bits/ Sample	Data Rate [Bits/Sec]
Structures						
InterModular Interfaces (x2)	Capture Latch Pos.	8	1	A	8	128
	Powered Bolt Pos.	24	1	A	8	192
	Attenuators Pos.	4	1	A	8	32
Common Berthing Adaptor (CBA)	Capture Latch Pos.	8	1	A	8	128
	Powered Bolt Pos.	24	1	A	8	192
	Attenuator Pos.	4	1	A	8	32
RMS (x2)	Motor Position	7	40	D	12	3360
	End Effector Position	3	40	D	8	960
	End Effector Force	4	20	A	8	640
	End Effector Moment	4	20	A	8	640
	Camera Status	2	1	D	1	2
Para. Antenna Gimbals (x2)	Motor Position	3	10	D	12	360
Omni Antenna Actuators (x4)	Antenna Position	1	.1	A	12	1
GNC Actuators (x6)	Equipment Position	1	.1	A	8	1

TABLE 8.1 *Summary of Downlinked Telemetry Data*

Component	Telemetry	# of lines	Freq [Hz]	Analog/ Digital	Bits/ Sample	Data Rate [Bits/Sec]
Power						
Fuel Cells (x2)	LOX Tank Density	1	1	A	8	8
	LOX Temperature	1	1	A	8	8
	LH2 Tank Density	1	1	A	8	8
	LOX Temperature	1	1	A	8	8
	Heater Status	1	1	D	1	1
	Coolant Pump Status	1	1	D	1	1
	Coolant Temperature	1	1	A	8	8
	H2 Pump Status	1	1	D	1	1
	Power Load	1	1	A	8	8
Propulsion						
Main Engines (x4)	Temperature	2	20	A	8	1280
	Pressure	2	20	A	8	1280
	Status	1	20	D	1	80
	Valve Status	4	20	D	1	320
Main Engine Tanks	LOX Tank Pressure	2	10	A	8	160
	LOX Tank Temp.	2	10	A	8	160
	LH2 Tank Pressure	2	10	A	8	160
	LH2 Tank Temp.	2	10	A	8	160
RCS Engines (x28)	Temperature	2	5	A	8	2240
	Pressure	2	5	A	8	2240
	Status	1	5	D	1	140
	Valve Status	2	5	D	1	280
RCS Fuel Tanks (x2)	MMH Pressure	2	1	A	8	32
	MMH Temperature	2	1	A	8	32
	NTO Pressure	2	1	A	8	32
	NTO Temperature	2	1	A	8	32
Thermal Control						
Smart Box:						
Thermometers	Temperature	18	1	A	8	144
Louvres (x2)	Louvre Angle	2	1	A	8	32
Radiator Heaters (x2)	Status	1	1	D	1	2

TABLE 8.1 *Summary of Downlinked Telemetry Data*

Component	Telemetry	# of lines	Freq [Hz]	Analog/ Digital	Bits/ Sample	Data Rate [Bits/Sec]
RCS:						
Thermometers	Temperature	64	1	A	8	512
Heaters	Status	64	1	D	1	64
Crew Module:						
Thermometers	Temperature	12	1	A	8	96
Thermostat	Temperature	1	1	A	8	8
Air Recycling Pump (x2)	Mass Flow	2	1	A	8	32
	Status	1	1	D	1	1
Freon Coolant	Temperature	2	1	A	8	16
Active Cooling Pump (x2)	Status	1	1	D	1	2
Louvres (x2)	Louvre Angle	2	1	A	8	32
Radiator Temperature (x2)	Temperature	1	1	A	8	16
Radiator Heaters (x2)	Status	1	1	D	1	2
GNC						
DCS (x4)	Vehicle State	1	4	D	16	64
	Status	1	1	D	1	1
	Rotation Rate	1	4	A	8	32
MIMU (x2)	Vehicle State	1	10	D	16	160
Star Tracker	Target Vector	1	10	D	16	160
Rendezvous Radar Transponder (x2)	Signal Strength	1	1	A	8	8
	Loop Stress	1	1	A	8	8
	In-Lock	1	1	Bi-level	1	1
	Power out	1	1	A	8	8
	Temperature	1	1	A	8	8
LNS (x2)	Target State	1	10	D	16	160
Docking Camera (x2)	Status	1	1	D	1	1
C^3T						
TDRSS Transponder (x3)	Signal Strength	1	1	A	8	8
	Loop Stress	1	1	A	8	8
	In-Lock	1	1	Bi-level	1	1
	Power out	1	1	A	8	8
	Temperature	1	1	A	8	8

TABLE 8.1 *Summary of Downlinked Telemetry Data*

Component	Telemetry	# of lines	Freq [Hz]	Analog/ Digital	Bits/ Sample	Data Rate [Bits/Sec]
EVA UHF Transponder (x2)	Signal Strength	1	1	A	8	8
	Loop Stress	1	1	A	8	8
	In-Lock	1	1	Bi-level	1	1
	Power out	1	1	A	8	8
	Temperature	1	1	A	8	8
CPU (x3)	Agreement	1	2	D	2	4
	Clock	1	2	D	32	64
	Status	1	2	D	1	2
Parabolic Antenna (x2)	Open circuit	1	1	A	8	8
Omni Antenna (x2)	Open circuit	1	1	A	8	8
EVA Omni Antenna (x2)	Open circuit	1	1	A	8	8
ECLSS						
Pressurized Oxygen Tank	Temperature	1	1	A	8	8
	Pressure	1	1	A	8	8
Cabin Air Storage System	Temperature	1	1	A	8	8
	Pressure	1	1	A	8	8
	Compressor Status	1	1	D	1	1
Cabin Fans	Air Flow	1	1	A	8	8
Flow Valves	Status	1	2	D	1	2
Cabin Sensors	Atmospheric Param.	6	1	A	8	48
Pressure Suit	Pressure	1	1	A	8	8
Fire Detection Sensors	Fire	6	1	D	1	6
Cargo Bay Lights	Status	1	1	D	1	1
Extravehicular Mobility Unit	Pressure	1	1	A	8	8
	Life Signs	6	1	A	8	48
Total:						**17190**

8.4.2 Telemetry Processing

Telemetry data from the various subsystems is monitored by software in the CPU. The CPU looks at each telemetry signal and compares it to a given preset range. If that signal is outside the range, the telemetry subsystem informs the command subsystem of the malfunction and the command subsystem reacts appropriately. The telemetry data is then sent on to the communication subsystem for downlink to ground stations.

Each telemetry signal is monitored at a specific frequency related to its importance. Due to this, there is a set order for reading the different signals. This order is set in advance and cycled through repeatedly. The number of signals, sampling frequency, and bits per sample, determine the telemetry data rate.

Additionally, the analog signals must be converted to digital form before they are read by the CPU. There are three types of analog signals: high-level, low-level, and passive. High-level analog signals can be directly converted to digital signals, whereas low-level analog signals must be amplified first. The passive analog signals provide a voltage drop when supplied with current, which can then be converted to digital form.

8.5 Communications

The communications subsystem supports external and internal communication. The external communications links allow information exchange between ground sources as well as the Shuttle and the Space Station, facilitating rendezvous, docking and tending procedures. The internal and EVA communications links allow communication between astronauts.

8.5.1 Communications Scenarios

There are 5 basic communications scenarios that Freebird accommodates: docked on station, rescue around the station, rescue elsewhere in LEO, tending in LEO, and tending in GEO.

While docked at Space Station Alpha, Freebird communicates with station personnel and systems through a link in the common berthing adaptor. This link provides the necessary bandwidth for all telemetry and voice data. The Space Station can then downlink Freebird's telemetry through its own communication system.

For rescues around Space Station Alpha, Freebird will either communicate directly with the station or through TDRSS, a set of two satellites in GEO that provide communications services to most of LEO.

For rescues elsewhere in LEO and standard tending missions, Freebird communicates through TDRSS.

For missions to GEO, which are all under teleoperation, Freebird communicates through the DSN, which consists of 3 ground stations around the planet that provide full coverage of all GEO orbits.

While in LEO, there will be periodic blackouts of TDRSS due to the Earth's shadow. In this case, Freebird will transmit telemetry data to the Indian Ocean Air Force Satellite Control Network Station. If this is not possible, Freebird will store the telemetry data and forward it to TDRSS when in view again.

8.5.2 Antenna Issues

Frequency Selection

The frequency selection for the communications system was driven by the receive frequency bands of the external systems that Freebird will communicate with. S-Band (2.025 - 2.3 GHz) was the only band that all of the major external systems (TDRSS, DSN, and ISSA) have in common. It was decided to go with double redundancy of the TDRSS S-band transponders, rather installing additional transponders for Ku-band, which would have given the system a backup in the case of Freebird's S-band transponders failing.

Data Rates

The maximum downlink data rate of the communications system is roughly 6.6 Mbps, which is the maximum data rate that DSN can receive. DSN has the lowest downlink data rate capability of the external systems we will be communicating with and so was the design limit. The teleoperated missions to GEO, which would be linked through DSN, have the highest data rate requirements of all the missions, due to the necessity of a video link, and so the system was not designed for any higher data rates. The total data rate includes continuous telemetry data, voice communication, and one channel of video. The voice will require 56 kbps, based on a sampling frequency of 8000 Hz, and 7 bits for each sample, including 1 control bit, using Delta Pulse Code

Modulation. The telemetry data from all subsystems comes to 17.2 kpbs (see Table 8.1 on page 156), which leaves 6.527 Mbps for the video link design. (see Table 8.2)

TABLE 8.2 *Communications Data Rates*

Information	Data Rate [Mbps]
Voice	0.056
Telemetry	0.017
Video	6.527
Total	6.600

Standard video communication requires around 44 Mbps, but by utilizing various compression schemes, 6.527 Mbps satisfies the requirements of teleoperation of Freebird. There will only be one channel of video transmitted while under teleoperator control, despite the fact that Freebird will have a total of four external cameras: two used for docking and one on the end of each manipulator arm. This decision was driven by data rate constraints and because the remote operator will only have a need for one of the cameras at a time, depending on whether they are docking, or moving one arm or the other. The uplink data rate is around 85 kbps, including voice data, while in LEO. In GEO, DSN can only uplink at a data rate of 2000 bps. Therefore, only high-level commands can be transmitted to Freebird while in GEO.

Antenna Placement

Freebird has two sets of antennas positioned on the "smartbox" module: two large parabolic antennas on either side of the spacecraft, and a pair of omni-directional antennas for S-band communications. Another pair of omni-directional antennas will be installed on the exterior of the Crew Module, for EVA communications. The parabolic antennas will be retractable for launch, and gimballed in two axes for pointing. The omni-directional antennas will be actuated

Link Sizing

The omni antennas are of standard design, roughly 50 cm in length and 5 cm in diameter, weighing about one kilogram. They have zero gain and broadcast uniformly in space, with the exception of the two imbedded in the body, which essentially broadcast to a hemisphere. The performance was evaluated for the scenario of routing the signal through TDRSS, and it was determined that the maximum data rate for the omni system is around 3300 and 10000 bps for uplink (TDRSS to Freebird) and downlink respectively. The omnis will not be used with DSN, unless the parabolic antennas experience a failure, because DSN is a single-access link.

The parabolic antennas were sized according to the worst case of the across-orbit TDRSS link, which would be when Freebird is on one side of the Earth, communicating to the TDRSS satellite on the far side. The calculations in the link budget (see Table 8.3) assumed a minimum gain at TDRSS of 32.1 dB, which would occur at the edge of the TDRSS field of view. The parabolic antenna will have a diameter of 1.2 meters, and will be fed with a power of 150 W.

TABLE 8.3 *Worst Case Link Budget for Downlink through TDRSS*

Parameter	Value	Units	Comment
Frequency	2.25	GHz	S-band
Transponder/Amplifier Power Output	150	Watts	

TABLE 8.3 *Worst Case Link Budget for Downlink through TDRSS*

Parameter	Value	Units	Comment
Transmit Antenna Diameter	1.2	meters	
Transmit Antenna Beamwidth	7.78	deg	
Transmit Antenna Efficiency	55	%	Standard for parabolic
Transmit Antenna Effective Gain	26.17	dBW	
Transmit Antenna Pointing Error	.5	degrees	
Transmit Pointing Loss	-0.05	dB	
Effective Isotropic Radiated Power	46.88	dBW	(Total power out of antenna)
Range	4.26×10^7	meters	LEO to GEO (across orbit)
Space Path Loss	-192.1	dB	
Atmos. Mean Physical Temp.	279.2	deg K	
Receive Antenna Diameter (TDRSS)	18	meters	
Receive Antenna Minimum Gain	32.1	dB	[4]
System Noise Temperature	1783 deg K	deg K	
Noise Spectral Density	-196.1	dBW/H	
Total Received Power	-113.1	dB	
Data Rate	6.6×10^6	bps	For video downlink
Bit Error Rate Desired	1.00×10^{-5}		
Required E_b/N_o	10		Energy per bit/noise density
Calculated E_b/N_o	14.8		
Margin	4	dB	

Parabolic Antenna Pointing Strategy

The antenna is pointed by a complex hand-shaking scheme with the target communications system, whether it be TDRSS or ISSA. GNC provides the system with a rough estimate of the position of the target, for example, TDRSS, from simple trajectory programs for each target system which are stored in memory. A few minutes before communications are scheduled to start, the target system points at the expected position of Freebird, (provided by ground stations) and beams a low data rate test signal. Freebird picks up this signal and returns it immediately, which signals the target to shift to pseudorandom code transmission, and then up to the full carrier signal, upon reply of each signal from Freebird. Once Freebird acquires the carrier signal it goes into an automatic carrier tracking loop, which measures signal strength received by the antenna and corrects the antenna pointing when the signal weakens.

8.5.3 Equipment Specifications

Freebird has two 1.2 m diameter parabolic center-feed antennas, two S-band omni-directional antennas, and two VHF antennas.

To enable communication through the antennas, the vehicle has doubly redundant TDRSS-2 transponders from Motorola[5]. Within the transponder-antenna path are filters and diplexers for signal cleaning and selection.

The power output of the transponders is only 2.5 or 5 W, depending on the configuration, so there will be a Traveling Wave Tube Amplifier (TWTA) between the transponder and the antenna, in order to boost the power up to the required 150 W. The possibility of utilizing a smaller, more reliable solid state amplifier was investigated, but currently, the TWTA is more efficient at powers above 10 W, and the development of higher power solid state amplifiers was not expected to be completed in time for utilization in this vehicle.

All of the equipment utilized by the communications subsystem has full operational capability

8.6 Failure Modes

The command, communication, control and telemetry subsystem has sufficient redundancy and reliability to meet the requirement of 0.999 reliability. Each of communications, command and telemetry meet the reliability required by NASA. The telemetry subsystem monitors failures within the vehicle and the CPU directs the secondary system to take over operations. All C^3T failure modes are detailed in Table 8.4.

8.6.1 Communications Failure Modes

There are two types of failures for the communications subsystem. The first is the loss of the communications link for the parabolic antennas, while the second type is hardware failure.

If the communications link between Freebird and either TDRSS or DSN fails, Freebird will lose voice and video communications with ground controllers. In the case of DSN, the telemetry link will also be lost due to the fact that telemetry is also sent over the parabolic antennas. After the loss of the communications lock, Freebird will attempt to reestablish contact following the same procedure used in establishing the original link.

The communications subsystem utilizes redundancy in all components, such that it can survive a single failure in any component. However, a double failure in most components would lead to a failure of the mission. Additionally, there are three separate internally redundant transponders, so that a double failure of the complete transponder would not shutdown the communication subsystem. In the case of the failure of a single component, the central processing unit would detect the failure in the telemetry data from that area, and route the communications signal around that component. The loss of the antenna pointing for both of the parabolic antennas will require Freebird to either point the antenna by reorienting the vehicle or utilizing the omni antenna as the sole mode of communication. This would reduce the data rate for uplink and downlink through TDRSS to 3300 and 10000 bps, respectively. The omni is capable of uplink and downlink data rates of and 2 and 250 kbps through DSN.

8.6.2 Command and Control Failure Modes

The Command and Control subsystem assumes a very critical role in a mission of Freebird, and requires a very high level of reliability to assure successful missions. Since this subsystem is implemented in the central computers, its failure modes can be taken to be those of the computer system. Due to the high level of reliability required, the computer system is implemented with double redundancy. Each computer module has a tested reliability of 2.5 failures per million hours of operation. Double redundancy, with the possibility of resetting any module that crashes improves the reliability to less than 0.8 failure per million hours of operation. During normal operation all three computer modules will operate concurrently, with the capability of removing one from the control system if there is problem. The failed computer would then be shutdown, reset, or re-programmed. With a single failure, ground control can either decide to continue the mission or to abort it and return to ISSA. With two computers failed there is an absolute need to return to ISSA, and triple failure would be mission critical and life threatening to any crew on-board.

TABLE 8.4 *Failure Modes*

<table>
<tr><th rowspan="2">Component</th><th rowspan="2">Failure Mode</th><th rowspan="2">Effect</th><th rowspan="2">Response</th><th colspan="2">Criticality</th></tr>
<tr><th>Crewed Config.</th><th>Teleop.</th></tr>
<tr><td rowspan="3">TDRSS Transponder and amplifier</td><td>single failure</td><td>No visible effect</td><td>Utilize backup</td><td rowspan="3">2</td><td rowspan="3">1</td></tr>
<tr><td>double failure</td><td>No visible effect</td><td>Utilize backup</td></tr>
<tr><td>triple failure</td><td>No Communi-cations</td><td>Return to ISSA</td></tr>
<tr><td rowspan="2">EVA Transponder and antenna</td><td>single failure</td><td>No visible effect</td><td>Utilize backup
Astronauts return to craft</td><td rowspan="2">2</td><td rowspan="2">N/A</td></tr>
<tr><td>double failure</td><td>No Communi-cations with EVA Astronauts</td><td>Astronauts return to craft</td></tr>
<tr><td rowspan="2">S-band antennas</td><td>single failure</td><td>No visible effect</td><td>Use other antenna</td><td rowspan="2">2</td><td rowspan="2">2</td></tr>
<tr><td>double failure</td><td>Communications through same antenna</td><td>Return to ISSA</td></tr>
<tr><td rowspan="3">CPU</td><td>single failure</td><td>No visible effect</td><td>Change voting scheme</td><td rowspan="3">1</td><td rowspan="3">1</td></tr>
<tr><td>double failure</td><td>No visible effect</td><td>Eliminate voting, return to ISSA</td></tr>
<tr><td>triple failure</td><td>No control</td><td></td></tr>
<tr><td rowspan="2">Comparator</td><td>single failure</td><td>No visible effect</td><td>Use backup arbitrator</td><td rowspan="2">2</td><td rowspan="2">2</td></tr>
<tr><td>double failure</td><td>No visible effect</td><td>Return to ISSA using only one CPU</td></tr>
<tr><td>Data Recording Unit</td><td>single</td><td>Can't record data</td><td></td><td>3</td><td>3</td></tr>
</table>

8.7 Summary Budgets

Table 8.5 gives a summarized version of the mass, power and volume budget for the whole subsystem. The command, communications, control and telemetry subsystem will cost approximately 6 to 8 million dollars for hardware, which is well within the allotted budget. The numbers for mass, power, volume, and cost for the various components either come directly from the manufacturer, or were arrived at by analogy.[6] Table 8.6 gives a summary of telemetry and command signal generated by components this subsystem.

TABLE 8.5 *Summary Budgets*

Component	# of Units	Total Mass [kg]	Peak Power [W]	Stand-By Power [W]	Volume [m^3]
TDRSS Transponder (Motorola)	3	19	135	9	.0114
EVA Transponder	2	6	20	2	0.004
Power Amplifier (Hughes)	3	15	106	10	0.0245
Parabolic Antenna (TRW)	2	5	0	0	-
S-Band Omni Antenna	2	2	0	0	0.0019
EVA Omni Antenna	2	2	0	0	0.0019
Internal Communication Network	2	2	10	0	0.0007
Harnessing	N/A	10	0	0	-
Processors (Honeywell)	3	10	36	25	0.0017
Data Recording Unit	1	2	2	2	0.0032

TABLE 8.6 *Telemetry and Command*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital	Command
TDRSS Transponder (x3)	Signal Strength	1	1	A	Coherent Mode
	Loop Stress	1	1	A	Enable Stand-by
	In-Lock	1	1	Bi-Level	PRN ON/OFF
	Power out	1	1	A	Subcarrier ON/OFF
	Temperature	1	1	A	
EVA UHF Transponder (x2)	Signal Strength	1	1	A	Coherent Mode
	Loop Stress	1	1	A	Enable Stand-by
	In-Lock	1	1	Bi-Level	PRN ON/OFF
	Power out	1	1	A	Subcarrier ON/OFF
	Temperature	1	1	A	

TABLE 8.6 *Telemetry and Command*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital	Command
CPU (x3)	Agreement	1	2	D -2 bit	
	Clock	1	2	D- 32 bit	
	Status	1	2	D - 2 bit	
Parabolic Antenna (x2)	Open circuit	1	1	A	
	Extended	1	1	A	Extend
Omni Antenna (x2)	Open circuit	1	1	A	
	Extended	1	1	A	Extend
EVA Omni Antenna (x2)	Open circuit	1	1	A	
	Extended	1	1	A	Extend
Data Recording Unit	Status	1	100	D	Read/Write

References

1. NSTS 07700, Volume XIV, Appendix 7;System Description and Design Data-Extravehicular Activities.
2. W. J. Larson and J. R. Wertz; Space Mission Analysis and Design.
3. Honeywell Inc., Space Systems Group, Las Vegas, May 6-7, 1993; 32-bit Space Processor Showcase.
4. TDN No. 101.2, Revision 6; Space Network Users' Guide.
5. Motorola Inc., Government Electronics Group; 5223 10-83, TDRSS User Transponders.
6. MIT Space System Engineering, Spring 1993; Project Perseus, Chapter 15, Command, Communication, Control, and Telemetry.

Chapter 9

Environmental Control and Life Support System

The main goal of the Environmental Control and Life Support System (ECLSS) is to provide a habitable and comfortable environment in which the crew of Freebird can function. This environment is contained within the crew module, a cylindrically shaped pressure vessel that is attached to the front of the base unit in the craft's crewed configuration (Figure 14.4 on page 247). The ECLSS has been designed to require the minimum mass, volume, power, and cost while still achieving acceptable safety and reliability values. The ECLSS has been assigned an overall reliability of 0.9994. Levels of redundancy within life critical and mission critical subsystems have helped to achieve this value.

9.1 ECLSS Requirements

Many of the ECLSS design requirements are driven by the Freebird top level system requirements and by human physiological and psychological tolerances. The primary system level requirements that drive the ECLSS design are to sustain both a crew of three for a four day satellite repair mission and a crew of eight for twenty-four hours in an emergency crew rescue mission. The additional ECLSS design requirements are listed below.

The crew module must:

- provide a cabin atmosphere which is maintainable both within the human comfort range and within the operating range of cabin equipment.
- provide a waste management system to dispose gaseous, liquid, and solid wastes.
- provide expendables such as oxygen, food, and water.
- safely accommodate crew members during all mission phases.
- provide extravehicular activity (EVA) capability and apparatus.
- provide emergency systems.

9.2 ECLSS Interfaces

From the responsibilities and requirements previously discussed, six subsystems within the ECLSS have been created. These groups include atmosphere control, waste management, expendables, cabin layout, emergency systems, and EVA. The functional block diagram, shown in Figure 9.1, presents an overview of the crew cabin design, showing the subsystems, the main subsystem components, and external and internal interfaces. The main external interfaces occur with the structure, power, and thermal systems. Many life critical components are partially or fully redundant; this redundancy is necessary to meet the high reliability required of a manned spacecraft.

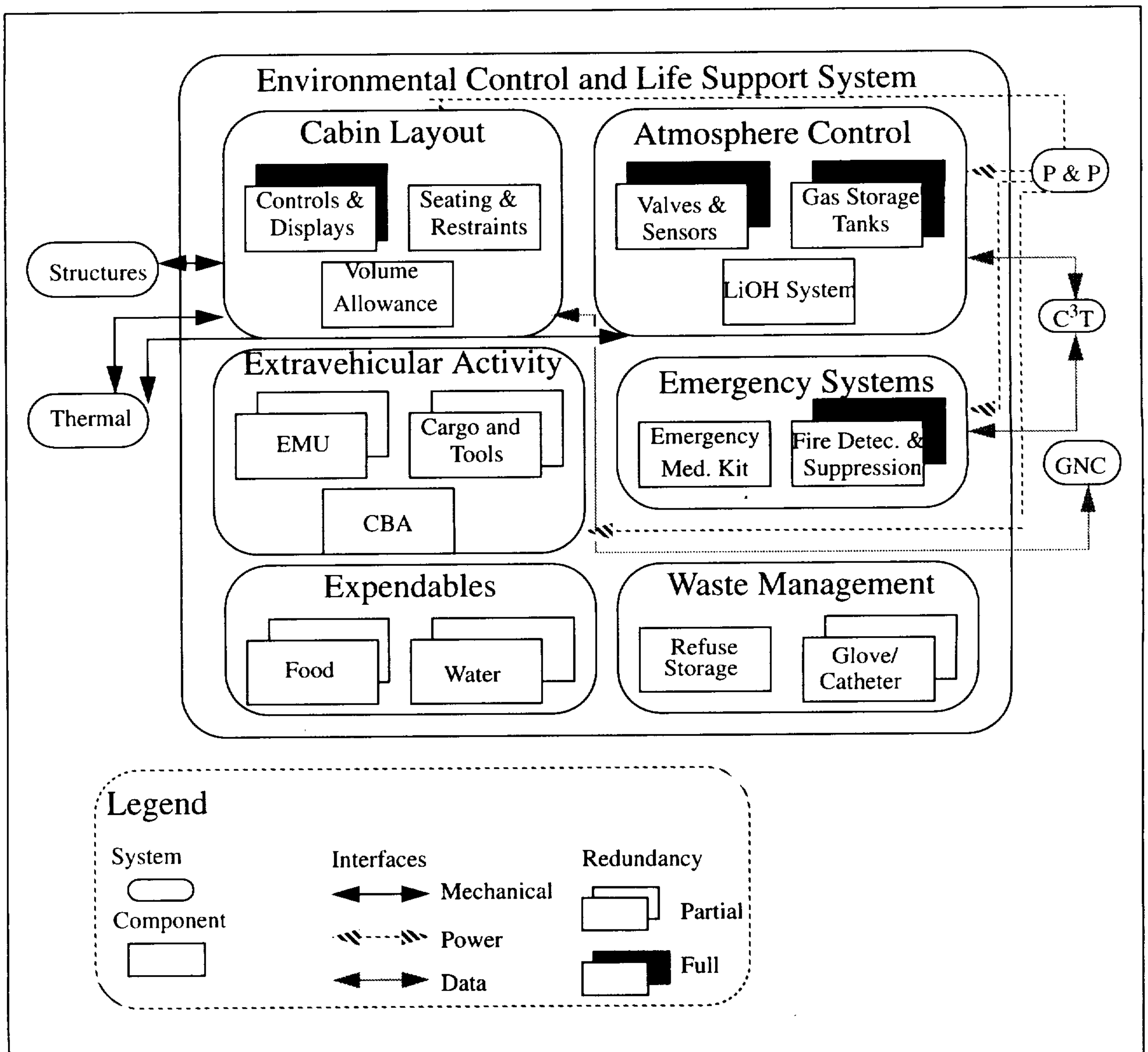

FIGURE 9.1 *Environmental Control and Life Support System Functional Block Diagram.*

9.3 Human Physiological Tolerances

In order to design the life support systems for Freebird, first it is necessary to understand human tolerances to the adverse conditions encountered in Earth orbit. Because the safety of the crew is crucial to any successful mission, the tolerance limits must be avoided when preparing an artificial cabin environment.

9.3.1 Pressure

Sea level pressure of 101.3kPa can be considered the optimum pressure environment for the human body. Freebird operates in a near-vacuum environment, therefore the effects of low pressures on the human body is the primary concern. A decrease in total atmospheric pressure may result in decompression sickness, commonly known as divers' "bends." The bends may occur when a decrease in pressure surrounding a fluid-filled cavity (such as the human body) forces gas trapped in the liquid to evolve out of solution. Initial human symptoms include pain around body joints, while severe cases of the bends can affect the lungs, heart, and central nervous system, causing breathing difficulty, shock, and even death. The occurrence of the bends limits the effectiveness of EVA performed by crew members, and will be discussed further in Section 9.8.2 on page 184.

When the pressure surrounding a volume of liquid is decreased, boiling of that liquid occurs at lower temperatures. At an ambient pressure of 6.3kPa, water vaporizes at human body temperature (37°C). If the cabin pressure falls below 6.3kPa, crew members' body tissues begin to vaporize, resulting in a life-threatening condition known as ebullism [1].

One final concern with pressure loss involves a condition known as barotrauma. Barotrauma is the destruction of body tissue due to any rapid change in pressure. Rapid changes in pressure can affect gas-filled body cavities, such as the lungs, gut, middle-ear, and sinuses. A severe and life-threatening case, such as the rupture of lung tissue can occur with sudden pressure changes of 10.7kPa and higher [1]. The danger of barotrauma requires that adjustments in cabin pressure be performed gradually [2].

9.3.2 Temperature

The ideal temperature for a lightly clothed person is between 11°C and 27°C [1]. Because temperature extremes in Earth orbit may range from -113°C to 1100°C, Freebird must be capable of both storing and radiating heat, depending on its orientation to the Sun. The human body cannot withstand a change in core temperature of greater than ±5°C for an extended period of time. Air temperatures above 37°C (human body temperature) are not desirable, because the body will cease to radiate heat and will begin to absorb heat, possibly forcing a hazardous elevation in core temperature.

While various layers of clothing can trap body heat and combat the effects of low air temperatures, the effects of the cold are equally dangerous. Local frostbite, and ultimately gangrene, can occur in tissue that has been exposed to intense cold. If core temperature falls below 35°C, hypothermia may develop.

While changes in core temperature are certainly undesirable, it is also necessary to understand the narrow human comfort range. Although air temperature conditions may not be immediately life-threatening, even mild thermal discomfort has a strong effect on human performance. It has been observed that at least 5% of any large group will be uncomfortable at any one air temperature. Thus, Freebird's thermal control system must provide a means of manual temperature control, via clothing choice or local air circulation, in order to maintain optimum crew comfort and performance [1].

9.3.3 Humidity

Percent relative humidity indicates the percentage content of water vapor in an atmosphere. At high levels of humidity, condensation occurs and bacteria and fungi growth increases in moist skin folds. The body's thermoregulation system can also be affected since high humidity compromises the body's ability to lose heat through the normal evaporation of perspiration. Low levels of humidity lead to drying of exposed mucous membranes of the nose and throat, inactivation of the protective cilia of the bronchial tubes (resulting in greater chance of infection), chapping of the lips, and drying of the eyes. The optimal water vapor pressure for human comfort is considered to be 1.33kPa. In past U.S. spacecraft, the vapor pressure has been adjustable from 0.8 to 1.9kPa (30% to 70% relative humidity) [1].

9.3.4 Gas Concentrations

Oxygen

The composition of Earth's atmosphere at sea level is 20.9% oxygen, 78% nitrogen, 0.04% carbon dioxide, and trace gases. More important than the concentrations are the partial pressures: 21.3kPa oxygen, 79.7kPa nitrogen, and 40.5Pa carbon dioxide.

Oxygen is certainly the most crucial component in any atmosphere, and hypoxia, the lack of O_2, may be the most serious hazard facing any astronaut. The effects of hypoxia are seen very rapidly, and a lower oxygen partial pressure results in even faster onslaught. At atmospheric oxygen pressures between 21.3kPa and 4.6kPa, mild hypoxia may occur, causing increasing visual, mental, and motor skills impairment as the pressure is reduced; however, these low partial pressures are survivable. Extended exposure to partial pressures below 4.6kPa ultimately result in unconsciousness and death within as few as 4 minutes.

Oxygen partial pressures above the normal sea level value can cause oxygen poisoning, or hyperoxia. However, Apollo astronauts exposed to increased oxygen pressures for up to two weeks demonstrated changes in blood forming tissue, but no adverse effects. Thus it is generally believed that hyperoxia is not a major concern on such short missions [2].

Finally, a high concentration of oxygen in a closed environment greatly increases the risk of fire, which certainly poses a threat to the crew.

Nitrogen

Although nitrogen is the largest component of the Earth's atmosphere, proper function of the human body is not directly dependent on nitrogen consumption. Therefore high concentrations of nitrogen do not appear to affect human functions. As evidenced by the early pure oxygen environments of Mercury, Gemini, and initially Apollo, low concentrations of nitrogen also do not cause adverse effects. In fact, a low nitrogen concentration can actually prevent decompression sickness, since less nitrogen is absorbed into body tissues.

Carbon Dioxide

Unusual levels of carbon dioxide can also affect health. When respiratory frequency is abnormally increased (hyperventilation), the lungs expire larger amounts of carbon dioxide. Since carbon dioxide in the body combines with water to form carbonic acid, reducing CO_2 levels forces the body chemistry to become alkaline. Symptoms of this hyperventilation effect include dizziness, anxiety, and muscle spasms.

Carbon dioxide intoxication can occur when levels of CO_2 in the blood rise slightly above the normal pressure of 5.3kPa. Heightened levels of CO_2 in the atmosphere also affect the acid-base chemistry of the body, causing symptoms of confusion, breathing difficulty, and rapid heart rate. Although the CO_2 pressure at Earth sea level is 40.5Pa, previous space missions such as the Shuttle have set a safe upper limit of 101Pa [1].

9.3.5 Radiation

Since ionizing radiation is capable of dissociating a substance in solution into its constituent ions, overexposure can be extremely harmful to living tissue. Short-term effects to increasing dosages of ionizing radiation range from minor blood changes (10 to 50rem) to vomiting, nausea, and bleeding (200 to 350rem) to certain death (above 600rem). Human tissue response to radiation varies markedly depending upon the rate of active tissue growth. Fast growth tissue, such as the gonads and bone marrow, are extremely sensitive to radiation, while slow growth tissue, such as muscle and cartilage, are quite insensitive. Table 9.1 shows NASA radiation exposure limits for astronauts. These values can be used to determine exposure levels and radiation protection for the crew of Freebird, using the conversion 1 rem is roughly equal to 1 rad in low earth orbit [2]. Methods of radiation protection are described in Section 3.4.2 on page 70.

TABLE 9.1 *NASA Radiation Exposure Limits for Crew Members [2].*

Constraint	Bone Marrow (rem at 5cm)	Skin (rem at 0.1mm)	Ocular Lens (rem at 3mm)	Testes (rem at 3cm)
1-year average daily dose	0.2	0.6	0.3	0.1
30-day maximum	25	75	37	13
Quarterly maximum	34	105	52	18
Yearly maximum	75	225	112	38
Career limit	400	1200	600	200

9.3.6 Acceleration

The zero-g environment has been thoroughly studied, and many long and short term effects on the human body have been observed. Since Freebird crewmembers originate from the zero-g Space Station environment, it is assumed that Freebird's crew will already be accustomed to the effects of weightlessness.

However, accelerations exceeding one-g can pose a hazard to a crew member depending upon the intensity, duration, and axis of placement. The body is most sensitive to g-loading in the +/- Z direction (feet to head) because abnormal blood pressures result in the brain. The best direction for withstanding g-loading is the X (chest to back) direction. Figure 9.2 explains how human acceleration tolerances vary with the direction of acceleration.

9.4 Cabin Layout

The interior cabin layout of Freebird's crew module is designed to allow the astronauts adequate space and comfort while minimizing the volume, mass, and cost required to achieve this goal. The main trade-off addressed involved evaluating crew comfort versus cost constraints. Factors that could affect crew psychological state and productivity, such as adequate personal space, are also addressed in the design. A value of $2m^3$ per person was chosen for minimum volume allowance [4]. Using the rescue configuration as a worst case scenario, in which the cabin must accommodate eight people, a minimum volume requirement of $20m^3$ was determined.

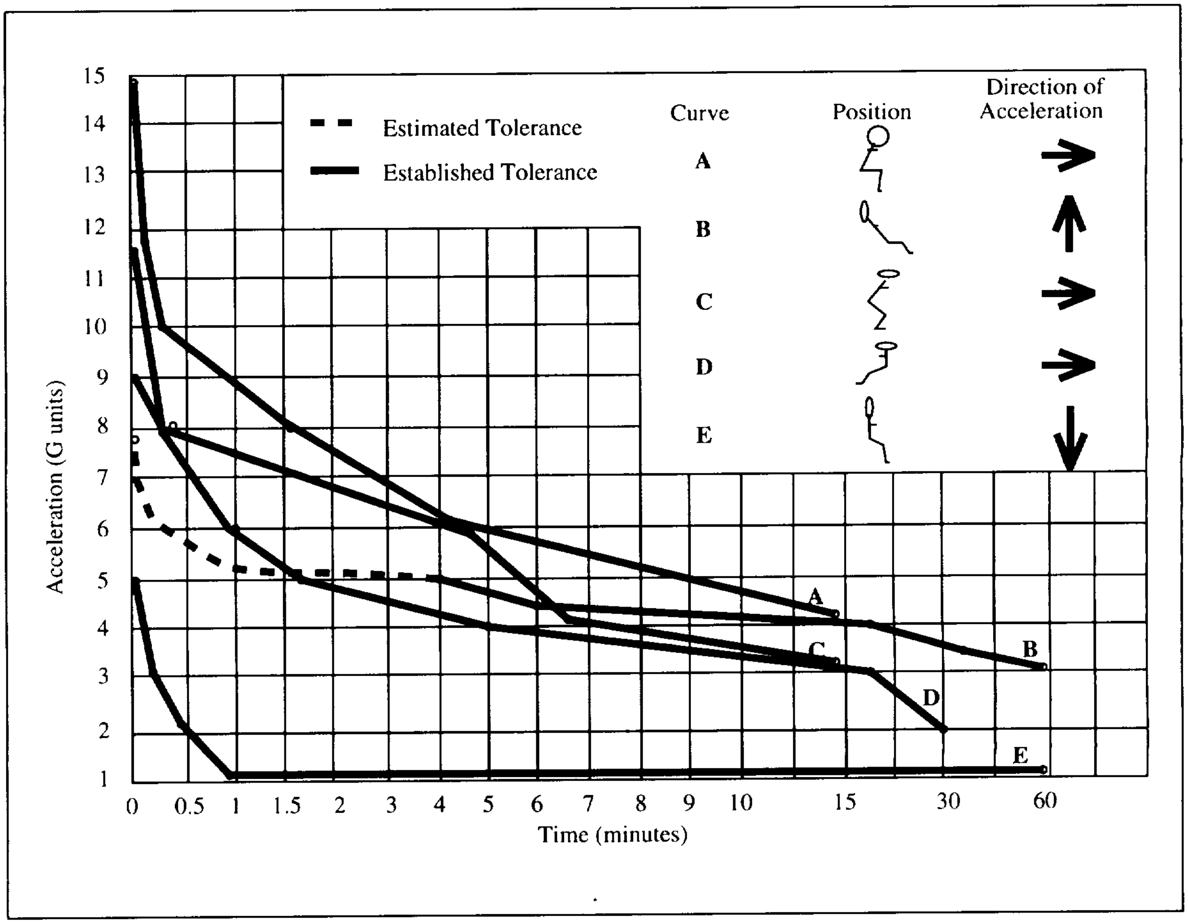

FIGURE 9.2 *Variation of Human Tolerance with Direction of Acceleration [3].*

9.4.1 Overall Cabin Layout

A diagram of the crew module's overall layout is shown in Figure 9.3. The crew module is comprised of a cylindrically shaped pressure vessel with a radius of 2.96m, a length of 4.7m, and a sixty degree cap at the front end. These dimensions were defined by the launch bay of the primary crew module launch vehicle, the Proton C. This module is configured to accommodate a three man crew during normal operations. Two pilot's chairs are positioned at the front end of the craft, near controls and displays necessary for communication and monitoring and control of Freebird. A third seat is positioned in the center of the craft for the remaining astronaut. Windows, displays, and controls, located above the seats, assist the operator of the Remote Manipulator System (RMS). This orientation provides the best view for the astronaut controlling the RMS because most EVA and repair operations occur directly above the windows. Towards the aft of the vehicle is the common berthing adaptor (CBA) under which the Extravehicular Mobility Units (EMU) for EVA are stored. A curtain can be placed across the aft section to provide space for privacy. An airlock was not chosen in this design for reasons that are further described in Section 9.8.1 on page 184. The components of the atmospheric control system are located beneath the cabin floor, and are accessible for maintenance purposes. Storage space for food, water, clothing, and waste is accessible on the walls around the cabin. Two cargo and EVA tool storage areas have been placed in the aft section of the craft and are accessible only from outside Freebird. Fire detec-

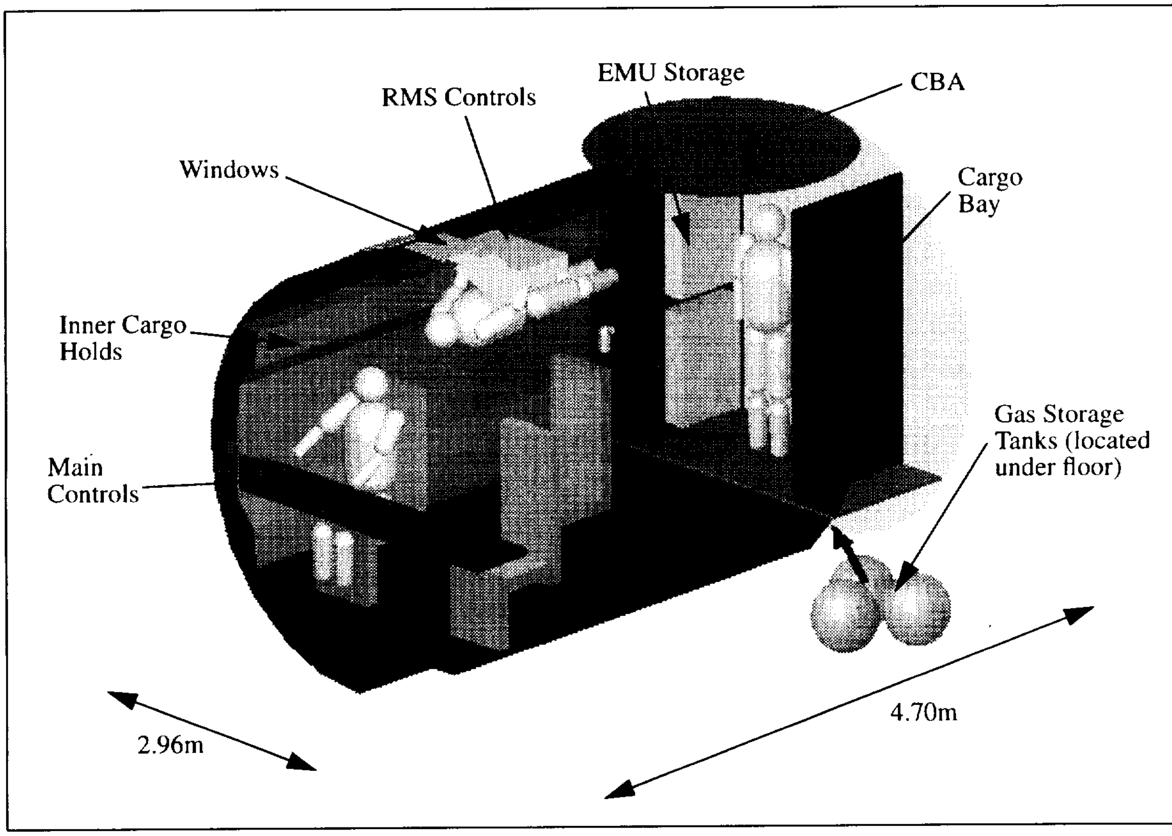

FIGURE 9.3 *Diagram of the Crew Module Cabin Layout.*

tion sensors and fire extinguishers are positioned throughout the cabin, and an emergency medical and tool kit is located within the wall cabinets.

9.4.2 Human Interface Environment

In order to properly operate Freebird in the crewed configuration, an interface is required between the crew and the spacecraft systems. The crew must be able to input information to the craft and receive information in return. In an enclosed environment and possibly an emergency situation this exchange of commands and information must be as accurate and efficient as possible. Controls and displays throughout the cabin allow for information exchange. In addition, windows near the RMS controls can assist in the control of the RMS during EVA and repair operations.

Controls

Astronauts aboard Freebird require controls in order to input information, to communicate, to control the RMS, to adjust the atmosphere control system manually, and to pressurize/depressurize the cabin. The main input devices are onboard keyboards and joysticks. The RMS controls are placed above the seats. Controls for positioning the spacecraft are located both near the RMS controls and on the front consoles near the pilots' chairs. Controls to pressurize/depressurize the cabin are located near the CBA at the back of the cabin. Redundant or manual controls are located in the front portion of the cabin near the pilots' chairs.

Displays

Displays show the status of essential subsystems aboard Freebird. Computer monitors are placed on the front console and in addition computer and television monitors are placed near the RMS controls.

Windows

Windows are placed on the ceiling near the RMS controls. In addition to providing the crew with a spectacular view, windows allow the pilot and the RMS operator a direct view of EVA activities, the RMS, and docking which may be critical if manual docking becomes necessary in an emergency.

9.4.3 Seating and Restraints

The cabin seats are required to withstand a three-g maximum load along the X axis, to be lightweight, compact, and cost efficient. The two seats chosen for the pilot positions are the standard shuttle commander/pilot seats, which are compact, light weight, and reliable. For the third seat in the standard configuration and the additional rescue seats required the standard shuttle specialist seats were chosen, which are light weight, compact, reliable, and stowable (see Figure 9.4).

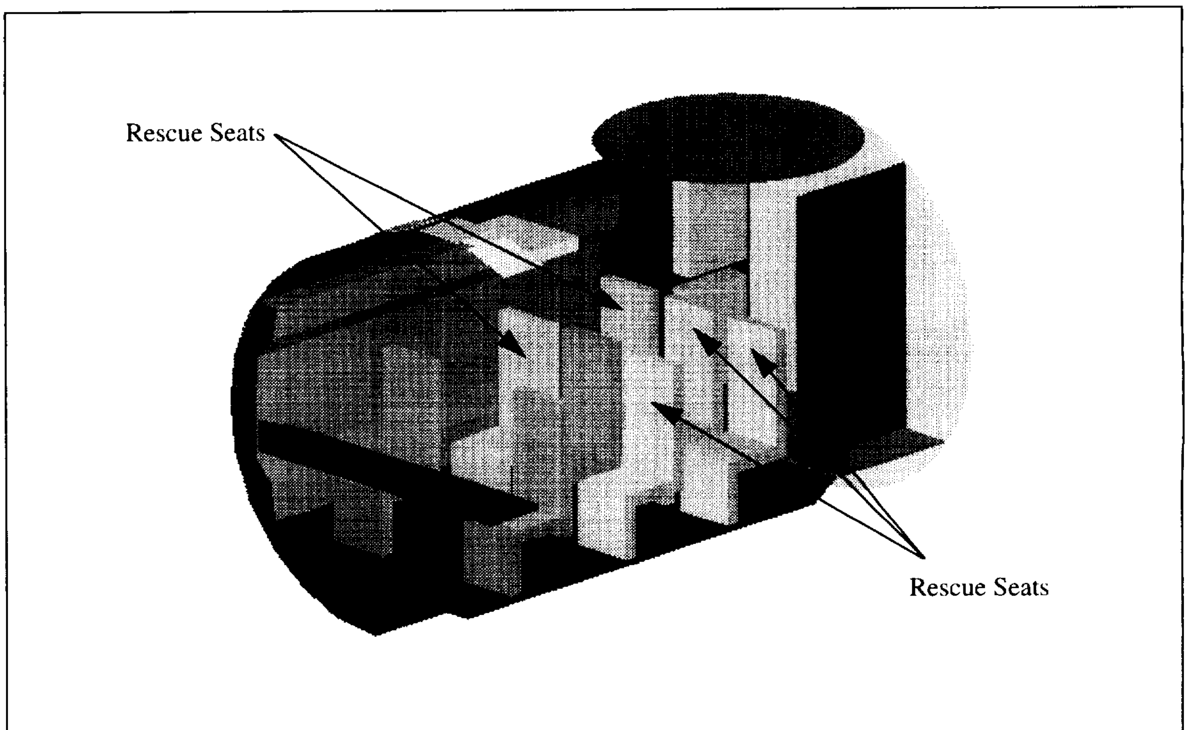

FIGURE 9.4 *Rescue Seat Configuration.*

The RMS operator must be securely positioned to reach both the RMS controls and the Freebird attitudinal controls. A standard foot restraint, such as the variable height work platform that is used in the shuttle aft crew station will be used to locate the RMS operator on the ceiling [5].

9.5 Cabin Atmosphere Control

Within the ECLSS, the atmosphere control system must be considered as the most mission critical subsystem. The exchange of gases between humans and their surroundings is the most immediately critical biological function; as mentioned earlier, death may occur in minutes during hypoxic emergencies. Thus the responsibility of the atmospheric control system lies in providing a comfortable and very reliable climate for the crew of Freebird.

A description of the trade-offs and choices made concerning various aspects of the cabin environment is given below.

Pressure

In the past, the pressure of spacecraft cabins has been maintained at either Earth sea level (101.3kPa) or lower pressures.Table 9.2 shows the main trade-offs involved between these two options.After examining the trade-offs involved, it has been decided that a combination of pressure levels, such as is used aboard the Shuttle, would best meet the mission needs. Because Freebird must dock with the International Space Station Alpha (ISSA) prior to embarking on a crewed mission, it is pressurized to 101.3kPa during stand-by and docking procedures. Cabin pressure is then lowered to 70.3kPa in order to reduce the required EVA pre-breathe time[1][5]. The 70.3kPa pressure level is equivalent to the pressure at 8,500ft altitude on Earth (the altitude of a typical Rocky Mountain ski resort), which is certainly a habitable pressure, even with normal N_2 and O_2 composition [1]. Because of weight constraints, Freebird lacks an airlock; instead, all crew members must don either pressure suits or Extravehicular Mobility Units (EMU) before the cabin is completely depressurized, during which time two astronauts egress to carry out EVA. Typically the rate of pressure change during such a depressurization is limited by specification to 0.7kPa/second [2]. However, Freebird will depressurize in approximately 10 minutes, at a slower rate of 0.012kPa/second due to safety and power concerns. Only after the final EVA is complete is the cabin pressure again increased to 101.3kPa.

TABLE 9.2 *Cabin Pressure Options*

Options	Advantages	Disadvantages
Sea Level Pressure	easy to acclimate to natural cabin atmosphere compatible to ISSA pressure	requires 3.5 hour EVA prep time
Lower Pressure	less structure required for pressure vessel less EVA prep time required	must increase oxygen concentration to prevent hypoxia (results in a fire hazard) incompatible with ISSA pressure

Temperature

The trade-offs involving thermal control of the cabin are discussed in Section 6.4.2 on page 112. The cabin temperature is maintained by a heat exchanger between 18°C and 22°C. This exchange of heat is monitored by an automated system with manual control override for individual crew comfort.

Humidity

The problems with humidity in spacecraft cabins almost always concerns excess levels. Humidity aboard Freebird is controlled within the active thermal control system, which is discussed in Section 6.4.2 on page 112.

C-3.

Gas Composition

In the past, the gas composition of a spacecraft cabin atmosphere has been either pure oxygen or a combination of oxygen and nitrogen. The various trade-offs concerning these two options are listed in Table 9.3. The pure oxygen environment has been eliminated primarily because of the severe risk of fire inherent in such an atmosphere. Since the tragic Apollo 1 fire in 1967, NASA spacecraft have utilized O_2/N_2 atmospheres as a standard design. For these reasons, Freebird also utilizes a mixed atmosphere.

Because astronauts must board Freebird from the ISSA, the atmosphere composition is maintained at ISSA levels of 80% N_2 and 20% O_2 during stand-by and docking procedures. During partial depressurization to 70.3kPa, the composition usually remains the same; however, the oxygen level is adjustable in case symptoms of hypoxia set in at such low pressure.

Supply of Atmospheric Gases

Since nitrogen is not used by the human body in any manner, the oxygen supply system is the main concern when providing necessary gases to the atmosphere control system. A variety of supply methods have been used aboard spacecraft in the past. Previous Russian spacecraft, such as Vostok, Voskhod, and Soyuz, have utilized a solid chemical system to supply oxygen to the life support system. In such a system, oxygen is released when a controlled amount of water is reacted with a solid alkali superoxide. As an additional benefit, the reactant product which results has the ability to absorb carbon dioxide. Although this cycle seems ideal, American experience with this supply method is limited, and information on the Soviet system is difficult to obtain. NASA has used either high pressure or cryogenic oxygen supply systems in past American space vehicles. Very short missions such as the Mercury series used oxygen stored in bottles, while the two-man Gemini capsule carried both pressurized bottles and cryogenic tanks of oxygen. All of NASA's larger and longer missions have demanded the use of a cryogenic supply system.

TABLE 9.3 *Cabin Gas Composition Options*

Options	Advantages	Disadvantages
Pure Oxygen	no EVA prep time required simple, one-component gas system	EXTREME FIRE HAZARD not compatible with ISSA undesirable hyperoxia symptoms
O_2 / N_2 Mixture	compatible with ISSA more natural breathing environment	requires some EVA prep time complex, two-component gas system

The oxygen consumed by a three-person crew for four days amounts to 19.2kg, including a factor of safety of two because of the extreme importance of adequate oxygen supply to crew survival. A comparison has been performed between a steel (pressure vessel steel HY-130) high pressure tank and a cryogenic tank, both designed to store 19.2kg of oxygen. Table 9.4 shows the calculated mass and volume of each tank, including the trade-offs that have been considered. Because of the reduced power, mass, and complexity, a high pressure oxygen tank system is used to supply oxygen to Freebird's atmosphere control system. This tank is pressurized to 22,700kPa (similar to the Shuttle nitrogen gas pressure vessels [5]) and is kept at a temperature of 298°K, so that no additional heating is required before the gas is introduced into the cabin. Because of the highly critical nature of this supply system, a redundant oxygen tank filled with 19.2kg of O_2 and with completely independent piping is used.

Because the crew module of Freebird lacks an airlock, it is necessary to perform a depressurization/repressurization cycle for each egress and ingress involved with EVA. This amounts to a total of six cycles and results in an additional 125kg of O_2 and N_2 required to repressurize the cabin six times. However, if during depressurization the cabin atmosphere is stored rather than vented overboard, this 125kg of gas can be saved. Thus Freebird carries an additional steel pressure vessel, to store the compressed cabin atmosphere during depressurization. Such a vessel requires 0.06m^3 in volume and weighs approximately 18kg. In addition a compressor with a mass of approximately 50kg is required to pump the cabin atmosphere into the storage tank. Assuming 70% efficiency, such a compressor requires a peak power of roughly 5kW if the rate of depressurization is maintained at 0.012kPa/second as mentioned in "Pressure" on page 177.

TABLE 9.4 *Comparison of High Pressure and Cryogenic Oxygen Tanks [6][7]*

System	Dry Mass [kg]	Volume [m^3]	Remarks
High Pressure	17.2	0.07	volume varies greatly with tank pressure mass varies greatly with tank temperature gas may require heating if tank temperature is low
Cryogenic	23	0.012	additional power required for cooling/heating

Atmospheric Control System Components

The atmosphere control system, shown in Figure 9.5 actually encompasses a variety of subsystems. Cabin air enters the flow loop through vents in the cabin walls. This flow is circulated through the system by a series of redundant fans. Within the system lies the scrubbing unit, responsible for removing both carbon dioxide and trace contaminants from the cabin atmosphere. The atmosphere scrubbing system is further described in Section 9.7.1 on page 181. The cabin thermal control system is also connected in the flow loop; excess heat and humidity are removed by a head exchanger. The fully-redundant oxygen supply system, consisting of two pressurized storage tanks, supplies fresh oxygen to the flow loop. During depressurization, a pressure suit is also connected to the cabin atmospheric control system, in order to protect the crew member remaining in the cabin during EVA. The pressure suit, modeled after the original Mercury suit, is pressurized to 55.16kPa, which alleviates the need to prebreathe prior to entering the suit [8]. Air evacuated from the cabin during each depressurization is compressed into a storage tank until it is reused during repressurization. Solenoid valves regulate and direct flow throughout the system. These valves are intended to be adjusted both automatically and manually via cabin controls.

9.6 Expendables

Because the various Freebird missions are expected to last up to four days, a supply of food and water must be carried aboard to sustain the crew. Replenishment is crucial to maintaining the crew at optimum performance levels. The quantity of expendables required aboard Freebird is driven by the four day satellite repair mission rather than the 24 hour crew rescue mission.

9.6.1 Water

Freebird is powered by fuel cells, which derive energy from the exothermic reaction of hydrogen and oxygen (refer to Section 4.3 on page 82). Water is a product of this reaction, and can be used for human consumption. However, this process requires a filtering system and a distribution unit. Because the fuel cells are located in

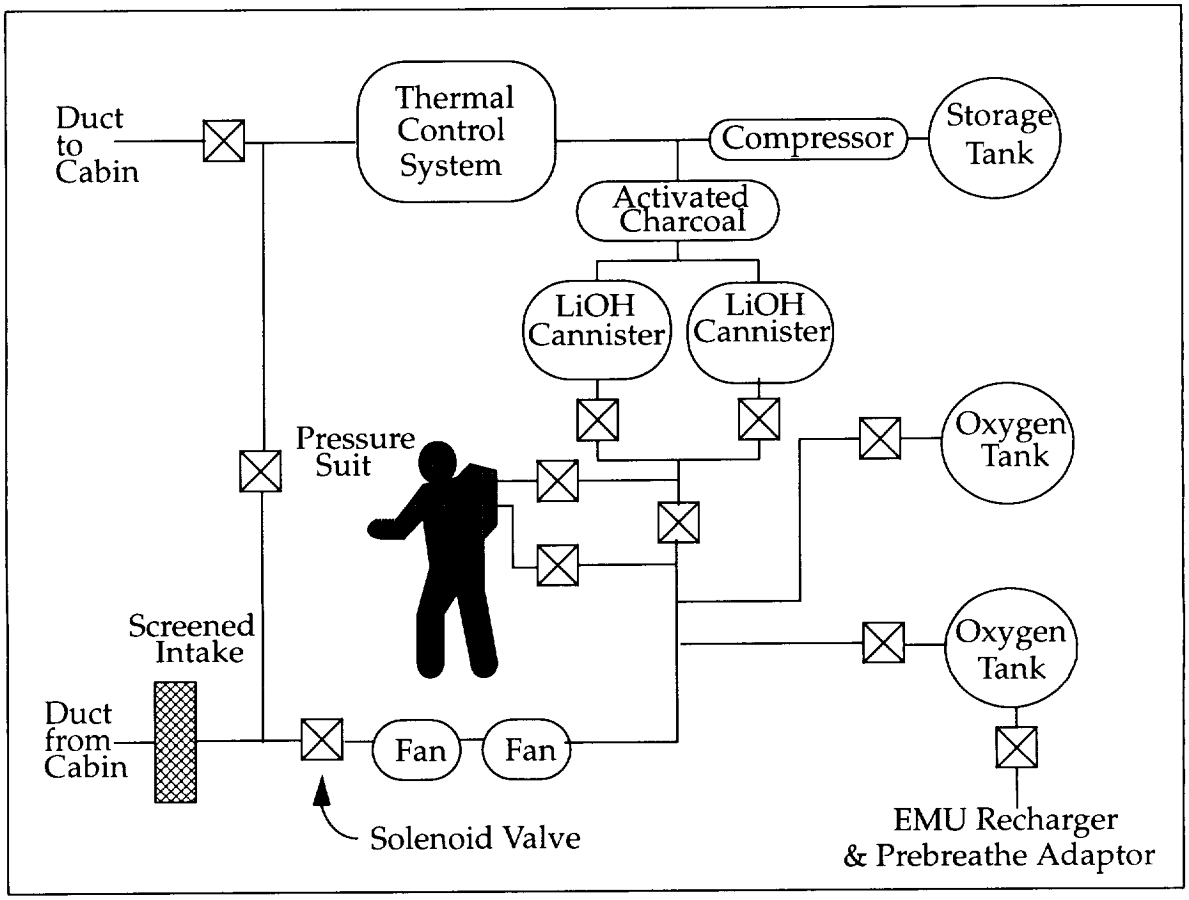

FIGURE 9.5 *Schematic Diagram of the Atmospheric Control System.*

the "smartbox" of Freebird, forming a connection with the crew module would require a bulky piping system. Given the short duration of the Freebird mission, stored water is used as the supply.

Hygiene

It has been determined that crew members may last up to four days without concerning themselves with personal hygiene. In addition, Apollo and Shuttle astronauts have completed two week missions without the use of a full-body shower; therefore, due to lack of necessity a shower is not included in the Freebird cabin. However, in case of tissue contamination from feces, urine, or other distasteful substance, Freebird is equipped with wet, bactericidal, and general utility wipes. These wipes are quite small and may be stored in pockets within the clothing of the crew. The self-contained hand washing system aboard the shuttle was not chosen because of the increased mass, volume, and complexity inherent in its design.

Consumption

Although a person at rest requires approximately 2.5L of water per day, this figure is doubled because the astronauts aboard Freebird will be performing strenuous EVA. Water for consumption is provided in refillable flexible pouches with a capacity of 2.7L, similar to those used during Project Mercury. In this manner, the pouches may be refilled by the Space Station water supply system during Freebird's stand-by mode. Ionic silver is added to this water as a preservative and sterilizing agent, since the taste of water deteriorates within 18 to 20 hours when stored at room temperatures. In addition, juice pouches may also be carried on board to provide a variety of beverages for the crew [1]. A total of 60kg of water is required aboard Freebird, which includes a safety margin of 1.2. Although water and food are considered non-mission critical, a small factor of safety is added to ensure the comfort and morale of the crew. The water bottles require 0.06m^3 of space within the cabin.

9.6.2 Food

Although humans can survive without food for many days, nourishment is provided aboard Freebird in order to maintain the crew's optimum performance levels. The basic energy requirement for a 70kg person at rest is 2,000kcal/day. This value can increase to 4,500kcal/day if strenuous activity such as EVA is undertaken. This upper caloric requirement roughly translates into 1kg (dry mass) of food per day, a value which depends strongly on the nutrients within the foodstuff [1] [9].

Because of the short mission, a food preparation system (rehydration, refrigeration, and heating) such as that aboard the Shuttle is considered too costly from a mass, volume, and complexity point of view. Therefore, food is provided in thermostabilized, dehydrated, and natural form, and is similar the packaged foods provided aboard the Shuttle [1]. Chemical heat packs may be provided as a simple method of heating certain food items to desired levels. Including a safety margin of 1.2, a total of 15kg of food is provided, requiring a volume of approximately 0.03m^3. Because the cabin is depressurized during EVA egress and ingress, the food and water must be stored in a pressurized container. This food storage container has a mass of approximately 1kg, a total volume of 0.1m^3, and can store up to 0.09m^3 of food and water.

9.7 Waste Management

The human body produces the majority of the waste on Freebird in the form of solids, liquids, and gases. Other wastes include used lithium hydroxide canisters, food wrappers, and towelettes. The waste management system on Freebird disposes of all waste produced on board cleanly and compactly to maintain a pleasant and healthy living environment.

9.7.1 Gas Waste

Carbon dioxide is the waste product of human respiration. Humans produce approximately 1kg/day of CO_2. To avoid an accumulation of CO_2 gas in the cabin atmosphere and to comply with NASA regulations (the partial pressure of CO_2 must not exceed 0.4kPa), Freebird contains a lithium hydroxide CO_2 scrubbing system [1].

Table 9.5 shows the different methods considered for carbon dioxide removal. The molecular sieve, solid amine, and liquid-sorbent/membrane contactor systems are all recyclable methods for removing CO_2 from the cabin atmosphere. These systems are appropriate for long-duration missions on a closed-loop environment spacecraft because they have long lifetimes, and they are able to convert the CO_2 to another usable product. These systems require more mass and power and are more expensive than a simple lithium hydroxide system. Because Freebird's missions are limited to four days, the lithium hydroxide system is used.

The LiOH scrubbing system consists of two canisters of lithium hydroxide set up in parallel. Cabin fans circulate the air through one of the canisters until it is used. The air is then circulated through the second canister, and the first is replaced by a new canister. The empty tanks will be stored and disposed of on ISSA. Each canister lasts approximately 24 hours; therefore, each mission will carry five new LiOH canisters.

TABLE 9.5 *Methods of Carbon Dioxide Removal*

System	Mass	Power	Method of Use
Molecular sieve	Medium	High - requires heat and vacuum source	Beds of zeolite material absorb CO_2 and water. CO_2 is then removed by heating the beds [10].
Solid amine	Medium	High	Two beds of solid amine alternately absorb the CO_2 and then desorb it to space vacuum. Beds are heated to remove CO_2 [1].
Liquid-sorbent/ membrane-contactor	Low	Medium	CO_2 rich air is circulated on one side of a hollow fiber membrane contactor and CO_2 is absorbed into the liquid sorbent. CO_2 is removed using a vacuum compressor. Sorbent is recycled [11].
Lithium hydroxide	Low for short-duration missions	Very low - fan used to circulate air through cabin	Cabin air is pumped through disposable LiOH canister. CO_2 combines irreversibly with the chemical and uncontaminated air passes back into the cabin [12].

9.7.2 Liquid Waste

The only liquid waste produced on board Freebird is human urine and perspiration. The astronauts produce approximately 1.4kg/day of urine [1]. This waste must be disposed of cleanly and easily both to avoid disease and to maintain a positive crew attitude. On Freebird, the astronauts use a urine receptacle assembly. The urine is stored on board to be disposed of when Freebird returns to the ISSA.

Several options were considered for a urine collection system on Freebird. The urine transfer system, used on Mercury, Gemini, and some Apollo missions, consisted of a rubber cuff connected to a flexible bag [12]. The waste was either immediately disposed of into the overboard dumping system, or stored in bags for later disposal. The urine receptacle assembly which was used from the Apollo 12 mission onwards, consisted of an open ended hand held cylindrical container. Urine was held in the container by the capillary action of a hydrophilic honeycomb screen. The urine was stored in a dump line until it was full, then the urine was disposed of through a valve open to the space vacuum [12]. The waste management system on the Space Shuttle is a commode/urinal system which resembles the common toilet. The urine is sucked by a fan into a waste water tank which is vented when full [5].

The factors used to choose the liquid waste collection system for Freebird include mass, volume, cost, power requirements, and appeal to crew. The waste collection system trade-offs are shown in Table 9.5. The urinal system is most appealing to the crew because it is clean and easy to use; however, it has a mass of 66kg and costs approximately $5 million [13]. Both the urine transfer system and the urine receptacle assembly have low weight, volume, cost, and power requirements.However, the urine receptacle assembly is more appealing because limited intimate contact is necessary. A system similar to the urine receptacle assembly is used on Freebird because of its low mass, cost, and power requirements, and its medium appeal to crew.

9.7.3 Solid Waste

Humans produce between 0.1 to 0.2kg of solid waste per day [1]. In an enclosed environment such as Freebird, it is important to dispose of this waste cleanly, efficiently, and easily. The waste must be managed so that odor

TABLE 9.6 *Liquid Waste Collection Systems*

System	Mass	Cost	Power Requirements	Appeal to Crew
Urine Transfer System	Low	Low	Low	Not popular because requires intimate contact
Urine Receptacle Assembly	Low	Low	Low	Somewhat unpleasing, but skin contact can be avoided
Urinal/ Commode	High	Millions of dollars	High	Popular - easy and clean to use

and bacteria are controlled. Solid waste is disposed of on Freebird using the glove method and plastic bags for storage. A privacy area is located in the back of the cabin under the common breathing adapter.

Two methods of solid waste management were considered for Freebird. The first was a commode similar to the one on the Space Shuttle. This commode resembles the common toilet. The solid waste is shredded by rotating vanes, deposited on the sides of the commode, dried, and stored [13]. The second method considered was the glove method used on the Apollo missions. A rubber bag which has strength and elasticity is taped to the buttocks. The feces are collected in the bag, and the bag is kneaded to mix in a bactericide which decomposes the waste. The bags are rolled to reduce volume and stored [12].

Table 9.5 shows the trade-offs for the two human fecal waste management systems. The commode is appealing to the crew because it is clean and easy to use; however, its mass, power, and cost requirements prohibit its use on Freebird. The glove method has low mass, power and cost requirements, and has been chosen for use aboard Freebird. In order to ease the burden of using the glove method, the astronauts are fed a diet high in fiber to ensure firm feces.

TABLE 9.7 *Solid Waste Collection Systems*

System	Mass	Cost	Power Requirements	Appeal to Crew
Commode	High	High	High	Popular because clean and easy to use
Glove	Low	Low	Low	Unappealing due to intimate contact required and time for use

9.8 Extravehicular Activities (EVA)

Extravehicular activity (EVA) by the crew of Freebird must be performed to accomplish the mission of repairing satellites in LEO and GEO. Additionally, EVA is required for disabled vehicle crew rescue and for Freebird refueling and repair.

9.8.1 Ingress and Egress

Ingress and egress of Freebird involves transfer from the 70.3kPa cabin environment to the near vacuum of LEO or GEO. Transfer can be accomplished by either airlock or depressurization.

An airlock avoids problems associated with exposing the crew cabin to zero pressure and low temperature, such as providing thermal control for cabin equipment. With an airlock, the cabin can be maintained at room temperature and normal pressures, allowing work to continue in a shirtsleeve environment. Two disadvantages of the airlock are the mass and volume penalties. An airlock, such as that aboard the Shuttle, has a mass of 700kg and occupies a volume of $4.22m^3$. Although an airlock has many advantages, the overwhelming need to keep the mass of the vehicle down necessitated removal of the airlock from the design.

Freebird utilizes cabin depressurization for ingress and egress to EVA. The crew members scheduled for EVA don EMUs (see Section 9.8.2 on page 184). Because the third crewmember does not have to perform EVA work, he or she dons a pressure suit that is not EVA rated. While the EVA crew prepares for egress near the hatch, the third member is restrained in a cabin seat. The atmosphere of the cabin is then pumped out and the exit hatch opened, exposing the cabin to the vacuum and subzero temperatures. All atmospheric pressurization and depressurization controls are easily operable while wearing the pressurized suit or EMU gloves. After the EVA crew leaves the cabin through the common berthing adaptor (CBA), the cabin is repressurized. The total time from start of depressurization to end of repressurization is no more than 30 minutes. The cabin is repressurized to allow the remaining crew member to effectively operate the RMS controls without the hinderance of the pressure suit. The RMS controller dons the pressure suit again prior to the second depressurization for ingress of the EVA astronauts.

9.8.2 Extravehicular Space Suit

The extravehicular space suit is essentially a self contained spacecraft designed to support an astronaut during EVA. It also provides mobility in order to allow completion of tasks outside the vehicle. The choice of the suit used for EVA directly impacts upon the preparation needed for EVA.

Denitrogenation and Prebreathing

When transferring from the higher pressure environment aboard Freebird to the lower pressure of a space suit, there is the danger of decompression sickness or "bends" ("Pressure" on page 177). For transfer into EVA, the evolution of nitrogen bubbles in the bloodstream causes the symptoms. If the pressure ratio between the nitrogen partial pressure in the blood (pN_2) and the suit pressure is less than 1.5 to 1.8, then no decompression sickness occurs. The incidence of bends increases when the ratio is above 1.8. In this case, the pN_2 level in the astronaut's blood must be lowered prior to EVA. This can be done by nitrogen equilibration and preoxygenation. Nitrogen equilibration involves lowering the cabin atmosphere pressure, and thus the nitrogen partial pressure, and allowing the crew members to equilibrate to the new nitrogen levels. No special equipment is required. Several hours are required for the nitrogen levels in the blood to reach equilibrium levels. Preoxygenation, also known as prebreathing, involves breathing pure oxygen through a mask before donning the space suit in order to displace the nitrogen from the blood. By using one of the methods or a combination of both methods the final pN_2 to final suit pressure ratio can fall to acceptable levels [2].

Suit Selection

Extravehicular space suits can be divided into two general classes: high pressure suits and low pressure suits. High pressure suits have an internal atmospheric pressure of at least 57.2kPa. Low pressure suits, such as the Space Shuttle's Extravehicular Mobility Unit (EMU), have a pressure lower than 57.2kPa. The EMU operates at a pressure of 29.6kPa. No prebreathing is required for the high pressure suits when transferring from a sea level pressure environment. A combination of nitrogen equilibration and preoxygenation is required for low pressure suits.

High pressure suits, such as the Mark III Zero Breathe Suit developed by NASA/Johnson Space Center, have been developed and tested on Earth. They have not yet been tested in space, but would allow EVA without pre-breathing [14]. The major disadvantage of high pressure suits is articulation in the glove. Currently the high pressure suit glove does not allow the same flexibility as the EMU glove. Manipulation intensive operations such as those in satellite repair would take more time and effort or be impossible with a high pressure suit. Additionally, in 1989 the former Space Station Freedom Program decided to use the Shuttle EMU for EVA at the Space Station, cancelling the use of high pressure suits[15]. Therefore, although the zero prebreathe time would greatly decrease the amount of preparation for EVA, a high pressure suit was not chosen because of the inadequate glove flexibility, untested design, and previous Space Station decisions. However, further development of high pressure glove technology may allow for future transition to a high pressure suit.

The Shuttle EMU was chosen as the EVA suit for Freebird (see Figure 9.6) because of its proven technology and immediate availability. Three EMUs are carried onboard the vehicle, though only two are used at any one time. The pressure within Freebird's cabin is lowered from 101.3kPa to 70.3kPa after release from the International Space Station Alpha (ISSA). The pressure ratio in this case is only 1.45, and no risk of bends is incurred. Since the EMUs operate at 29.6kPa, additional prebreathing must be done to avoid the bends (ratio of 70.3kPa to 29.6kPa equals 2.38). If the body is allowed to equilibrate to 70.3kPa for 24 hours, the prebreathe time is only 40 minutes. For EVA from a sea level pressure of 101.3kPa, a prebreathe time of 3 hours is necessary. Since the maximum travel time to a LEO asset is 6 hours, there is not enough time for complete equilibration, although nitrogen levels do drop during the transit. Therefore, prebreathing of 3 hours is needed before the first EVA. Since the crew cabin is kept at 70.3kPa for the rest of the mission, the remaining EVA's only require 40 minutes of prebreathing each [16].

Extravehicular Mobility Unit Components and Related Equipment

The EMU primarily consists of a Space Suit Assembly (SSA), which is the main body of the suit, and a Life Support System (LSS), which maintains the environment within the suit. The mass of a charged EMU is 127.3kg. The astronaut first dons a liquid cooling and ventilation garment, which resembles long underwear. This provides temperature control of the astronaut by drawing off heat using tubes with cooling water and sublimating the water into space. A urine collection device draws off liquid waste while an in-suit drink bag supplies water for ingestion. The SSA itself is made up of a fiberglass upper torso and a rolling convolute, fabric joint system. The upper and lower torso is composed of Dacron, nylon, and Teflon fabrics to provide micrometeoroid and thermal protection. The joint system keeps the internal volume constant through the range of motion necessary to do EVA work, thus allowing flexibility at all times. A helmet with visor, made of a Lexan-polycarbonate shell, provides protection from debris and radiation while allowing the normal field of vision with the head in the normal, face-forward position. The gloves are custom made to each astronaut. The LSS is located on the back of the SSA. It can function up to eight hours before requiring recharge. This includes 30 minutes for egress, 6.5 hours of useful EVA time, 30 minutes for ingress, and a 30 minute reserve. The 30 minute reserve of oxygen is supplied by the secondary oxygen pack which operates an open loop supply at reduced suit pressure. The LSS has closed loop ventilation, removing excess carbon dioxide using lithium hydroxide (LiOH) canisters, which are held in the Contaminant Control Cartridge (CCC). It also holds activated carbon to clean the air of any contaminants and odors. An internal fan provides ventilation. The display and control module provides information on all vital systems to the astronaut [17].

The power is supplied by a 4.3kg silver-zinc battery. Stored dry, it is activated by filling with electrolyte. After activation, it has a 90-day shelf life. The battery is designed for 6 discharge/recharge cycles. Recharging of one battery requires a power of 22.6W and takes 20 hours to complete. A total of 10 batteries are carried on board the Freebird during a mission. The extra batteries replace the spent batteries, which are recharged during stand-by mode[18].

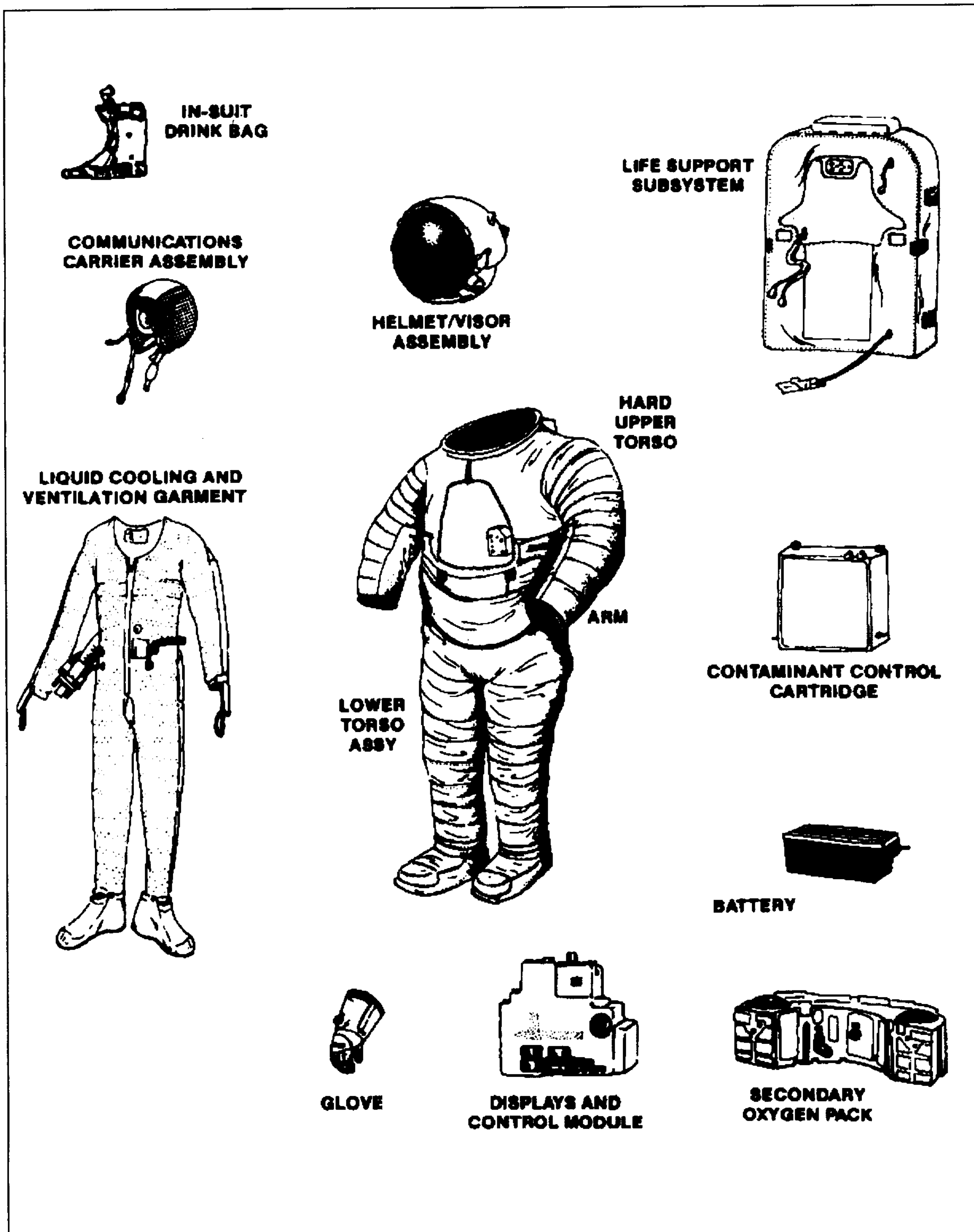

FIGURE 9.6 *Extravehicular Mobility Unit (EMU) [17].*

After each EVA the EMU must be recharged. This includes replacing the battery, refilling the oxygen tanks, replacing the spent LiOH canisters, removing urine waste, replacing the drinking water, and replacing the spent cooling water. Oxygen and cooling water connections are located within the Freebird cabin. The battery and LiOH canisters are removed manually. The total time for recharging is 3 hours [19].

Communication between astronauts in EVA and with Freebird is carried through a two-way communications carrier assembly. It is made up of a microphone and headset and transmits and receives in the Ultra-High Frequency (UHF) radio range. Its transmission power is 0.25W, enough for a communication range between 70m and 9km, depending on orientation with Freebird's antennas and blockage from satellite structures [17].

Attachments to the EMU to facilitate EVA maneuvering and satellite repair are the EMU lights and the EMU TV. Batteries to these accessories are nonrechargable and must be replaced after each EVA. An EMU TV receiver and video processor assembly on board the Freebird receives the video signals from the EMU TV and conditions them for presentation to Freebird's monitors [18].

9.8.3 Exterior Human-Vehicle Interfaces

To perform satellite repair the astronauts need interfaces with the outside of Freebird's crew module. These include cargo holds, handrails, foot restraints, and lights.

Two cargo holds are located near the rear of the crew module (Figure 9.3 on page 175). The volume of each cargo hold is approximately 1 m^3. One cargo hold contains the tools necessary for satellite repair. The tools are divided into those for satellite repair, those for Freebird repair, and general purpose tools. EVA power tools use separate EMU batteries for energy. Additional space is available for specialized tools for specific repair missions. The second cargo hold contains new modules or components to be installed in the dysfunctional satellite [18].

Handrails on the side of the vehicle allow movement of the astronauts to the satellite and the cargo holds. As the astronauts exit the vehicle they attach tethers to the nearest handrail to secure them to the vehicle. The astronauts are required to have at least one tether connection to the vehicle or the satellite at all times. Thus, to shift from one handrail to another, a second tether is attached to the second handrail before disconnecting the first tether. The maximum spacing of the handrails is 60cm to always allow a tether connection. Both handrails and handholds are available to allow translation near the CBA, to the cargo holds, and to the vicinity of the satellite. The handholds are designed to a minimum crew-induced zero-g load of 830N [17].

Lighting will be supplied by two floodlights to allow work while traversing Earth's shadow. Since Freebird spends about 40% of the time in darkness at LEO, the lights are necessary for effective repair and crew safety. The lights consume 250W of power each during EVA. EMU lights supplement the EVA lights.

9.8.4 Satellite Acquisition and Repair

During a nominal four day mission, three EVA's are performed. Two crewmembers perform the EVA while the third crewmember remains inside to operate the RMS. In order to prevent fatigue, the crewmembers rotate EVA shifts, and thus perform a total of 2 EVAs each. Therefore, each crewmember must be able to both perform satellite repair in EVA and operate the RMS.

The first step after the satellite is within view is to stabilize it and provide a hard contact between it and Freebird. Utilizing the Manned Maneuvering Unit (MMU) for this purpose was considered at the early design stage. The MMU would provide more safety for the Freebird because only the astronaut in the MMU would approach the satellite before stabilization. The MMU option has been eliminated because of its prohibitively high mass and volume. Instead, the Freebird uses its RMS arms to directly stabilize and grapple the satellite. The satellite is then secured with one of the arms. The two crewmembers scheduled for EVA perform prebreathe as explained previously. Since the pressure suit operates at 55.2kPa, no prebreathing is required for the third crewmember. After depressurization the outer door is opened, the astronauts attach their tethers to the appropriate handrail and translate to the cargo holds. The necessary repair tools are removed and stowed upon EMU. The module cargo hold is also accessed if parts or modules must be replaced.

One astronaut is secured to the RMS by a foot restraint along the mechanical arm. The other astronaut is secured either to the Freebird outer surface or to the satellite itself. For the most efficiency and least fatigue, free floating is discouraged while maneuvering modules or using tools [20]. Foot restraints on Freebird or the satellite provide secure bases to work from. The RMS controller uses the views from the cameras on the arm and the EVA helmet as well as the RMS window to move the RMS arm and the EVA astronaut to appropriate locations around the satellite. Loading on the EMU suit is minimized by keeping objects held by an astronaut within his or her reach envelope. Additionally, metabolic constraints keep the workload within defined limits. The power output of the astronaut cannot exceed 293W over the entire EVA, 469W in a given hour, or 586W for 15 minutes. Exceeding these limits may result in overheating of the suit environment. The astronaut also cannot output less than 117W for more than 30 minutes or risk hypothermia. Work on the satellite during a given EVA cannot exceed 7.5 hours because of the oxygen supplies of the EMU. Before ingress, the satellite is properly configured so should immediate return to ISSA prove necessary, no further EVA's are required. The tools and any removed modules are stowed before reentry into the vehicle. Maximum EVA work time before ingress back to the vehicle is 7.5 hours. Following ingress and removal of the EMU suits, refurbishing of the suits is completed.

The EMU releases oxygen, water vapor, and electromagnetic radiation to the surrounding environment. This may be of concern for sensitive satellite surfaces that may be contaminated. The EMU discharges oxygen and sublimates water into the surrounding environment. Any sensitive satellite components must be shielded should the emission levels prove threatening. Additionally, EVA procedures can be modified to reduce the contamination.

EVAs are limited to LEO only [21]. GEO EVAs were ruled out because of the high levels of radiation present created by high energy electrons bombarding the suit. EVAs in LEO are restricted to altitudes below 600km because of greatly increasing radiation levels from the Van Allen Belts at higher altitudes. In addition, polar EVAs are not permitted because of high radiation levels from other sources, including flares and galactic radiation [22].

9.8.5 Disabled Spacecraft Crew Rescue

EVA's can be performed to rescue crewmembers from malfunctioning spacecraft such as the Space Shuttle or the Russian Soyuz. If the vehicle has a CBA, docked transfer is used. Otherwise, one of Freebird's RMS arms grapples the other vehicle. Once this hard connection is in place, the crew prepares for egress as described previously. The EVA astronauts then use the arm to translate across to aid the other crew. The crew of the other craft is expected to have enough EVA suits or personal rescue enclosures to transfer all members to Freebird. Once transfer of the crewmembers is complete, the Freebird EVA crew disengages the RMS and reenters the cabin. The cabin is repressurized and the rescued crew members take their seats for the trip back to ISSA.

9.9 Emergency Systems

Freebird is equipped to handle basic emergencies while in flight. These emergencies include medical problems, cabin atmosphere variations, and fire.

9.9.1 Medical Equipment

The cabin contains a medical kit stocked to accommodate two types of medical scenarios: work related incidents and spontaneous medical events. Work related incidents include trauma, exposure to toxic substances and radiation, burns, and electric shock. Spontaneous medical events range from constipation and diarrhea to fatigue.

The medical kit contains an emergency pack and a bandages/medications kit. The emergency pack contains sutures, disposable thermometers, a stethoscope, and injectable medications. All medical emergencies are stabilized on board, and then attended to when the craft returns to the Station. The bandages/medications kit includes bandages, adhesive tape, gauze bandages, pain killers, decongestants, antibiotics, anti-diarrhoeal preparation, and eye drops [1].

The cabin contains a tool box stocked with simple emergency repair tools to temporarily fix a system which will be properly fixed when Freebird returns to the ISSA. This tool box contains basic hand held tools and duct tape.

9.9.2 Atmosphere Regulation

The atmospheric partial pressure of the cabin gases must be monitored on board Freebird. It is also necessary to be able to detect low levels of toxic substances in the cabin atmosphere. Freebird contains electrochemical amperometric gas sensors which monitor the partial pressure of carbon dioxide and oxygen, as well as carbon monoxide, nitrogen dioxide, ammonia, and moisture in the air [23]. The electrochemical amperometric gas sensors are used because they require little power, have a fast response time, are small in size, and have multi-gas sensing capability.

To remove trace contaminants from the cabin atmosphere, activated charcoal is located downstream of the lithium hydroxide canisters used for carbon dioxide scrubbing. Within the charcoal, carbon monoxide is converted to carbon dioxide by an ambient temperature catalytic oxidizer, while filters remove airborne particulates [24].

9.9.3 Fire

Early detection and quick suppression of a fire is essential in a small, closed environment such as Freebird. A fire can spread quickly and severely contaminate the cabin atmosphere. Early detection of a fire allows a reasonable amount of time for effective fire protection actions, and reduces the amount of contaminants that are released into the cabin environment. Two fire detection methods have been considered for Freebird: the photoelectric cell and the ionization chamber. Both detection systems are similarly effective; however, the ionization chamber is used because it has a faster response time than the photoelectric cell [25]. The ionization fire detector on Freebird contains a particle separation device upstream of the sensor to prevent non-hazardous aerosols from setting off a false alarm. Six fire detectors are located around the cabin.

An effective and clean means of fire suppression is necessary for the survival of the crew and the maintenance of a constant cabin atmosphere. Two fire suppression systems have been considered for Freebird: a carbon dioxide suppression method and a Halon suppression method. The Halon system is used because it is less toxic and it requires less volume to suppress a fire than the CO_2 system. Halon 1301 is easily scrubbed from the cabin. The Halon will decompose into acid gases which are absorbed by the LiOH used in the CO_2 scrubbing system. The rest of the Halon will be absorbed by the charcoal located in the LiOH system [25]. Freebird contains four portable halon fire extinguishers and a supply tank of Halon 1301.

9.10 Failure Modes

The major failure modes for ECLSS are shown in Table 9.5.

A structural failure of the crew seats may result in life-threatening injury to crewmembers. To prevent such a failure, the actual flight hardware is subjected to extensive infant mortality testing to reduce the possibility of failure.

A failure of the atmospheric control system may involve a fan motor malfunction, halting air circulation throughout the cabin, and preventing normal operation of the CO_2 removal system and thermal control system. The chance of such a failure is reduced through redundancy. Another failure involves a defective sensor or valve, causing fluctuations in cabin pressure, temperature, and gas concentrations. Redundancy, a pressure suit, additional crew clothing, and a portable O_2 supply can alleviate the effects of such a failure.

Rupture or leakage of the O_2 tank may result in explosion or loss of oxygen supply. A fully redundant tank in conjunction with pressure sensors and extensive testing can mitigate the effects of such a malfunction.

A tear or puncture of the pressure suit results in the loss of suit pressure. Immediate repressurization of the cabin may prevent such a failure from becoming life-threatening.

Failure of the carbon dioxide removal system may occur if the lithium hydroxide crystals become saturated with CO_2. This may cause an increase in the CO_2 concentration in the cabin. Redundancy can alleviate the effects of such a failure.

A failure in the fire extinguisher may result in the rapid spreading of a cabin fire. Redundant extinguishers located throughout the cabin help prevent this failure from becoming a threat to the crew.

Leakage of the portable O_2 supply results in the loss of the emergency oxygen supply. This life-threatening failure is prevented by carrying redundant portable supplies aboard Freebird.

Leakage or failure within the EMU may result in a loss of suit pressure and lack of air circulation. The EMU Secondary Oxygen Supply provides a partial redundancy as a response to such a life-threatening failure

9.11 Summary Budgets

Table 9.9 shows the mass, power, and cost breakdown of the components of the ECLSS. The summary is given specifically for the satellite repair mission involving a crew of three astronauts. On a rescue mission, the cabin accommodates five additional seats, requiring an additional mass of 100kg. Both the satellite repair and crew rescue mission place similar demands on the ECLSS.

Table 9.10 shows the telemetry requirements for the components of the ECLSS.

TABLE 9.8 *Failure Modes*

Component	Failure Mode	Effect	Prevention or Response	Criticality	
				Crewed Config.	**Teleop.**
Human Interface Controls	Control failure Broken monitor	loss of communication	Secondary monitors / controls	2	N/A
Restraints, Seats	Structural failure	Injury to crew members	Infant Mortality Testing	1	N/A
Atmosphere Control System	Fan motor failure	No air circulation	Redundancy	2	N/A
	Failure of atmosphere sensors or valves	Pressure, temperature, gas concentration changes	Redundancy, pressure suit, clothing, portable O_2 supply	1	N/A
O_2 Tank	Leakage / Blockage	Loss of O_2 supply	Redundancy	1	N/A
	Rupture	Explosion, over-pressurization	Testing, tank pressure sensors	1	N/A
Pressure Suit	Tear or puncture	Loss of suit pressure	Testing, repressurization	1	N/A
Carbon Dioxide Removal System	LiOH saturation	Rise in CO_2 concentration	Redundancy, replace LiOH canisters	1	N/A
Cabin Air Storage System	Rupture of storage tank	Explosion, overpressurization	Testing, tank pressure sensors	2	N/A
	Compressor motor failure	Unable to depressurize	Testing	2	N\A
Food, Water, and Containment System	Containment leak / spoilage	Fewer provisions available	Provision rationing	2	N/A
Emergency System	Broken fire extinguisher	fire spreads	Redundancy	1	N/A
	Leakage of Portable O_2 Supply	Loss of emergency O_2 Supply	Testing Redundancy	1	N/A
EVA Tools	Breakage / Inappropriate sizing	Unable to finish repair	Alternate tools	2	N/A
Extravehicular Mobility Unit (3)	Leakage / Pump failure	Pressure loss/ No air circulation	Secondary Oxygen Supply	1	N/A

TABLE 9.9 *Summary Budgets [2][3][5][6][8][21][26][27]*

Component	#	Mass [kg]	Avg. Power [Watts]	Peak Power [Watts]	Hardware Cost [$K]
Lighting	4	5	30	50	423
Human Interface Controls	1	20	100	200	1,693
Restraints, Seats	3	60	0	0	1,587
ATMOSPHERIC CONTROL SYSTEM					10,000
- Pressurized Oxygen Tank	2	35	0	0	
- Oxygen in Tanks	2	38	0	0	
- Cabin Air Storage System	1	68	--	5000	
- CO_2 Removal System	2	10	0	0	
- LiOH Cannister	5	25	0	0	
- Cabin Fans	2	6	60	60	
- Flow Valves	11	6	0	5	
- Sensors	6	6	0.5	1	
Pressure Suit	1	10	0	0	847
Food, Water, and Containment System	1	80	0	0	776
Waste Disposal System	1	5	0	0	423
EMERGENCY SYSTEM					
- Portable O_2 System	8	16	0	0	1,355
- Fire Detection Sensors	6	3	1.5	2	200
- Fire Extinguishers	4	16	0	0	212
- Medical Kit	1	2	0	0	21
EVA Tool Set	1	100	0	0	100
Extravehicular Mobility Unit	3	380	0	0	30,000
Extra EMU batteries	10	43	0	20	100
EVA Lights	2	20	--	500	423
Total	--	954	192	--	50,688

TABLE 9.10 *Telemetry*

Component	Telemetry	# of lines	Freq. [Hz]	Analog/ Digital	Command
Human Interface Controls:					
Monitors	Status	1	1	D	
Joysticks	Position	6	20	A	
RMS Monitors	Status	1	1	D	
RMS Joysticks	Position	6	20	A	
Pressurized Oxygen Tank	Temperature	1	1	A	
	Pressure	1	1	A	
Cabin Air Storage System	Temperature	1	1	A	
	Pressure	1	1	A	
	Compressor Status	1	1	D	On/Off
Cabin Fans	Air Flow	1	1	A	Fan Velocity
Flow Valves	Status	1	2	D	Open/Close
Cabin Sensors	Atmospheric Param.	6	1	A	
Pressure Suit	Pressure	1	1	A	
Fire Detection Sensors	Fire	6	1	D	Alert Crew
Cargo Bay Lights	Status	1	1	D	On/Off
Extravehicular Mobility Unit	Pressure	1	1	A	
	Life Signs	6	1	A	

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Chapter 10

Verification and Validation

The following chapter is a detailed approach to the verification and validation of Project Freebird. A rigorous testing methodology is essential to this type of manned aerospace system to ensure adequate reliability. While such a testing program can be quite costly and time consuming, it allows the system to adhere to its functional requirements in a variety of mission scenarios.

10.1 General Test Procedures

The overall testing approach will follow many of the recommendations set forth by government standards. In addition, it will employ advanced testing and management approaches to minimize the cost of testing. The Project Freebird testing program consists of a series of functional and environmental tests at different levels of development and integration. Such a program allows for the functional requirements of the system to be met in the most efficient way possible.

Testing and reliability concerns have already been used in the design process of Project Freebird. The design was molded through an iterative process into one that allows for ease of testing and a high level of reliability. Thus, verification and validation issues have already been built into the system.

The goals of the verification and validation program are quite simple, but reaching these goals required careful planning and teamwork. Overall, the testing program has been designed to reach a reliability of .9944 while maintaining a certain schedule and minimizing cost. Thus, Project Freebird will use testing methods that will allow for a high level of efficiency.

10.1.1 Management Philosophy

The use of Total Quality Management (TQM) tools in the testing and production of Project Freebird will drastically minimize cost and improve efficiency. In some recent development programs, this type of management has been able to cut testing costs by 25%. Such a program encourages continuous improvement, employee involvement, and customer satisfaction. The main aspect of this management strategy as applicable to this project is feedback. By using test results to improve the production and testing process, Project Freebird will be able to move away from fault detection and move towards fault prevention. In addition, the testing program will be flexible to allow for the type of continuous improvement that will generate greater efficiency.

There are two major levels of testing management that such a program must employ. These are the intradepartmental and interdepartmental levels. Intradepartmental management will contain working groups that usually focus on a certain process. Interdepartmental management will govern the interfacing between groups that focus on different tests or on different production tasks.

Performance Refinement Teams (PRTs) will be organized at the intradepartmental level. These teams will meet regularly and should consist of all employees from a natural work group. In addition, appropriate support personnel will be included in these teams. During the meetings, the members will discuss approaches and implement actions focused on improving day-to-day operations and actions within the group.

Process Improvement Teams (PITs) will be organized at the interdepartmental level and these will also meet regularly. They will consist of the PRT group leaders from every group in an affected organization. The PITs will discuss their operations and processes to allow everyone to learn valuable lessons about all stages of production and testing. It will also be an appropriate forum for feedback among groups in order to form a better overall production/testing flow. Before the production and testing of Project Freebird begins, there will be a TQM steering committee formed in order to implement these programs. In addition, there will be time to train and orient team leaders so that the teams will be able to work most effectively.

10.1.2 Testing Philosophy

The overall test philosophy of Project Freebird is intended to prove the integrity of the system with the least financial and schedule risk. Years of industry verification and validation experience have led to several important elements of any testing program. The key elements to be used here are:

- Continuous monitoring of all performance parameters during all portions of environmental exposure.
- Testing in a sequence that places the most stressful environments first.
- Conducting functional tests immediately before and after each environmental test.
- Utilizing forced infant mortality.
- Performing qualification and acceptance tests in the same test chamber.
- Allowing for protoflight tests whenever possible.
- Utilizing technology risk management in forming a test program.
- Conducting system-level qualification on the flight test article.

Unsatisfactory test results in this program will be dealt with swiftly. Any change in the performance of a test article that prevents it from meeting all of its requirements will be considered a failure and corrective action will be taken immediately. Following this action, the test will be repeated until it results in satisfactory performance. However, if the correction affects the certainty of previous tests, then the article will be retested until the results are acceptable.

Forced infant mortality will allow for greater testing efficiency of Project Freebird. It is the combination of selectively testing immature technologies more rigorously and applying decreasing environmental levels to all testable items. The goal of such a program is to learn as quickly as possible which design ideas are bad ones, and to avoid accumulated value in components, subsystems, or systems that have latent failure modes. By applying forced infant mortality, the project will save money by noticing certain failure modes early in the program. The testing sequences have been formed with this concept in mind.

The protoflight option will be used extensively in this program. Protoflight testing is the qualification testing of the actual flight articles. This technique eliminates the need for developing a prototype article and decreases testing redundancy. Protoflight tests usually employ somewhat decreased environmental levels, and almost always decrease the testing duration. This minimizes the fatigue of the articles while ensuring that they can handle the appropriate stresses. There is some risk in using protoflight articles however, and they should only be used in redundant strings of components and subsystems. Also, protoflight testing should be avoided with those articles which are critical to the safety of the mission. In addition, when an article is technologically immature, protoflight testing will not be used and a prototype will be built. For the mature technology however, protoflight testing will be used. At the systems level protoflight testing will be used exclusively to eliminate the need of a full-scale system prototype.

10.1.3 Testing Process

There are several different types of verification methods that will be used throughout the production process. Testing is the most extreme method, and will be used sparingly. At all other levels of development, the following methods will be used to determine the quality of an item:

- Analysis: A technical or empirical method that uses mathematical calculations or computer simulations to predict actual design operation. It will be used to verify requirements when testing is impractical, but will only be used when the established techniques are adequate to provide confidence.
- Demonstration: A process where actual crew conduct or use can verify compliance with a requirement.
- Inspection: A visual analysis method which verifies design features, compliance to drawings, workmanship, and physical condition. This method will be used after each test to insure that no visible damage has occurred.
- Similarity: The process of verification that uses prior test data when an article is similar in design and manufacturing process to another article which has been previously verified to equivalent or more stringent specifications.

Figure 10.1 shows the sequence of different tests, all of which are important to the entire verification and validation process. The development tests are those examinations conducted to obtain information that assists in the design and manufacturing process. It is only used on those objects with the lowest technological maturity, and often helps plan for a technological innovation. Its main purpose is to discover problems, or to determine the type of technology to use. It includes tests of structural and thermal models to confirm environmental criteria.

The qualification tests demonstrate that a component, subsystem, or system performs as required in the simulated environment. These are the most rigorous of tests and measure up to the factor of safety. They only test a representative sample of the population and are not conducted on actual flight components. The purpose of these tests is to check the design, manufacturing, and assembly process.

The acceptance tests demonstrate the acceptability of an item for delivery and act as quality control. All of the actual flight components must undergo these tests. The loads imposed by the tests are kept to a minimum to decrease added fatigue. If the protoflight option is used on certain articles, then the protoflight tests levels and durations are used in place of the qualification and acceptance tests.

The integration and systems tests evaluate the functionality of the system during its integration. In addition, these tests subject the entire system to environmental levels that provide a final system qualification. The integration tests will also occur on-orbit while the modules are constructed or directly following reconfiguration of the modules for a certain mission.

The pre-launch tests are the last verification performed on the ground. It is the final check to insure that the system has not been damaged during shipment to the launch site. In addition, it demonstrates the compatibility of the system with the launch vehicle. Once the system reaches orbit, a series of in-flight tests will be conducted which qualify the vehicle for the environment in which it was designed. These tests also ensure that the launch and ascent environment did not degrade the integrity of the system.

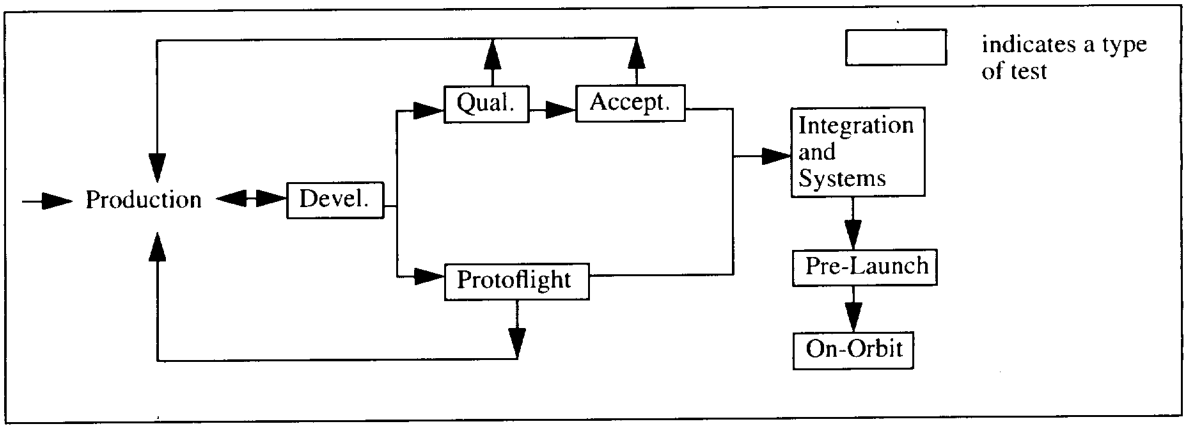

FIGURE 10.1 *Test Flow Diagram.*

10.2 Ground Testing

Ground testing is a cost-effective method of ensuring that a system functions properly and safely. The lessons learned through this type of testing allow problems to be fixed early without the difficulty of an on-orbit repair. It also allows for a constant refining of the production and design process.

10.2.1 Testing Parameters

There are several environmental and functional tests which Project Freebird will undergo. Definitions of these tests are as follows:

Acceleration (A): Demonstrates that the system can withstand the acceleration loads imposed on it in the launch, ascent, and operational stages of its lifetime. Each component that is sensitive to this environment must undergo ±20g's for five minutes in each of the orthogonal axes. If there are no failure modes associated with this environment then analysis will be used.

Acoustic (AC): Demonstrates the ability of the system to withstand the acoustic and vibrational environments imposed during launch and ascent. This test depends upon a component's response to the acoustic environment.

Burn-in (BI): Ensures that the system will be able to handle the thermal environment. It requires 300 hours of continuous operation (including 18 thermal cycles) with the last 100 hours failure free. It will be conducted in the same set up as the thermal tests. Previous thermal testing will be counted towards the time (or cycle) goal.

Burst Pressure (P): Verifies the design integrity of the system and its lack of brittleness. All pressurized components will undergo this test. Pressurized articles will be proof-pressure tested to 1.5 times the maximum operating temperature at the critical use temperature for three complete pressure cycles. Burst tests will be conducted at this temperature at 2.0 times the maximum operating pressure. The lines, hoses, and pressure components will be tested at 2.0 times the proof pressure and 4.0 times the burst pressure.

Electromagnetic Compatibility (EMC): Verifies that equipment will not be susceptible to EM environments during operation. It also ensures that the space vehicle has an adequate margin in a simulated launch and orbital EM environment. This test subjects the system to all possible launch modes in various launch and orbital configurations. All electrical and electronic components must undergo this test.

Functional (F): Establishes correct performance of hardware and software and proves that no degradation occurs during the environmental tests. It is conducted before and immediately after each environmental test. Abbreviated functional tests will occur during the environmental tests to discover intermittent failures.

Leak (L): Verifies leakage rate requirements. It will be conducted on all pressurized components before and immediately after thermal and random vibration acceptance and qualification tests.

Life Cycle (LC): Verifies the design life performance. It checks the ability of the system to withstand the maximum duration of cycles that it was designed for without fatigue or failure.

Pyro Shock (PS): Verifies acceptable performance during and after pyro actuated functions. It is required on all pyro operated devices, deployables, and other equipment capable of imparting a shock impulse to the vehicle.

Random Vibration (RV): Demonstrates the capability of the system to withstand the imposed dynamic loads of the launch vehicle. The system is tested in three orthogonal axes.

Sine Vibration (SV): Determines the resonant frequencies and the damping coefficient. It also identifies resonant conditions which could result in flight failure. The system is tested in three orthogonal axes.

Thermal Balance (TB): Verifies the adequacy of the thermal control system. These tests will be conducted on the fully assembled payload while using analytical verification on the lower levels of assembly.

Thermal Cycling (TC): Verifies the operational ability of a design over a complete temperature range. It should be conducted when an item is not susceptible to a vacuum environment.

Thermal Vacuum (TV): Demonstrates the ability of an item to perform in the thermal vacuum environment of space. It is required if a unit has a failure mode associated with a vacuum environment.

10.2.2 Test Tolerances

Table 10.1 shows the test requirements for various testing methods at the component and subsystem level. (dB represents decibels.) The numbers in parentheses following any test level or duration describe the minimum level that must be achieved in that test. When a test is not mentioned, then the values are the same for all levels of testing. These values are described in the definitions for each test in Section 10.2.1.

All system level testing will be performed on the actual flight vehicle. This testing will adequately qualify the system without degrading its useful lifetime. Table 10.2 shows the level and duration required for these tests.

10.2.3 Testing Breakdown

Through the course of testing Project Freebird, there will be different tests conducted at different levels of assembly. Throughout the verification process, depending upon the maturity of the completed system, testing at the component, subsystem, and system level will be used in order to verify that it has been properly designed and produced. Such intermediate testing allows problems to be discovered early while at the same time allowing the entire system to be qualified. Table 10.3 and Table 10.4 show the different tests required of specific components and subsystems. Table 10.3 indicates whether protoflight testing (*PF*) will be used, in which case the qualification tests will be conducted at the protoflight levels and there will be no acceptance tests. The order of these tests is detailed in Section 10.2.4 on page 207.

TABLE 10.1 *Component and Subsystem Testing Levels*

Test	Qualification	Protoflight	Acceptance
Acoustics:			
Amplitude	Accept + 6dB (>144dB)	Accept + 6dB (>141dB)	95% Flt (>138dB)
Duration	(3.0)*Accept	(3.0)*Accept	Flt Length (>1 Minute)
EMC:			
Susceptibility (RF)	Flt Predicted + 20dB	Flt Predicted + 20dB	None
Emissions (RF)	For Pyro Cir. > 60dB	For Pyro Cir. > 60dB	None
Pyro Shock:			
Level	(2.0)*Accept	(1.5)*Accept	95% Flt
Quantity	3 Shocks/Axis	3 Shocks/Axis	1 Shock/Axis
Random Vibration:			
Amplitude	(4.0)*Accept	(2.0)*Accept	95% Flt
Duration	(3.0)*Accept	(3.0)*Accept	Flt/Axis (>1 Minute/Axis)
Sine Vibration:			
Amplitude	190% Flt	190% Flt	None
Sweep Rate	.5 Oct/Minute	.5 Oct/Minute	None
Temperature:			
Level	Flt +/- (10 to 21° C)	Flt +/- (5 to 16° C)	Flt +/- (0 to 11° C)
Duration	24 Cycles	4 Cycles	8 Cycles

TABLE 10.2 *System Testing Levels*

Test	Protoflight Level
Acoustics:	
Amplitude	95% Flt + 3dB (>141dB)
Duration	Flight Length (>1 Minute)
EMC:	
Susceptibility (RF)	Flt Predicted + 20dB
Emissions (RF)	For Pyro Cir. > 60dB
Pyro Shock:	1 to 3 Device Firings
Temperature:	
Level	Flt +/- (5 to 16° C)
Duration	4 Cycles with 8 Hours at Each Extreme

TABLE 10.3 *Component and Subsystem Qualification Tests.*

Components/Subsystems	PF	F	L	PS	AC	RV	SV	A	TC	TV	P	EMC	LC
C^3T													
TDRSS Transponder	X	X	X	X		X				X		X	
EVA Transponder	X	X	X	X		X				X		X	
Power Amplifier		X		X		X				X		X	
Parabolic Antenna	X	X				X		X		X			
S-Band Omni Antenna	X	X				X		X		X			
EVA Omni Antenna	X	X				X		X		X			
Internal Communication Network		X		X		X				X		X	
Harness		X		X		X				X		X	
Processors	X	X		X		X				X		X	
Data Recording Unit		X	X	X		X				X		X	
ECLSS													
Oxygen Tank		X	X			X				X	X		
Extravehicular Mobility Unit		X	X		X					X	X	X	
Extra EMU Batteries	X	X				X				X			
CO2 Removal System		X	X			X				X	X		
Atmosphere Control System		X	X	X		X				X	X	X	
Cabin Air Storage System		X	X			X				X	X		
Pressure Suit		X	X			X				X	X		
Waste Disposal System		X	X			X				X			
Food, Water, and Containment System		X	X			X				X	X		
LiOH Cannister		X	X			X				X	X		
Human Interface Controls		X	X	X		X				X		X	
Lighting	X	X	X	X		X				X		X	
Restraints, Seats	X	X			X					X			
Cabin Fans		X	X			X				X			
Portable O2 System		X	X			X				X	X		
Fire Detection System		X	X	X		X				X	X	X	
Fire Extinguisher		X	X			X				X	X		
GNC													
DCS with Sun Fans	X	X	X	X		X		X		X		X	
Rendezvous Radar Transponder		X	X	X		X				X		X	
MIMU	X	X	X	X		X		X		X		X	
MANS System	X	X											
MAPS	X	X											
Star Tracker	X	X	X	X		X		X		X		X	
LNS		X	X	X		X	X	X		X		X	X
Docking Camera	X	X	X	X		X				X		X	
Power and Propulsion													

TABLE 10.3 *Component and Subsystem Qualification Tests.*

Components/Subsystems	PF	F	L	PS	AC	RV	SV	A	TC	TV	P	EMC	LC
Propellant Tanks		X	X	X	X		X			X	X		X
Plumbing - Valves and Pipes		X	X			X				X	X		
RL10A-4 Engine		X			X		X			X	X		X
Thermal Venting System		X	X			X				X	X		
RCS Tanks		X	X			X				X	X		
RCS Valves and Pipes		X	X			X				X	X		
RCS Thrusters		X				X				X	X		
Fuel Cell Powerplant		X	X	X	X					X	X	X	
Power Conditioner	X	X	X	X		X				X	X	X	
EPS Wire Harness		X				X				X			
Fuel Cell Tanks		X	X			X				X	X		
Propellant Management Device		X	X			X				X	X	X	
Refueling Interfaces		X				X				X			
Structures													
Pressure Vessel		X	X		X					X	X		X
Internal Structure		X			X					X			
Rad/Debris Shield		X			X		X			X			X
Common Berthing Adapter		X			X					X			
Intermodular Interface		X			X					X			
Remote Manipulator System		X	X	X	X		X	X		X	X	X	X
Primary/Truss		X			X					X			
Internal		X			X					X			
Thermal													
Heat Pipes		X	X			X				X	X		
Radiators		X				X				X			
Insulation		X				X				X			
Cold Plates		X				X				X			
Phase Change Device		X				X				X			
Heaters	X	X	X			X				X	X	X	
Louvres		X				X				X			
Louvre Mechanical System		X	X			X				X	X		
Coolant Pump	X	X	X			X				X	X	X	
Coolant Piping		X	X			X				X	X		
Air Recycling Unit		X	X			X				X	X	X	
Thermometer	X	X				X				X			

TABLE 10.4 *Component and Subsystem Acceptance Tests.*

Components/Subsystems	F	L	PS	AC	RV	TC	TV	BI	P
C^{3}T									
TDRSS Transponder									
EVA Transponder									
Power Amplifier	X		X		X	X		X	
Parabolic Antenna									
S-Band Omni Antenna									
EVA Omni Antenna									
Internal Communication Network	X		X		X	X		X	
Harness	X		X		X	X		X	
Processors									
Data Recording Unit	X	X	X		X	X		X	
ECLSS									
Oxygen Tank	X	X							X
Extravehicular Mobility Unit	X	X	X	X			X	X	X
Extra EMU Batteries									
CO2 Removal System	X	X			X		X		X
Atmosphere Control System	X	X			X		X	X	X
Cabin Air Storage System	X	X							X
Pressure Suit	X	X			X		X		X
Waste Disposal System	X	X			X		X		
Food, Water, and Containment System	X	X			X		X		X
Emergency System	X	X	X		X		X	X	X
Human Interface Controls	X	X			X		X		
Lighting									
Restraints, Seats									
Cabin Fans	X				X		X		
Portable O2 System	X	X							X
Fire Detection System	X	X	X		X		X	X	X
Fire Extinguisher	X	X							X
GNC									
DCS with Sun Fans									
Rendezvous Radar Transponder	X	X	X		X	X		X	
MIMU									
MANS System									
MAPS									
Star Tracker									
LNS	X	X	X		X		X	X	
Docking Camera									

TABLE 10.4 *Component and Subsystem Acceptance Tests.*

Components/Subsystems	F	L	PS	AC	RV	TC	TV	BI	P
Power and Propulsion									
Propellant Tanks	X	X			X		X	X	
Plumbing - Valves and Pipes	X	X			X		X		X
RL10A-4 Engine	X			X			X	X	
Thermal Venting System	X	X			X		X		X
RCS Tanks	X	X							X
RCS Valves and Pipes	X	X			X		X		X
RCS Thrusters	X				X		X	X	
Fuel Cell Powerplant	X	X	X		X		X	X	X
Power Conditioner									
EPS Wire Harness	X				X		X		
Fuel Cell Tanks	X	X			X		X	X	X
Propellant Management Device	X	X			X		X	X	X
Refueling Interfaces	X				X		X		
Structures									
Pressure Vessel	X	X							X
Internal Structure	X			X			X		
Rad/Debris Shield	X			X			X		
Common Berthing Adapter	X			X			X		
Intermodular Interface	X			X			X		
Remote Manipulator System	X		X	X			X	X	
Primary/Truss	X			X			X		
Internal	X			X			X		
Thermal									
Heat Pipes	X	X			X		X		X
Radiators	X				X		X		
Insulation	X				X		X		
Cold Plates	X				X		X		
Phase Change Devices	X				X		X		
Heaters									
Louvres	X				X		X		
Louvre Mechanical System	X				X		X		
Coolant Pump									
Coolant Piping	X	X			X		X		X
Air Recycling Unit	X	X			X		X	X	X
Thermometer									

The system tests are limited due to the difficulty of demonstrating the performance of the entire system on the ground, and due to the risk of decreasing the system lifetime through rigorous testing. Project Freebird will go through a series of six different environmental tests at the system level. These are pyro shock, acoustic, thermal vacuum, thermal balance, burst pressure, and electromagnetic compatibility. In addition, it will go through thorough functional tests at various stages during the integration and environmental acceptance process. Section 10.2.4 describes the sequences of these tests.

10.2.4 Sequential Testing

Figure 10.2 shows the component and subsystem level qualification and acceptance test sequence. The protoflight sequence for a certain component or subsystem will follow the qualification sequence. The tests that each of the components and subsystems must undergo are shown in Table 10.3 and Table 10.4. The tests not designated for a specific article will be omitted from the sequence. The test acronyms are described in Section 10.2.1 on page 200.

Qualification
Test Sequence: F -> L -> F -> PS -> F -> L -> AC or RV -> L -> F -> SV -> F -> A -> F -> L -> TC - -> L -> F -> L -> TV -> L -> F -> P -> L -> F -> EMC -> F -> LC -> F

Acceptance
Test Sequence: F -> L -> F -> PS -> F -> L -> AC or RV -> L -> F -> L -> TC -> L -> F -> L -> TV - -> L -> F -> BI -> F -> P -> L -> F

FIGURE 10.2 *Component and Subsystem Test Sequence.*

The test sequence for the integration and system level tests has been carefully designed. The full sequence of the environmental tests is shown in Figure 10.3. There are three major phases of functional tests at the integration and systems level. The first requires functional tests to be conducted on individual components and subsystems as they are integrated into the system. All failures at this stage should be corrected and retested before integrating the article with the entire system.

Next there will be combined system tests which ensure that all of the subsystems function together properly. The first set of data from these tests will be used to establish a baseline for comparison with all further functional tests. A functional test will occur before and immediately after each environmental test at the integrated system level.

Following all of the environmental tests and as close to launch as possible will be the integrated systems test. This verifies the functionality of the integrated system following all of the environmental tests and checks the operational readiness of the system. All of the operational modes, in addition to subsystem performance, will be demonstrated during this test.

In addition, there will be a pre-launch system validation. It will be performed at the launch site and will duplicate the system protoflight tests as closely as possible. This verifies that the transportation and handling have not degraded the vehicle's characteristics and is extremely important due to the large shipping distances this system will undergo. Finally, there are the pre-launch tests which establish that the interfaces between the system and the launch vehicle are compatible and are within specifications.

Integration and System Level

Environmental Test Sequence: F -> PS -> F -> AC -> F -> TV -> F -> TB -> F -> P -> F -> EMC -> F

FIGURE 10.3 *System Environmental Test Sequence.*

10.2.5 Verification Facilities

There will be two major facilities used in the Project Freebird test program. The Jet Propulsion Laboratory (JPL) in Pasadena, California will be used to complete the environmental tests at the component and subsystem level. The experience of JPL, along with the wide range of environmental testing facilities available there, make it the best choice for these tests.

The Space Systems Automated Integration and Assembly Facility at the Kennedy Space Center will be used for integration and system level environmental tests. This facility, designed to integrate and test the space station, has the tools necessary to qualify a large manned vehicle. As a back up facility, the Johnson Space Flight Center will be used.

10.3 In-Flight Testing

Once delivered to orbit, in-flight tests will be conducted to evaluate the vehicle in its performance atmosphere. As the final stage of testing, in-flight tests are vital as they will test the vehicle's reaction to the space atmosphere. This ensure proper performance during future mission scenarios. All components and subsystems will be installed and subjected to in-flight testing to ensure that the complete, integrated vehicle will operate safely and satisfactorily.

In-flight tests differ from ground tests in their scope and intensity. While ground testing is in-depth and repetitive, in-flight testing checks the basic system functions needed for the mission requirement to ensure that the subsystems are operating properly.

10.3.1 In-Flight Testing Philosophy

In-flight tests are performed to check for damages incurred from launch loads and procedures. All vehicle components and subsystems must survive in-flight testing to prove their mission-worthiness. In-flight tests also evaluate the vehicle to assure that all high-level requirements have been achieved during design and production. High-level requirements that are not ment may necessitate alternate mission scenarios or, in the worst case, a return to the design and/or production stages. The final driver behind in-flight testing is the vehicle's exposure to its mission atmosphere - namely space. These tests will study the vehicle's reaction to its initial exposure to the space atmosphere and record any complications that were not encountered during ground testing.

10.3.2 Criteria for Success and Failure

The in-flight test results will be examined and compared to established requirements. These requirements set the success standards for each subsystem. Different requirements are fixed for each of the different vehicle subsystems.

Those subsystem test results which lie within their required standards will have passed the in-flight tests and will be considered successful. These subsystems will then be labeled flight-worthy and mission-ready.

Those subsystem test results which lie outside of their required standards will have failed in-flight testing. These subsystems must then be modified or repaired at the component level while on-orbit or, in the worst case scenario, be returned to Earth and to the design and/or production stages.

Once integrated, the vehicle, with all components and subsystems in place, will be tested and subjected to a set of performance requirements. If the vehicle's test results fall within these requirement standards, the vehicle will be considered safe and ready for its initial mission. If the results fail to meet these requirements, the vehicle will be modified or repaired until the test results are successful. The worst case would consist of the vehicle continuing to fail these in-flight tests and, therefore, it would have to be disassembled and return to Earth for modifications and further ground testing

The success and failure criterion for each subsystem and for the entire vehicle are based upon each unit's criticality to the safety and success of the mission. Subsystems which are vital to the survival of the crew are given strict test requirements: the subsystem must operate properly the first time it is activated. Subsystems which are important for the mission objective but whose failure would not be life threatening for the crew are given two chances to pass the in-flight test.

- Power & Propulsion Systems: Due to the importance of the vehicle's power systems and engines during all missions scenarios, this subsystem must run successfully on its first testing attempt - the system power-up and engine firing must work properly the first time they are activated.
- Environmental Control & Life Support Systems: Since this subsystem is responsible for guaranteeing the lives and safety of the astronauts, it is required to function satisfactorily on the first testing attempt. If the readings taken during in-flight testing do not meet safety standards, the ECLSS subsystem will not be given a second chance. It will be repaired or, in the worst case, returned to Earth.
- Command, Communications, Control, & Telemetry Systems: Because many failures of this subsystem are considered to be life threatening, this subsystem is given one chance to pass its in-flight test. If the subsystem fails to function properly during its first in-flight run, it is considered un-flight worthy and must be for repaired and/or replaced.
- Structure System: The vehicle's structural integrity is vital to every possible mission scenario - it's failure mode presents the most mission critical and life threatening problem for the crew. For this reason, the structural body of the vehicle must maintain the highest level of safety. The structures subsystem must pass all first-run inspections before the vehicle will be deemed mission-ready. If the structural integrity of the vehicle is at all questionable, the vehicle will remain at the Space Station for repair or be returned to Earth. The Remote Manipulator System (RMS) arms will be given two chances to function successfully.
- Thermal System: The thermal subsystem must pass it's in-flight test on the first attempt as it is a vital factor in the survival of the crew. If the readings taken during in-flight tests fail to fall within safe parameters, the subsystem must be repaired or replaced.
- Guidance, Navigation, & Control Systems: While the GNC subsystem is important to the successful completion of a mission, it is not considered life-threatening when it enters a failure mode. Therefore, during in-flight testing, it will be given two attempts to meet it's in-flight testing requirements. If the

subsystem fails on the first run, it will be given a second chance before it is returned for repair or replacement.

The in-flight testing requirements for each subsystem are summarized in Table 10.5.

TABLE 10.5 *Criteria for Success and Failure.*

Subsystem	In-Flight Testing Requirements
P&P	One engine firing in two attempts; one power-up in one attempt.
ECLSS	One nominal life support reading in one attempt.
C^3T	One communication and telemetry check in one attempt.
STRUCTURES	Interface verification and visual inspection; one RMS procedure in two attempts.
THERMAL	One nominal radiation and temperature reading in one attempt.
GNC	One attitude adjustment and check in two attempts; one RCS firing in two attempts.

An in-depth discussion of the in-flight test procedures as well as a schedule for in-flight testing may be found in Section 10.3.3.

10.3.3 In-Flight Testing Procedure & Schedule

A testing sequence flow-down and schedule have been developed to ensure that the in-flight tests are conducted efficiently. If closely followed, the schedule will minimize the time and cost of the in-flight tests.

The testing sequence will begin when the vehicle's subsystems are delivered to orbit. Once separated from the launch vehicle, the first stage of the sequence will begin with the vehicle assembly. Assembly will consist of connecting the base and crew modules and verifying that their interfaces are secure. Next, the subsystem tests will be performed and the results evaluated according to the success and failure criteria. The Integrated Vehicle Test will then be executed which consists of running all of the subsystems concurrently. Once the vehicle has successfully passed all of these tests it will be considered flight-worthy and mission-ready. Figure 10.4 shows a diagram of the testing sequence flow-down.

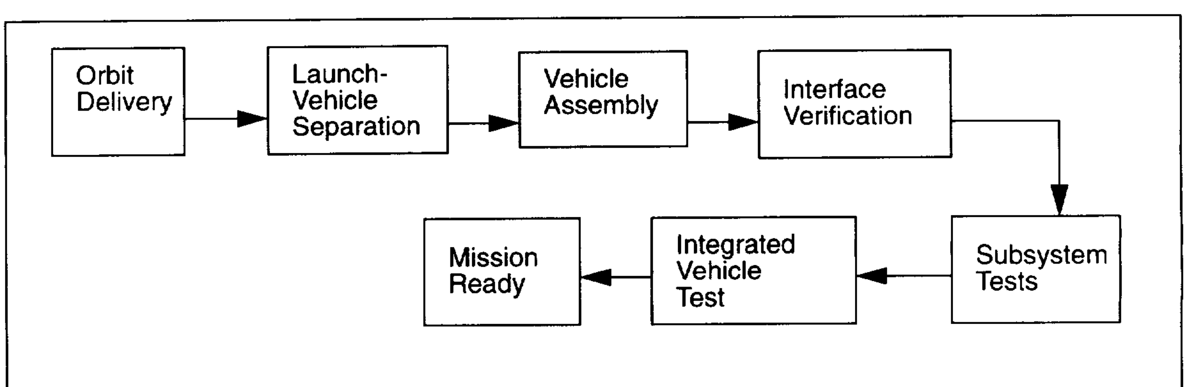

FIGURE 10.4 *Testing Sequence Flow-Down.*

A testing procedure and schedule protocol was developed to ensure that the vehicle will be mission-ready for the delivery date specified in the high-level requirements. Ample time has been allotted for each step of the testing sequence to allow for unforeseen complications or failures as well as for any subsystem calibrations that need to be performed. Table 10.6 summarizes the in-flight testing procedures for all of the vehicle's subsystems.

The in-flight tests required to ensure the safe and successful operation of each subsystem are explained below.

- Power & Propulsion Systems: The vehicle's systems will be powered-up and the main engines as well as the RCS thrusters will be fired to verify that this subsystem functions safely and properly.
- Environmental Control and Life Support Systems: Temperature, pressure, and radiation level readings within the crew cabin will be monitored and evaluated to verify that the subsystem is operating within safe parameters. Gas mixture readings will also be taken to confirm that the air in the crew module is in fact breathable. The crew module will then undergo the de-pressurization and re-pressurization protocol which will be performed during future missions when astronauts exit the vehicle for EVA work. This aspect of the subsystem's in-flight testing will evaluate its pressurization procedures as well as verify that all other subsystems can function in the de-pressurized, vacuum-like atmosphere.
- Command, Communications, Control, &Telemetry Systems: The in-flight test of this subsystem consists of an audio, visual, and command communications check with Ground Control and the Space Station as well as telemetry verification of all vehicle systems that require monitoring.
- Structure System: The vehicle's structure will be visually inspected during on-orbit assembly to check for any cracks or breaks that might have been incurred during launch or launch vehicle separation. During actual in-flight testing, the subsystem interfaces will be inspected to verify that they are intact and fully functioning, and the vehicle will be visually inspected once again to verify it's integrity. The RMS arms will be activated and required to manipulate a large satellite-like object to verify that they function properly.
- Thermal System: Temperature and radiation level readings will be taken and evaluated during in-flight testing to ensure that they are within safe parameters and provide no possible threat to the vehicle's crew.
- Guidance, Navigation, & Control Systems: An attitude adjustment will be made by the GNC subsystem and checked to ensure that it was executed successfully. The RCS thrusters will also be fired to test the vehicle's maneuvering capabilities.

TABLE 10.6 *Subsystem In-Flight Testing Procedures*

Subsystem	In-Flight Testing Procedure
P&P	System power-up; main engine & RCS thruster firings
ECLSS	Temperature, pressure, radiation, &gas mix. readings; de-pressurization sequence check
C^3T	Audio, visual, & command communications check; telemetry check
STRUCTURES	Visual inspection; interface verification; RMS manipulation
THERMAL	Temperature & radiation level check
GNC	Attitude adjustment and check; RCS maneuver and check

Once the individual subsystem in-flight tests have been performed, the Integrated Vehicle Test will be conducted. During most mission scenarios, several, if not all, of the subsystems will be operating at once, thus the Integrated Vehicle Test ensures that this multi-operational mode is possible and successful.

The in-flight testing schedule begins with on-orbit delivery and launch-vehicle separation. The next stage involves vehicle assembly - attaching the two modules - and the verification of the interfaces between each of the subsystems. During this stage, all parts of the vehicle will also be visually inspected to check for structural integrity and any damage that could have been incurred during launch procedures. The next few steps of the in-flight testing schedule involve the subsystem tests. The Structures and P&P subsystems will be tested first to verify the safety of the vehicle for astronaut personnel and to provide the power necessary to test the other subsystems. The ECLSS and Thermal subsystems will be tested next to evaluate the safety of the crew module. Finally, the C^3T and GNC subsystems will be tested to check all vehicle command and control functions. A summary of the in-flight testing schedule may be found in the Table 10.7.

TABLE 10.7 *In-Flight Testing Schedule.*

Stage	Test/Procedure
Stage 1	On-Orbit Delivery and Launch-vehicle Separation
Stage 2	Vehicle Assembly and Interface Verification
Stage 3	STRUCTURES and P&P Subsystem Tests
Stage 4	ECLSS and THERMAL Subsystem Tests
Stage 5	C^3T and GNC Subsystem Tests
Stage 6	Integrated Vehicle Test
Stage 7	System Calibration and Vehicle Storage Facility Docking

After the testing sequence and schedule has been completed the vehicle will be ready for its first mission. The Integrated Vehicle Test should be performed each time the vehicle is preparing to embark on a mission to verify that all subsystems are up and running properly.

10.4 Verification and Validation Summary

The verification and validation program that is described above has been designed to allow for a reliability of .9944 for Project Freebird. Through innovative management and testing approaches, the system has achieved a high level of reliability while keeping the costs to a minimum. Logical testing sequences, rigorous interface validation, and flexible management are all important elements of this efficient program. Overall, the test procedures allow the system to meet its ten year lifetime at a low cost while operating through a variety of mission scenerios.

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Chapter 11

Risk and Reliability

Risk and reliability considerations have been incorporated into the design of Freebird. For any space based project, a certain amount of risk must be accepted. This chapter contains a basic analysis of Freebird's reliability. In addition, the testing program outlined in Part II Chapter 10 has been designed to increase the confidence and accuracy of this reliability estimate.

11.1 System Reliability

Overall system reliability is the main driver that determines system parameters. Reliability can be significantly increased by minimizing single point failures. This is most easily done using redundant systems. However, Freebird needs to have a low mass due to the high fuel to structural mass ratio needed for operation in polar and geosynchronous (GEO) orbits. For this reason, high component reliability was emphasized. Only the most critical components have redundant backups. Critical components are those whose failures are either life or mission critical. Individual component risk levels can be found in Appendix C on page 309.

11.1.1 Recommended System Reliability

The recommendation for overall system reliability was .9944, implying 9944 successful ten year lifetimes out of 10,000 vehicles built. The NASA standard for crewed missions is .994. While all systems are included in the calculation of the overall system reliability, launch reliability of the vehicle is not included in this value. The reliability numbers are based on a ten year vehicle lifetime with twelve missions per year.

Subsystem reliability recommendations were determined by giving equal weight to all subsystems. This produces a uniform subsystem reliability of .999 by taking the sixth root of .994. However, the Verification and Validation group determined that Environmental Control and Life Support System (ECLSS) should provide a higher reliability than .999. Therefore, ECLSS was assigned a reliability of .9994. This higher number was used to take up any inadequate reliability from other groups. The recommended numbers for subsystem reliability are shown in Figure 11.1 .

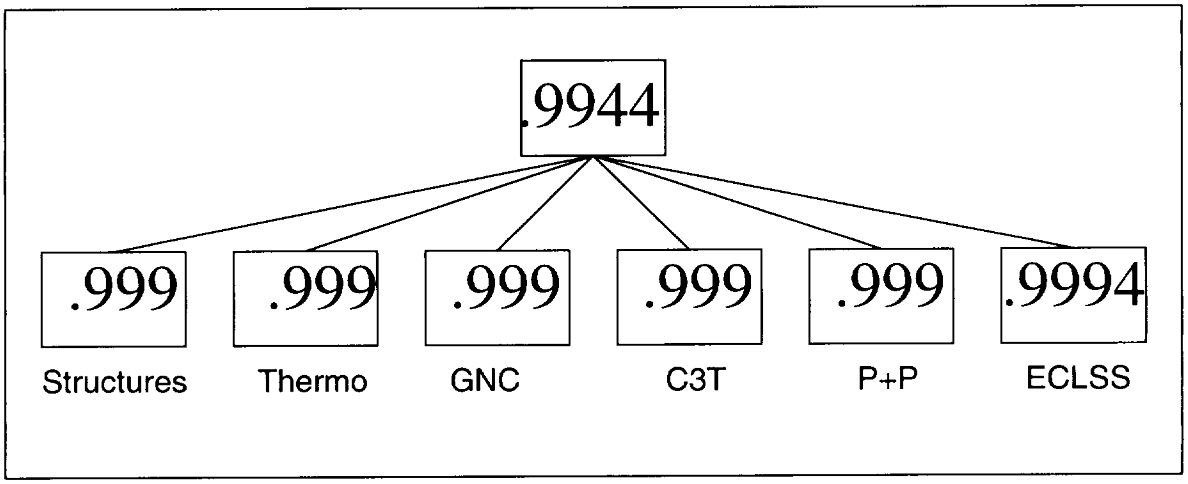

FIGURE 11.1 *System Reliability Goals.*

11.1.2 Actual System Reliability

The actual overall system reliability is .9921, which is the best system reliability achieved in the design process. Overall system reliability breakdown is shown in Figure 11.2 .

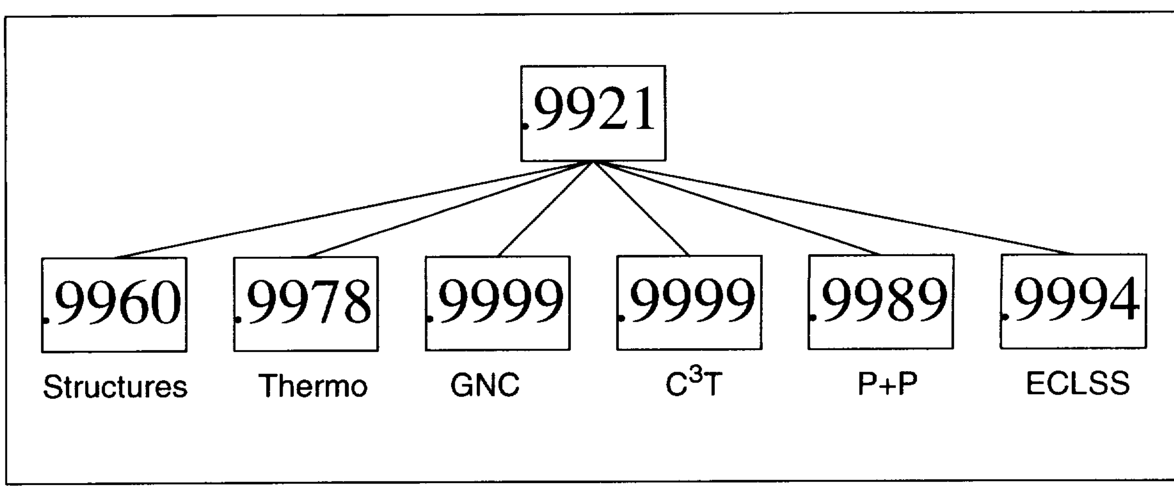

FIGURE 11.2 *Actual System Reliability.*

11.1.3 Iteration Process

The iterative process for determining overall system reliability involved the calculation of reliability from the bottom up. This means starting at the component level, using component reliabilities. The calculated number was then checked against the recommended overall system reliability. If there was a difference between the two numbers, an effort was made to improve subsystem reliabilities to better meet the goal. Where subsystems were lacking in reliability, an effort was made to provide redundant systems to increase reliability. A flowdown of this iterative process is shown in Figure 11.3 . Calculation of reliability is based on the probability of failure. For systems in series, the reliability numbers of each component are multiplied to achieve final reliability:

$$R_T = R_1 R_2 \ldots R_n \qquad \textbf{(Equation 11.7)}$$

where R_T is total reliability, and R_1, R_2 etc... are component reliabilities. For parallel systems and redundancy, the following equation is used:

$$R_T = R_1 + R_2 - R_1 R_2 \qquad \textbf{(Equation 11.8)}$$

If there are more than two parallel components, the equation is used multiple times. This is for single redundancy when the backup system is not used until the primary system fails. In the cases were the subsystem did not meet its reliability goal, redundancies were examined. However, as has been prevalent throughout this section, vehicle mass turned out to be a greater design driver.

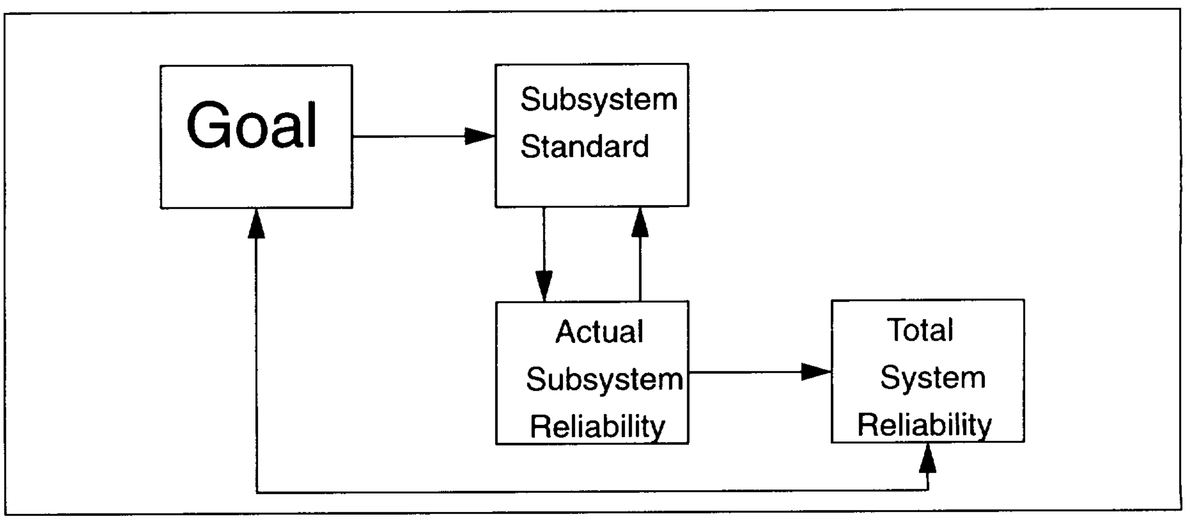

FIGURE 11.3 *Reliability Iteration Process.*

11.2 Subsystem Acceptance

Subsystem acceptance was based on how closely the subsystem met its recommended reliability. Acceptance implies that the subsystem made the best trade possible to achieve the best reliability could. Redundancy was a focus in cases where the reliability was low or the component was critical.

11.2.1 Subsystem Reliabilities

Subsystem reliabilities were determined from numbers given for factory produced components and numbers determined by an analogy to similar components. Components for which reliability numbers were not available were given a reliability of .999. This was done primarily for newer technology. Testing procedures, and design parameters can be constructed which will give a reliability of .999. This is one of the goals of the testing period of the project.

Only components with a risk level of 1 or 2 were used in the calculations. Lower risk components were not determined to be mission critical, and were not used. Figure 11.4 through Figure 11.10 on page 221 show the flowdown of the critical components of the seven subsystems. Where applicable, the components are in functional order from left to right. Shaded blocks imply complete component redundancy. Shaded corners imply internal redundancy. The reliability number in each of these blocks is the reliability after redundancy calculations are applied. Blocks in parallel imply functional redundancy. A failure of one of the parallel branches does not imply mission failure, simply a compromised performance.

Figure 11.4 shows the Thermal reliability flowdown. There are two systems in the Thermal group, one of which is active, and the other passive. The passive system is highly reliable since it has no moving parts. The heaters and the pump are both redundant. This was necessary since they both have moving parts. The louvres have moving parts, but have been found to be highly reliable.

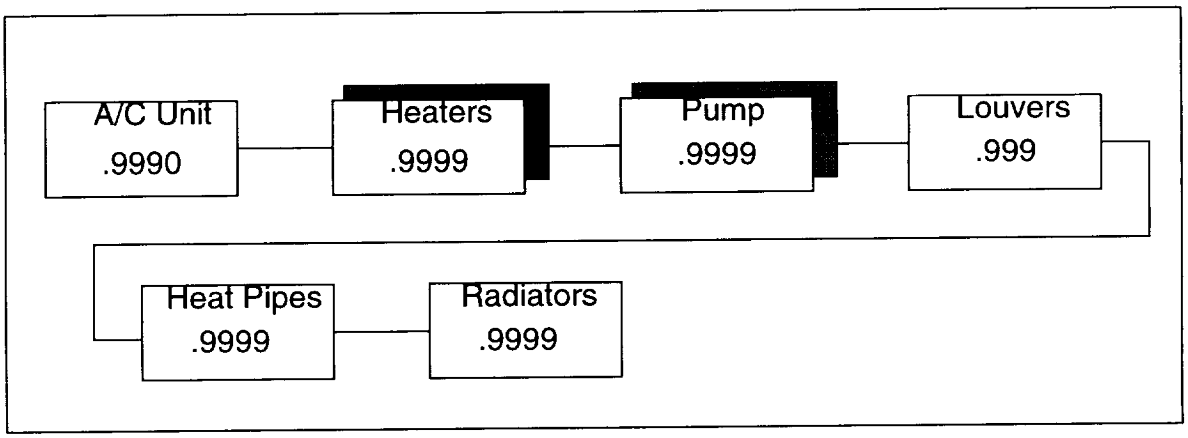

FIGURE 11.4 *Thermal Control System Reliability Flowdown.*

Figure 11.5 shows the Structures reliability flowdown. There is no particular functional order to the Structures flowdown. The "Structure" box represents all the support structure of the vehicle. "Shielding" implies radiation as well as debris shielding. The two arms are not redundant since both are used simultaneously. However, in the event of a failure of one of the arms, most missions can still be carried out. Therefore, the failure of an arm is not mission critical.

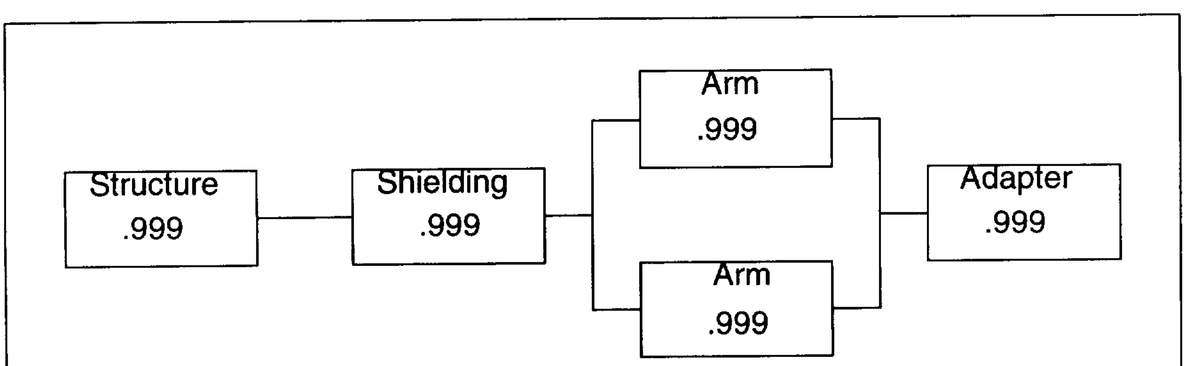

FIGURE 11.5 *Structures, Materials, and Mechanisms System Reliability Flowdown.*

Figure 11.6 Shows the Guidance, Navigation, and Control (GNC) reliability flowdown. Most systems in GNC are redundant due to the fact that they are highly mission critical. Ground tracking is used for the start up of missions. This is not likely to fail since it simply supplies location information on the asset to be tended. For LEO and GEO missions, the star sensor is used for tracking. The ground tracking radar can be used to navigate back to LEO from GEO in the event that the Star Tracker fails.

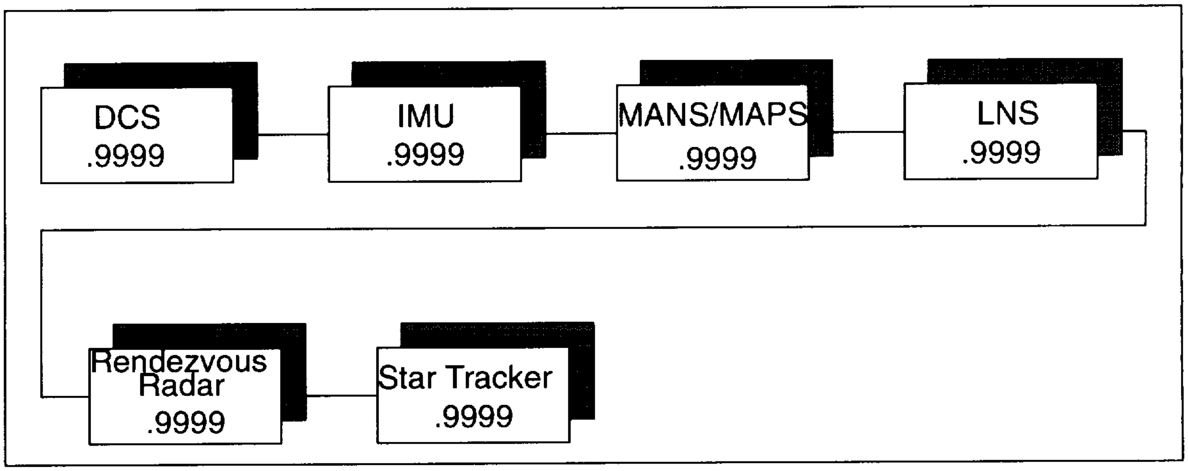

FIGURE 11.6 *Guidance, Navigation, and Control System Reliability Flowdown.*

Figure 11.7 shows the reliability flowdown for the Command Communications Control and Telemetry (C^3T) system. Like the GNC system, all of the systems are highly critical, and complete redundancy is incorporated into all the critical components. There are three CPU's with each one coming on line in the event of failure of the previous CPU. This is due to the fact that all functions of the system are directly or indirectly tied to the CPU. The CPU's reliability is 1.000 due to the fact that the significant figures of it's reliability round to that value.

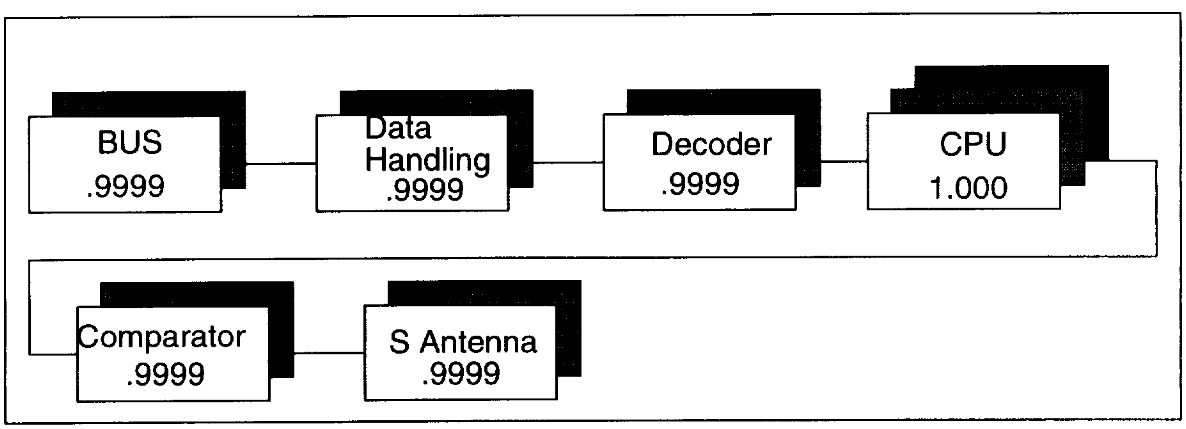

FIGURE 11.7 *Command, Communications, Control and Telemetry System Reliability Flowdown.*

Figure 11.8 shows the flowdown of the Power system. The Power system is linear in nature. Due to this fact, all components are redundant.

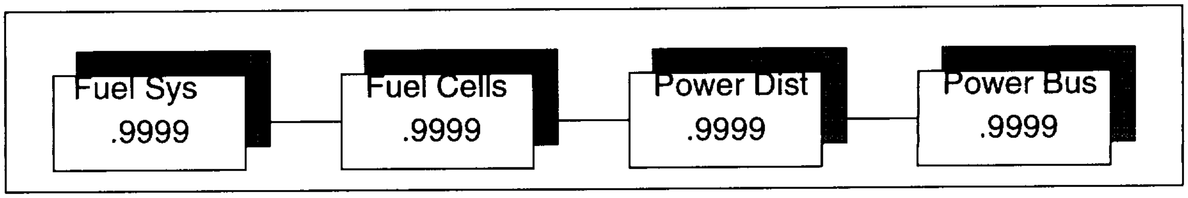

FIGURE 11.8 *Power System Reliability Flowdown.*

Figure 11.9 shows the Propulsion flowdown. There are four main engines, but the system can function with only two engines if necessary. During normal operation all four engines are used. There are two centralized Reaction Control System (RCS) tank systems. Both are connected to each other, allowing fuel to flow between the two. In the case of a failed tank, the connection is closed, and the RCS operates off the one remaining tank. The RCS thrusters are redundant so that if one set fails, there are RCS thrusters in other positions which can continue to maneuver the craft.

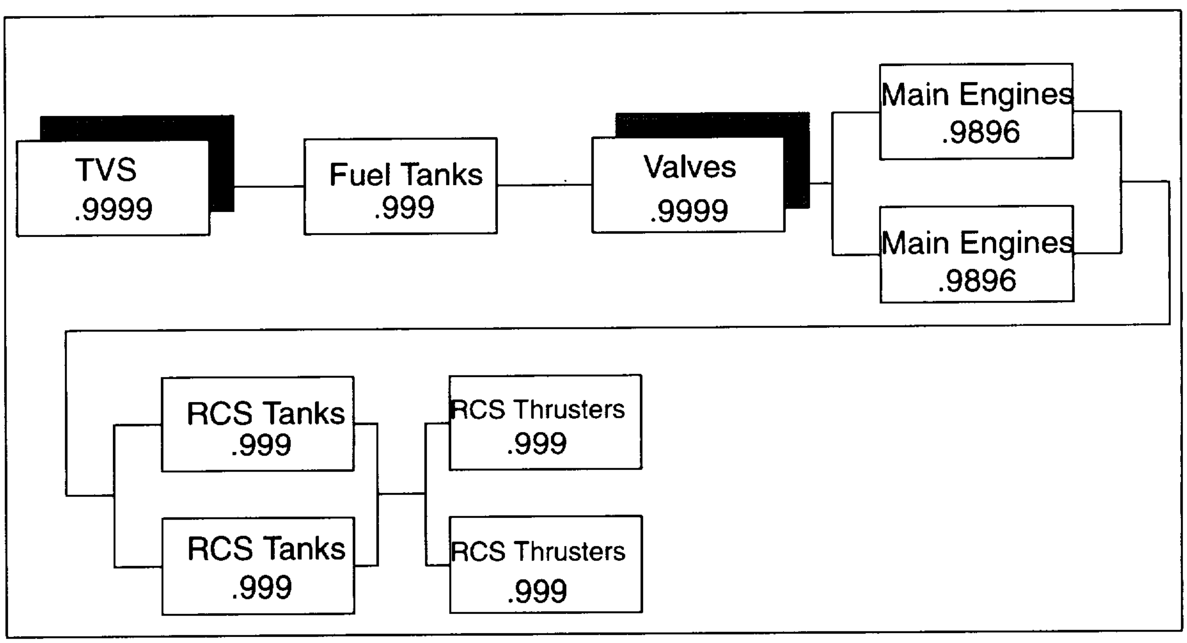

FIGURE 11.9 *Propulsion System Reliability Flowdown.*

Figure 11.10 shows the ECLSS reliability flowdown. The fire suppression system, as well as the atmosphere control, Extra Vehicular Activity (EVA), and Controls and Display systems are all internally redundant. However, since they deal with many sensors and controls, they are redundant until all the sensors or controls have failed. Failure of one of the components decreases the functionality of the overall system, but the system will still function within acceptable parameters.

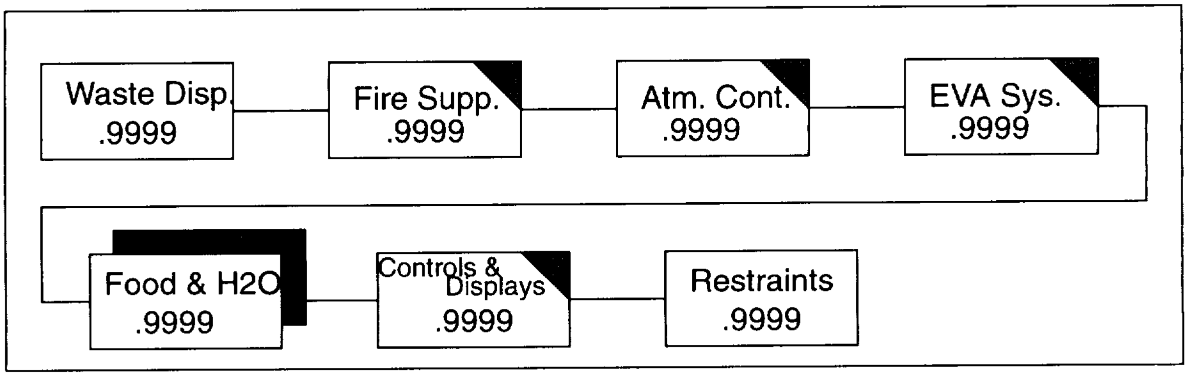

FIGURE 11.10 *Environmental Control and Life Support System Reliability Flowdown.*

11.2.2 Risk Trades

Risk trades were made at the subsystem level to increase reliability. Emphasis was placed on redundancy for high risk and critical components. Wherever possible, other systems were configured such that they could perform functions of a failed system, though at a compromised level. When making risk trades, a main driver was the overall cost of the system. Low mass in turn provides low cost, and was the main cost indicator. In most cases, a trade was made between increasing mass and reliability. When the increase in reliability no longer justified the increase in weight, the trade was complete. A flow chart of the decision process is below in Figure 11.11 .

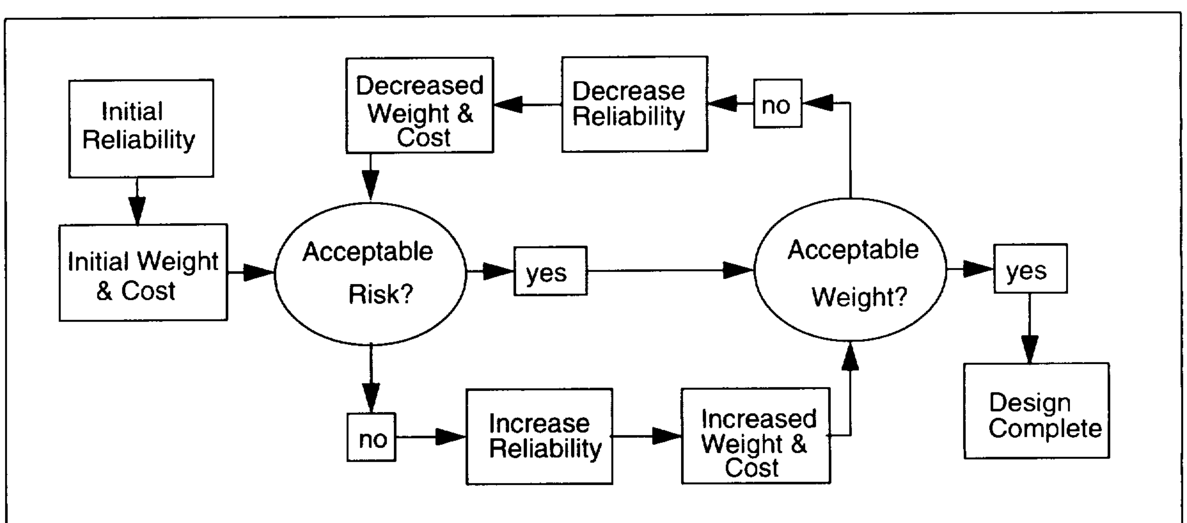

FIGURE 11.11 *Reliability Versus Weight Decision Process (Iteration).*

11.3 Technology Risk Management

Technology Risk Management, or TRM, is a method of reducing the risk, cost, and time of technology development for Project Freebird. Whenever possible, subsystems are comprised of proven technologies. Undeveloped or partially developed technologies must be flight-qualified in time for integration. The procedure for TRM is to identify immature technologies, plan for alternatives in case the new technology cannot be qualified, and develop an appropriate schedule to meet the delivery date. This schedule must include both the primary and alternative technologies.

11.3.1 Technology Maturity

The technology maturity level is critical in determining how and when a specific component should be incorporated into the Freebird schedule. A more developed technology may be purchased directly from a supplier, whereas a more immature item may need to be designed, analyzed, and tested early in the vehicle development stage. The level of technology development can be estimated using Table 11.1. Each subsystem component is listed in Appendix C on page 309 with its respective technology development level.

TABLE 11.1 *Technology Development Level* [1].

Readiness Level	Definition	Risk Level
1	Basic principles observed	High
2	Conceptual design formulated	High
3	Conceptual design tested analytically or experimentally	Moderate
4	Critical function/characteristic demonstrated	Moderate
5	Component or breadboard tested in relevant environment	Moderate
6	Prototype/engineering model tested in relevant environment	Low
7	Engineering model tested in space	Low
8	Full operational capability	Low

11.3.2 Immature Technology and Alternatives

New technologies with relatively little development or testing must be addressed first. These include any subsystem components with moderate to high levels of development risk as defined in Table 11.1. Components in this group must be developed and tested well before integration to allow time to switch to alternatives if necessary. This leaves time for development of backup technologies if a component cannot be completed for some reason. The critical components of Freebird which must be developed first, the alternative technologies, and the impact on Freebird are described below.

- The vehicle's propulsion system is designed to include four RL10-A4 engines, made by Pratt and Whitney, which are flight-qualified for 100 starts and 5.25 hours of burn time. Currently, these engines are only qualified for only 20 starts and 3000 seconds burn time. By using an advanced engine the number of necessary engine replacements can be greatly reduced. See Part II Section 5.4.2 for further details. The advanced qualification process must be completed well before the engines are required for vehicle integration. The alternative to this technology is to use the proven RL10-A4 engines and accept the risk and cost of the increased number of replacements.

- The propulsion system fuel tanks are to be made of a composite material and lined with titanium. This technology has never been fully tested and flight-qualified. The design was chosen due to the composite's strength and mass properties, as well as limitations on tank mass. (See Part II Section 5.4.2.) The alternative to the composite tank material is stainless steel which is more commonly used and fully tested. The impact on Project Freebird is an increased tank mass.
- The Freebird power system includes fuel cells which are downsized from those used on the Space Shuttle (STS). The power requirements for Freebird are much less than those for STS, and thus redesigned fuel cells will provide significant mass savings (See Part II Section 4.3.4.). The alternative option is the STS fuel cells, with a reduced amount of fuel appropriate for Freebird's power needs.
- Freebird's radiation and debris shielding is a Mesh Double Bumper (MDB). For a complete discussion about MDB see Part II Section 3.4.1. This technology needs to be further developed, tested, and flight-qualified. If the MDB cannot be qualified, Whipple shielding, which is a proven protection method, will be used. The result of using this more mature technology would be a significant increase in mass.
- Project Freebird includes a Remote Manipulation System (RMS) comprised of two arms. These arms are a variation of the STS RMS now in use. Part II Section 3.6.1 gives specifications for the RMS. The alternative technology is the STS RMS. The impact of using this alternative is somewhat reduced mission performance.
- For Freebird's guidance, navigation, and control (GNC) a new type of control software is needed. Such a software, Microcosm's Autonomous Planning System (MAPS), is in the first stages of development. This technology is necessary for the Freebird missions and will be developed by Microcosm under contract, or alternatively, by the GNC system designers and engineers.
- Autonomous docking with Space Station Alpha will be accomplished using a Laser Navigation Sensor (LNS). This system is currently under development by Draper Laboratory. It is not fully tested or flight-qualified, but must be in time for subsystem and vehicle integration. If the LNS is not functional, Freebird is capable of manual and remote teleoperated docking.

11.3.3 Development Schedule

Once time estimates for each component development have been made, a complete schedule can be formulated. Figure 11.12 shows an example of a component development schedule employing TRM for the spacecraft RMS and clearly shows how the development times overlap. If the first technology meets its schedule, development of the alternative is not needed. Some technologies will be developed simultaneously to preserve the project schedule and to allow the opportunity for the immature technology to be completely developed within a certain time margin if possible.

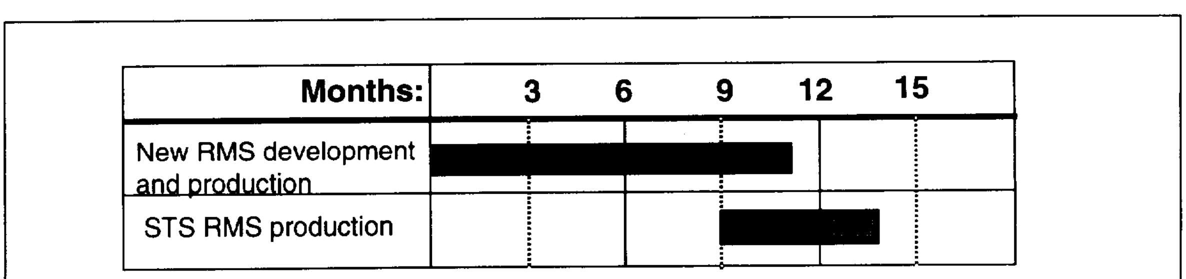

FIGURE 11.12 *Example Development Schedule.*

11.3.4 Cost implications

Although TRM may require simultaneous component development, ultimately the project schedule will be preserved and cost minimized. By providing alternative components during the design phase, a clear plan of action is prescribed when one technology will not meet qualification in time for integration. This saves money by not delaying delivery or compromising the Freebird objectives.

References

1. Larson, Wiley J. and James R. Wertz, eds. *Space Mission Analysis and Design*, Second Edition. Torrance, CA: Microcosm, Inc., 1992. Page 732.

Chapter 12

Cost and Producibility

Cost and Producibility are important factors for Project Freebird. In addition to setting an initial budget of approximately $1 billion, an attempt has been made to design the vehicle to maximize its ability to produce revenue from satellite servicing. It is hoped that much of Freebirds' operating costs can be offset by private sector utilization of the system.

The project costing was accomplished through two approaches. The first was a top down approach and the second was a bottom up component based approach. Both methods were employed to insure a good level of reliability in the cost area.

12.1 Work Breakdown Structure

The Work Breakdown Structure (WBS) can be broken down on the program level and the system level. It is a tool for identifying the multiple systems and subsystems which are part of Project Freebird. Each group has its own responsibilities and contributes to both the design and the cost analysis of the system.

12.1.1 Program Level Work Breakdown Structure

The program level WBS describes the structure of the entire design division. It includes Program Management and Program Support during the Research Design Testing & Evaluation (RDT&E), Production and Operations phases. The program level WBS for Project Freebird (Figure 12.1) contains the Vehicle Design as a system level subgroup. Very often the cost of the project is not only driven by the vehicle design but by other factors as well. These factors appear as overhead and can be an appreciable fraction of the final vehicle cost.

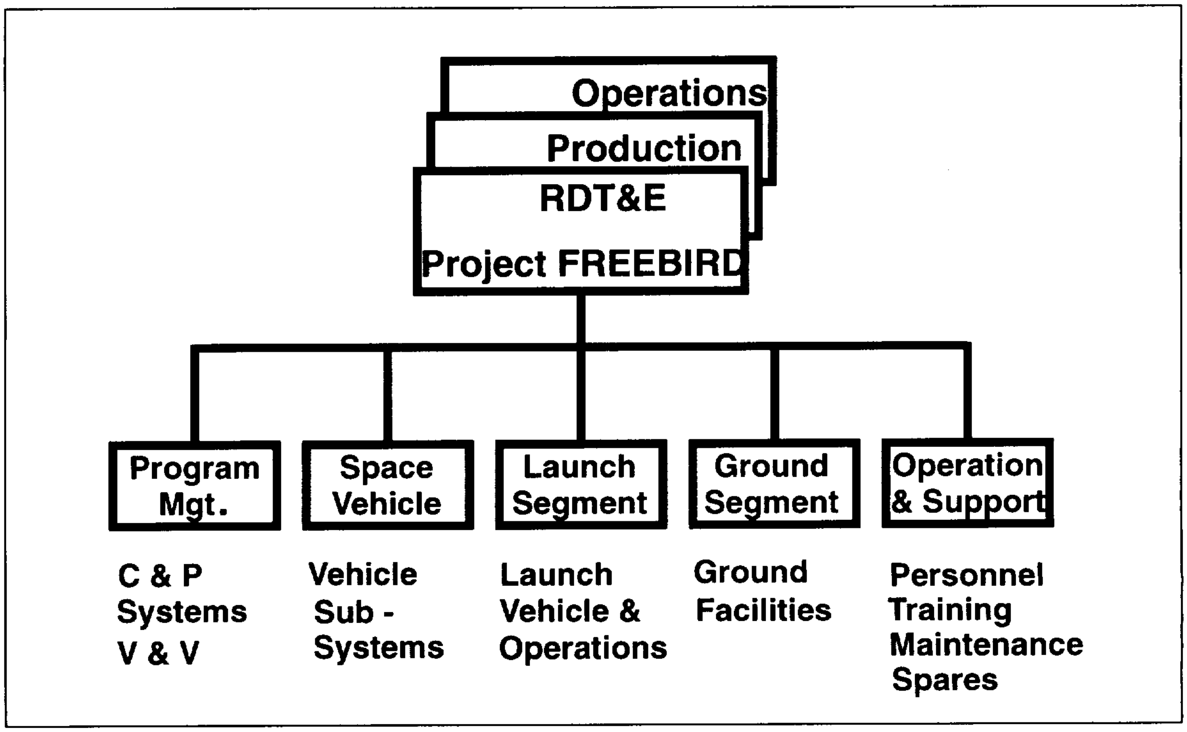

FIGURE 12.1 *Program Level Work Breakdown Structure.*

Cost analysis on the program level is primarily a function of overall program complexity. Larger, more involved system designs require a greater level of design support. Additional cost-drivers can be identified for each system level group (Table 12.1). These cost drivers have been identified in an attempt to both monitor and model the final program cost.

TABLE 12.1 *Program Level Cost Drivers.*

Project System	Cost Drivers
Program Management	Overall Project Size, Project Structure, & Overhead
Space Vehicle	Cost Drivers for Individual Sub-Systems
Launch Segment	Space Vehicle Size & Weight, & Launch Vehicle
Ground Segment	Support Software
Operation & Support	Delivery Costs to Orbit, Refueling, & Maintenance

12.1.2 System Level WBS

The system level WBS includes the subsystems and subsystem elements which comprise the Space Vehicle itself. The system level WBS for Project Freebird (Figure 12.2) breaks the design of the vehicle down into six major subsystems: Structures and Mechanisms; Thermal control; Environmental Control and Life Support Systems (ECLSS); Guidance, Navigation, and Control (GNC); Communications, Command, Control, and Telemetry (C^3T); Power and Propulsion (P&P).

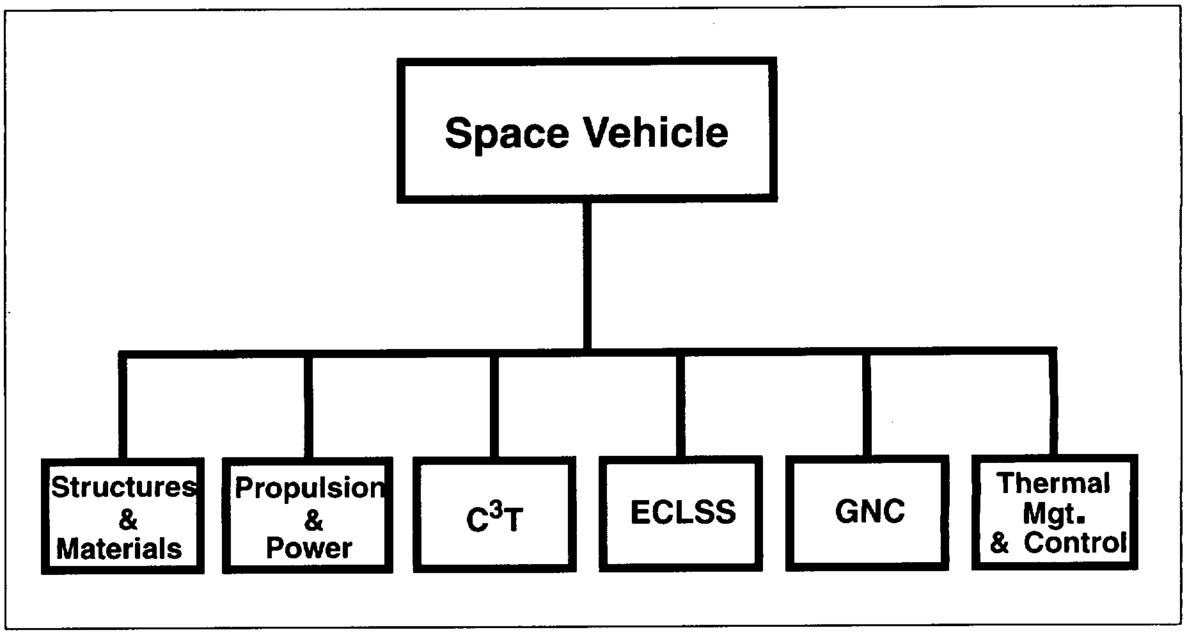

FIGURE 12.2 *System Level Work Breakdown Structure.*

Cost Analysis on the system level is primarily accomplished by a combination of top down and bottom up estimation. An analogy or costing model is established which allows a budget estimation. As the design becomes further refined, the budgeting is also adjusted. With design refinement a large degree of bottom up estimation becomes applicable. Direct quotes and Off-the-Shelf (OTS) prices are integrated into the overall cost model and the budgeting estimates are adjusted accordingly. To begin bottom up estimation the subsystem cost drivers were identified (Table 12.2).

TABLE 12.2 *Space Vehicle Design Cost Drivers.*

Vehicle Subsystem	Cost Drivers
Structures & Mechanisms	Remote Manipulating System & Loading
Thermal Control	Active Temperature Control for Cryogenic Fuel
ECLSS	Crew Size
GNC	Laser Navigation System & Navigation Software
C^3T	Closed Circuit System for Remote Operation
P & P	Main Engine

12.2 Cost Modelling and Determination

Cost modelling for project Freebird is accomplished through a combination of top-down and bottom up costing models. A top down estimate is produced as a guide based on previous space systems. This allows a budget cap be placed on the individual vehicle subsystems. For some program level costs (RDT&E, Program Management, Ground Operations) the top down estimation is considered highly dependable and remains the primary source of analysis. See Appendix D paragraph D.1.1 on page 313 for a more detailed description of top down estimation and a detailed accounting of the cost estimates produced.

Bottom Up cost modelling was accomplished as the design was refined through the use of direct quotes and a detailed subsystem level model produced by Lockheed. The Lockheed model uses system weight (or other variable dependent on the specific subsystem) as an estimate of hardware cost. Hardware cost is then used to determine design & development and prototype costs. Two alterations were made to this model. Direct quotes of hardware cost were used as inputs when available. And the practice of protoflighting components was permitted when feasible. Protoflighting components saves approximately 70% of original hardware cost from the final integrated component cost. See Appendix D paragraph D.1.2 on page 314 for a more detailed description of bottom up estimation and a detailed accounting of the cost estimates produced.

12.3 Subsystem Cost Trades

Structures and Mechanisms

The modelling of the structure subsystem costs is primarily accomplished through either direct manufacturer quotation or by modified analogy. Designer estimates of cost, complexity and newness of the technology have been combined into a set of ground rules used for costing. All cost figures are in FY94 dollars. Life cycle costs, maintenance costs, and ground support costs are not included in this section and are accounted for in total project cost.

The mission limitations impose cost trades in the areas of debris shielding, radiation shielding, overall vehicle weight, and the RMS system. Vehicle weight concerns are addressed through the use of composite structural members. These components have essentially been flight tested. Consequently, only a small complexity/new technology adjustment factor is added to the base estimates of cost. The savings in weight with the composite design is of such magnitude that considerable cost could have been endured before the choice was eliminated.

A new debris shielding system developed by the military for armored vehicles has been chosen by the Structures group. Despite the shield system's lack of flight testing, its mechanical simplicity and makeup are such that conclusions regarding its performance can be easily argued. Consequently, a small adjustment in cost for newness/complexity seems justified. Again, the weight savings were substantial.

Radiation shielding half thicknesses are determined by crew mission requirements and consequently the actual trades were determined by a direct material property comparison. Material selection has been reviewed from a cost approach and a final design utilizing aluminum has been chosen.

The RMS system selection has special problems related to vehicle length, mass/moment stability and mission requirements. A desire for a newly engineered system that is shorter and more flexible is estimated to be close to $200 million. Despite the relatively large cost in RMS redevelopment, most of the mission scenarios call for power and flexibility not currently available in the Shuttle RMS system. Additionally, proposed methods of vehicle self assembly and maintenance could not be achieved without the attributes of the new RMS specifications. Consequently, a new small and more powerful version of the RMS will be developed for Freebird. Despite the relatively large cost, savings in other areas will allow the new RMS system to be utilized without exceeding the subsystem budget.

The overall structure subsystems cost, as adjusted for all the discussed design trade-offs, can be found as part of Appendix D.4 on page 316 .

Thermal

For the thermal subsystem little information was available to begin an accurate bottom up cost estimate. Therefore the cost estimate was based primarily on a bottom up analogous estimation produced by Lockheed for NASA. The original estimate budgeted about $10 million for hardware, $10 million for subsystem integration, and $7 million for system RDT&E. This $27 million budget did not include final system integration costs. The actual total cost for the Thermal subsystem is presented in Appendix D.5 on page 317 .

The thermal subsystem does not contain a governing cost driver. The cost is driven by the nature of the heat distribution the system must satisfy. A significant cost trade involves the choice between active and passive thermal control. Because of the large cooling requirement of the cryogenic fuel active control is necessary. Also a new mechanical louver system will be installed in Freebird. Though the technology for this system is rather new, it utilizes current technology.

ECLSS

The modelling of the ECLSS subsystem cost is primarily accomplished through analogies or estimates given by experts in industry. A costing model (Appendix D.2 on page 314) supplied by Lockheed Corporation has been used as a guideline in determining the contribution of the ECLSS systems costs for the total project. ECLSS is one of the most expensive subsystems due primarily to the level of system redundancy and testing.

ECLSS cost drivers were primarily centered on crew size. Both repair and rescue configurations of the crew module, govern the dimensions of the ECLSS subsystem. The major cost driver is the Atmosphere Control System. The complexity, redundancy and testing needed for this part of the subsystem significantly outweigh other aspects of the system. A list of all of ECLSS components and their costs can be found in Appendix D.6 on page 318 .

GNC

Cost determinations for the Guidance, Navigation, and Control section is primarily completed through cost models provided by Lockheed. Few cost trades were made because of limited choices in technology. The altitudes being flown by Freebird and the level of autonomy prescribe much of the technology which is used. Because of these factors two relatively immature pieces of software were chosen; MANS and MAPS. These packages were chosen because they are being developed for commercial use, therefore costs are potentially lower. The costs for these new technologies were governed by the size of the computer code.

The major cost drivers for this subsystem are MAPS and the Laser Navigation System(LNS). The alternative to the LNS is the Image Processing System(IPS). IPS is still in early development stages and would therefore be potentially more expensive than LNS. MAPS software is a cost driver because it is also still in the early development stages. However, because this software is being developed for commercial applications, much of the cost may be absorbed by the time it is needed for Freebird. All other components in the GNC system are OTS technology and thus do not drive the cost for this system. Total component cost breakdown can be found in Appendix D.7 on page 319 .

C^3T

The level of technology required by C^3T is defined by the high orbits Freebird will be using. This causes difficulty in communication, particularly for video downlinks for autonomous missions. Because of these video requirements, more powerful transponders are required from that which is commonly used. The transponders are OTS technology, but they are still the most expensive components for the subsystem. The antennae are the only custom made components. Since antenna design is relatively mature, their cost is not expected to be a major driver for the subsystem. Total component cost breakdown can be found in appendix A, Appendix D.8 on page 319 .

Power

Cost determination for the hardware and design of the power production and distribution components was completed through the use of the costing models mentioned previously. A large portion of the hardware costs were obtained by using OTS prices. Primary cost trades pertaining to power hardware include means of power production (Solar vs. Fuel Cell) and power storage (Battery Type). Fuel Cells were chosen to satisfy mission goals at a cost of $5 million including RDT&E. Satisfaction of mission requirements and design cost considerations outweigh the initial cost of these systems due to the relatively small component costs relative to design costs. Final hardware costs are estimated to total approximately $10 million including RDT&E. Detailed cost estimates on the component level can be found in Appendix D on page 313 .

Further integration costs were obtained through analogy to previous NASA cost studies and labor considerations. A significant integration cost driver is the modular nature of Project Freebird; each module must interface with the vehicle's main power distribution network in a similar manner to the C^3T interfaces. Subsystem integration costs were estimated at $5 million. This number is based on component complexity, expected labor hours, and parametric estimation of integration costs.

Propulsion

Cost determination for the propulsion systems hardware was completed primarily through an approximate cost quote directly from Pratt and Whitney, the manufacturer of the RL10-A4. Additional propulsion system costs included fuel tanks and the Reaction Control System (RCS). These costs are determined through a combination of OTS prices and bottom up estimation. The primary cost trade for the propulsion system hardware is engine choice. This affects both delivery cost and operations and maintenance cost. An optimal engine satisfies mission parameters while being long-lived. Lifetime is more important than initial hardware cost due to the large cost of integration and delivery to orbit. To maximize engine lifetime, the engines must be requalified with an additional cost of approximately $40 million. This will extend engine lifetime to 100 starts, a 333% improvement. Total component costs for Propulsion are estimated at $95 million including RDT&E. Detailed cost estimates on the component level can be found in Appendix D on page 313 .

Subsystem integration costs for the engines are small due to the self contained nature of the engines upon delivery by the manufacturer. Integration costs for the rest of the subsystem are modeled to total approximately $10 million.

12.4 Design and Production Cost

Total design and production costs are estimated using analogy, primarily the OTV estimate from Lockheed (Appendix D.2 on page 314). These costs are a percentage of hardware costs, with contingency for complex systems. Contingency is also added for immature technology.

12.4.1 Total Hardware and Production Cost

Each subsystem component must be integrated into the vehicle upon delivery. These integration costs are quite high, often comprising approximately one half of the final hardware cost. Integration costs become a significant cost driver and must be considered during the design process. Therefore these costs are an important factor when attempting to model or determine the cost of a system.

Each subsystem must be considered separately since each has characteristics which make it unique. Some subsystems include system integration costs as part of their subsystem integration (eg. power, thermal, ECLSS). Other subsystems contain large, extremely heavy components which must be aligned very precisely (eg. propulsion, structure).

Cost models including estimation by analogy, complexity factors and labor considerations are applied to determine the approximate production and integration costs. System level integration costs are estimated to total $130 million including integration design.

12.4.2 RDT&E Support Cost

Support costs for RDT&E are the costs for verification and validation(V&V) of components and of systems, and the costs for research for immature technology. Past space projects have required an additional 40% of hardware costs for final verification and validation. V&V costs for Freebird are divided into three sections: component testing, integration testing, and final vehicle testing. To minimize cost, Freebird components are protoflighted when possible. Prototyping is only used when safety dictates.

Additional research costs are required for immature technology. Several subsystems have components which need further research. Qualification costs are required for the RL10-A4 engines. Qualifying the engines will prevent the need for frequent replacement of the engines and thus save on overall cost. Development costs are also a major factor for the redesign of the fuel cells. The new design entails scaling down the space shuttle fuel cells, and increasing the fuel cell lifetime. This will reduce the weight of Freebird, as well as reducing replacement costs. Development costs are also required for immature technology, such as the MAPS software package.

12.4.3 Summary of Project Cost

Using the modified Lockheed OTV model, the total system costs were estimated. This cost includes program management, complete space vehicle costs, launch segment costs, ground segment costs, and operations and support costs. Table 12.3 shows the system costs for Freebird up to and including launch, which is $1.5 billion.

TABLE 12.3 *Total System Costs Using Modified Lockheed OTV Model (FY94).*

	Cost Estimate ($M)
Program Management	272
Space Vehicle	834
Launch Segment	135
Ground Segment	223
Operations and Support	36
Total	1,500

12.5 Special Producibility Cost/Problems

Producibility problems tend to involve the same items that are identified as cost drivers in the subsystems. The exception centers around the final vehicle assembly and testing.

The re-qualification of the RL-10A engines for 100 starts will be a major issue. This will require a very extensive testing regime which could add significant cost and delays to the overall project. Despite these concerns, re-qualification is absolutely necessary if Freebird is to have any basis as a cost effective system for servicing existing orbital assets. Failure to re-qualify the engines will result in the need to replace the propulsion pack after approximately 5 missions. This would add a cost of about $20 million to each of the missions, making a financial return on each mission improbable. Additionally, the use of composite propellant tanks will require additional testing in the space environment to insure their behavior.

The structures subsystem production problems are centered on the RMS. The RMS redesign vastly outweighs the other producibility items that might be considered. There are several models of the RMS that could be used for Freebird but none, with the exception of the shuttle RMS, has been flown. The testing and validation time for these redesigned systems could be extensive. Additionally, conditions simulating the actual operating environment will have to be maintained during testing to ensure an accurate representation of in flight performance. An alternative to ground testing might be accomplished by using the Shuttle as a test platform for any new RMS design.

GNC and C^3T subsystems are faced with a number of producibility problems. At issue are the LNS and MAPS components. All three of these components are still under development and all exhibit serious operational problems. It appears that there is a strong industry dedication to resolving these problems and producing workable units. However, an exact time frame is not available and it can only be presumed that engineers will resolve their difficulties prior to the commencement of the final design stages of the Freebird project. Additionally, software development for these systems presents a significant challenge. The software must be exhaustively tested to insure that proposed autonomous missions can be guaranteed a high level of reliability and safety.

In addition to subsystems problems there are significant problems regarding the final vehicle assembly. The method of assembly, special assembly equipment and facility availability may present formidable obstacles to the project in terms of cost and total project time.

Foremost is the method of assembly and the specialized equipment necessary to carry out that assembly. Since the vehicle has no parallel in size and shape, all the assembly equipment must be fabricated, calibrated and designed to accurately produce conditions similar to weightlessness.

The obstacles presented by the facilities revolve around availability and level of cleanliness needed for final assembly and testing. Since the vehicle is intended to employ as much proto-flight equipment as possible, a high degree of cleanliness will be needed to insure the integrity of these subsystems. The best parallel to this task is that endured by the Hubble Space Telescope. The HST was a proto-flight system that was clean room assembled, tested and then shipped to the launch site under conditions that would be similar to Project Freebird.

12.6 Production Schedule

A general schedule for the balance of the Freebird Project is as shown in Figure 12.3 . The schedule was developed as a composite of several systems that appear equal in magnitude and complexity. Special consideration is given to the current industry trend of integrating final design with tooling, procurement and manufacturing in order to minimize total time and expense in production. The total project is estimated to take between 36-40 months to complete. This time schedule does not reflect shortfalls in funding, political considerations or changes in mission priorities.

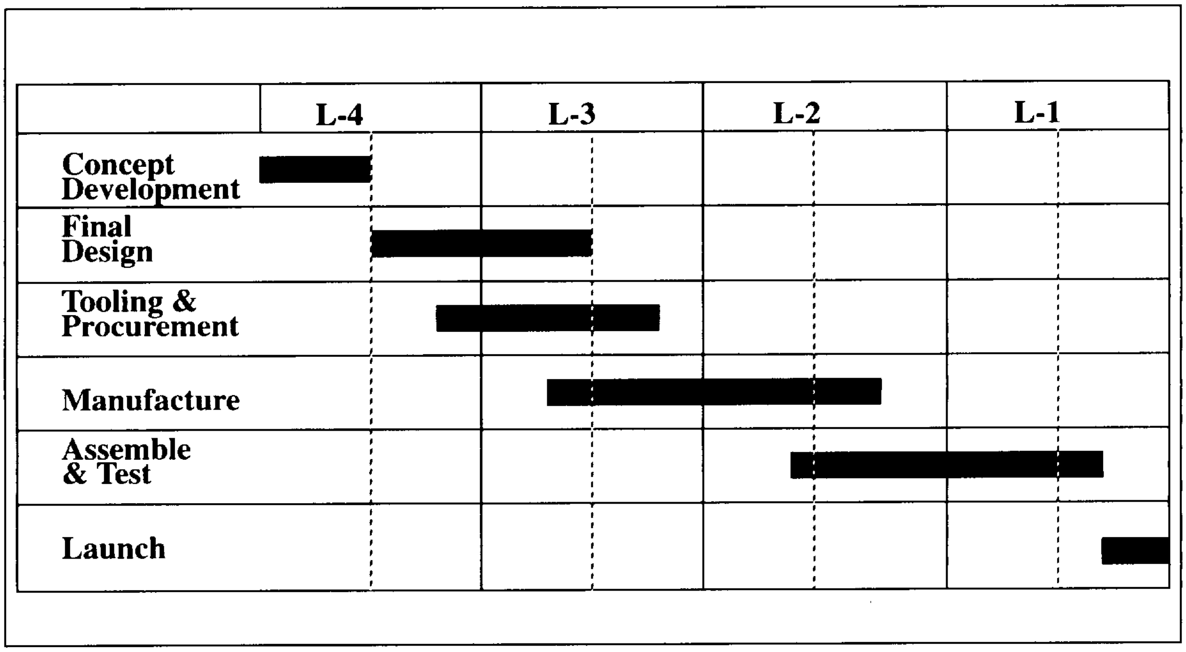

FIGURE 12.3 *General Schedule, Including Contingency Time.*

In order to determine the approximate time involved with each area of the individual subsystems, a contingency time chart, Figure 12.4 , was developed. This chart shows the contingency assigned to each area of the subsystem as an added percentage of time. The assignment of the contingency level is determined by analogy to similar systems and then is adjusted for Freebird specifics. For example, many aspects of the GNC system involve standard components, however, the software development time is expected to be significant. Consequently a rating of high (30%) is added to the final design time.

The subsystem schedule, with adjusted contingency times, is located in Appendix E on page 325 . The chart also shows the final integration, verification and validation time as well as final testing and launch.

TABLE 12.4 *Contingencies Used for Master Schedule.*[a]

	Design	Procurement/Tool	Construct
Structures & Mechanisms	High	Low	Med
P&P	Med	High	Med
Thermal	Med	Med	Low
GNC	High	Low	Low
C^3T	Low	High	Low
ECLSS	High	Med	Med

a. *High corresponds to 30% contingency; Medium corresponds to 20% contingency; Low corresponds to 10% contingency.*

References

1. Larson, Wiley J., and Wertz, James R. (editors) 1992, Space Mission Analysis and Design, 2nd ed.,Torranace California: Microcosm, Inc.

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Chapter 13

System Interfaces

To accomplish the planned missions, Freebird must interface with the International Space Station Alpha (ISSA), the Vehicle Storage Facility (VSF), the Space Shuttle, Soyuz, ground stations, tended assets, payloads, and the environment. This chapter details the various mechanical, communication/data and power interfaces between the vehicle and external factors; internal interfaces are discussed in individual system chapters. The vehicle will be based in the VSF, will pick up crew and supplies at ISSA before every crewed mission, and will be in communication with ISSA as well as ground stations throughout the mission. The vehicle will be expected to perform crew rescues from Soyuz and the Shuttle. Freebird will be capable of maneuvering and manipulating its payload and tended assets as necessary. Freebird will experience a wide range of environmental conditions over the course of its lifetime (see Section 1.4.2 on page 29), and the vehicle must provide a means of dealing with the negative environmental effects.

13.1 Mechanical Interfaces

Freebird was designed to have mechanical interfaces with the VSF, ISSA, the Shuttle, Soyuz, tended assets, its payload, and its environment, as illustrated in Figure 13.1. The vehicle will reside in the VSF, so it requires a system of tethering, discussed in more detail in Section 4.4 on page 279. To enable docking with the Station, the Shuttle, and Soyuz, the vehicle is equipped with a Common Berthing Adapter which holds the two spacecraft together, while forming an airtight seal to allow transfer of crew, cargo, and supplies. Payloads will be secured to the front of the craft while in transit, and will be appropriately positioned in orbit by means of the Remote Manipulator System (RMS). On crewed missions, the crew module acts as the payload and attaches to the base unit by means of an intermodular interface. The modular coupling will be accomplished by teleoperated use of the arms. Tended assets will also be grappled and manipulated as necessary with the RMS. The hull provides protection from thermal, micrometeorite and radiation effects.

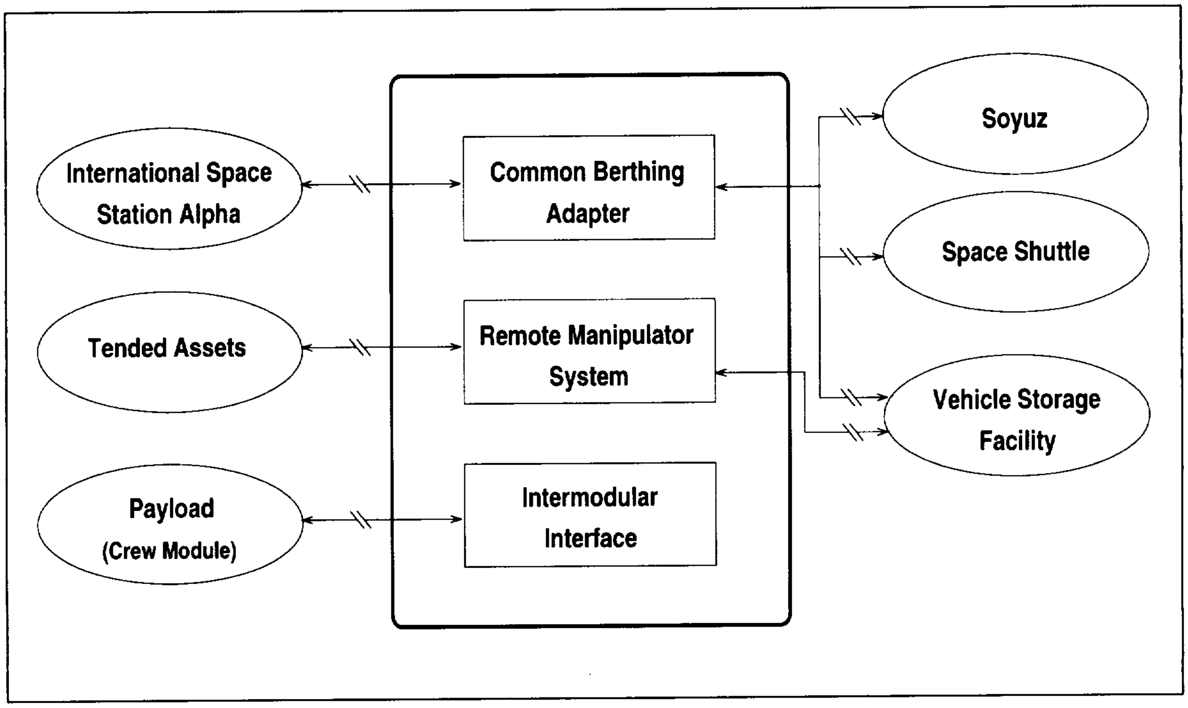

FIGURE 13.1 *System Mechanical Interfaces.*

13.2 Communication/Data Interfaces

Freebird requires communication interfaces with ISSA, ground stations, the Shuttle, Soyuz, and tended assets. As shown in Figure 13.2, Freebird will be in constant communication with the ground station and ISSA, and will communicate with the other systems as necessary. ISSA and ground act as Mission Control, monitoring the progress of the mission and issuing commands. Teleoperation will be controlled from the ground, necessitating a communication link capable of transmitting video with acceptable resolution. Communication with the ground station will be routed through either the Tracking and Data Relay Satellite System (TDRSS) or the Deep Space Network to achieve the necessary coverage and bandwidth. Soyuz and the Shuttle need to communicate with Freebird for position and condition updates during crew rescue. Tended assets may need adjustments in attitude or configuration to facilitate servicing, so Freebird will communicate with them through the ground station.

13.3 Power Interfaces

Freebird was designed to have power interfaces with its payload and the VSF. Figure 13.3 shows this relationship. The payload's power source may not function in transit (solar panels which deploy in orbit, for example), so the vehicle allows its payload to draw power from Freebird's supply to maintain stand-by operations. On crewed missions of course, the crew module will require a steady supply of power to maintain Environmental Control and Life Support System (ECLSS) functions. While not on a mission, the vehicle will be stored in the VSF, drawing a minimal amount of power from the facility's solar panels to maintain stand-by operations.

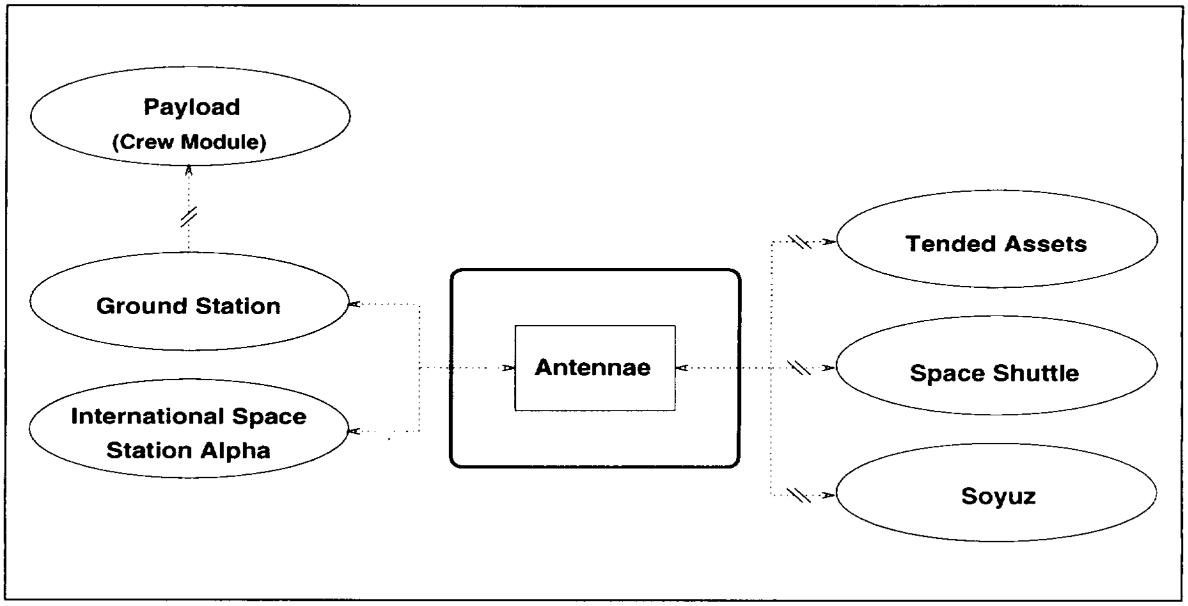

FIGURE 13.2 *System Communication/Data Interfaces.*

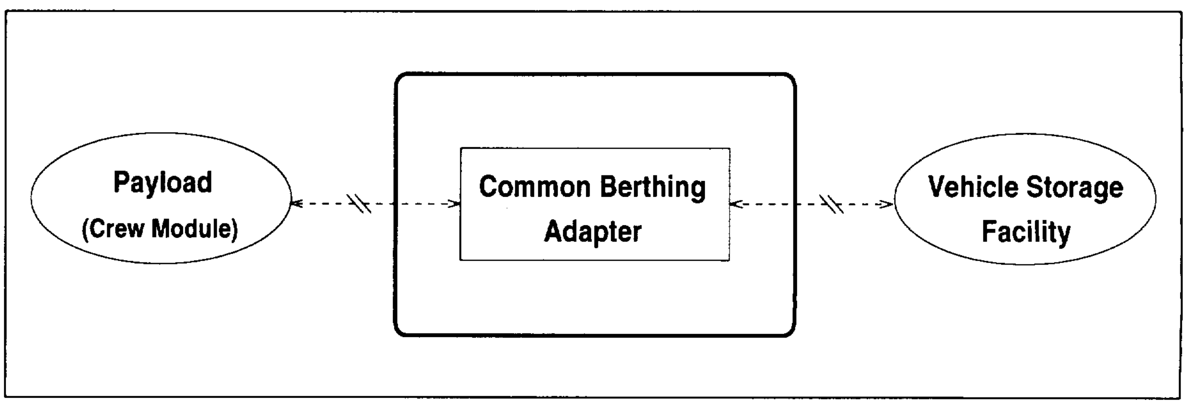

FIGURE 13.3 *System Power Interfaces.*

Chapter 14

Vehicle Design Summary

This chapter summarizes the current Freebird design, both in terms of the vehicle configurations that can be used as well as the overall mass, power, and data budgets.

14.1 Vehicle Configurations

Freebird has a total of four modes in which it can be operated. These four modes are the permutations of having crewed versus teleoperated mission and whether or not auxiliary fuel packs are installed.

The other major configuration occurs during the boost phase, when additional support structure is temporarily placed as "packing material" around the weakest links of the system.

14.1.1 Launch

Since orbital insertion of the Freebird actually involves two separate launches, both launch configurations are shown in Figure 14.1. This configuration has the base unit launching in an Ariane V and the crew module launching a Proton C.

Note that there will be additional structure surrounding the vehicle and reinforcing the structure, a necessity for Freebird to withstand launch loads. This material is discarded in orbit and is not a constituent portion of the vehicle. Depending on which launch vehicle is used, slight modifications can be made to the configuration, but the basic mechanism will remain identical. The backup launch vehicles are both owned by the United States: the Titan IV will be used as a backup on the launch of both the propulsion module and base unit and the Titan III will be the backup for the launching of extra fuel and the crew module.

For refueling missions, Figure 14.2 shows the fuel tank configuration for a Proton launch. Note that the fuel tanks occupy almost the entire capability of the Proton, filling it both in volume and in lifting capability. If a Titan III is to be used, in emergency or backup situations, less fuel will have to be launched due to the reduced lifting capability of this vehicle.

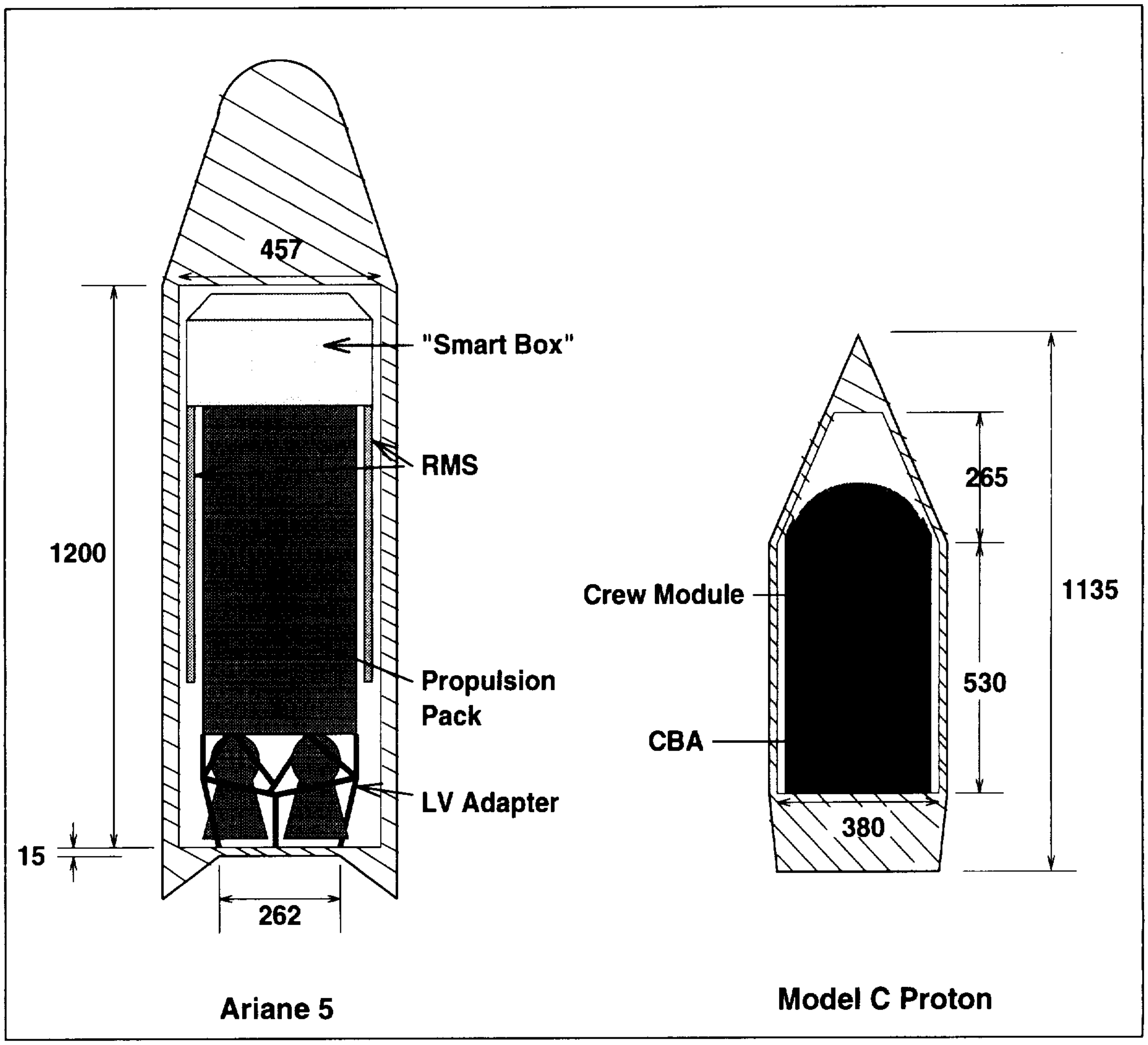

FIGURE 14.1 *Launch Configurations.*

A launch of extra fuel tanks to be used in extended missions will look almost identical to an extra fuel launch; however it will include additional structural support as with the launch of the base unit or the propulsion unit to mitigate the effects of the launch loads.

To launch a replacement propulsion module, the configuration shown in Figure 14.3 will be used. To accommodate the propulsion pack, a vehicle the size of the Ariane V or the Titan IV will be required.

14.1.2 Crewed Mission

In a crewed mission scenario, the crew module will be mated to the base unit at the Vehicle Storage Facility (VSF). In addition, extra fuel packs may be added to the vehicle to extend the range of the mission. Figure 14.4 shows a rendered figure of the configuration for a crewed mission with the extra fuel tanks added.

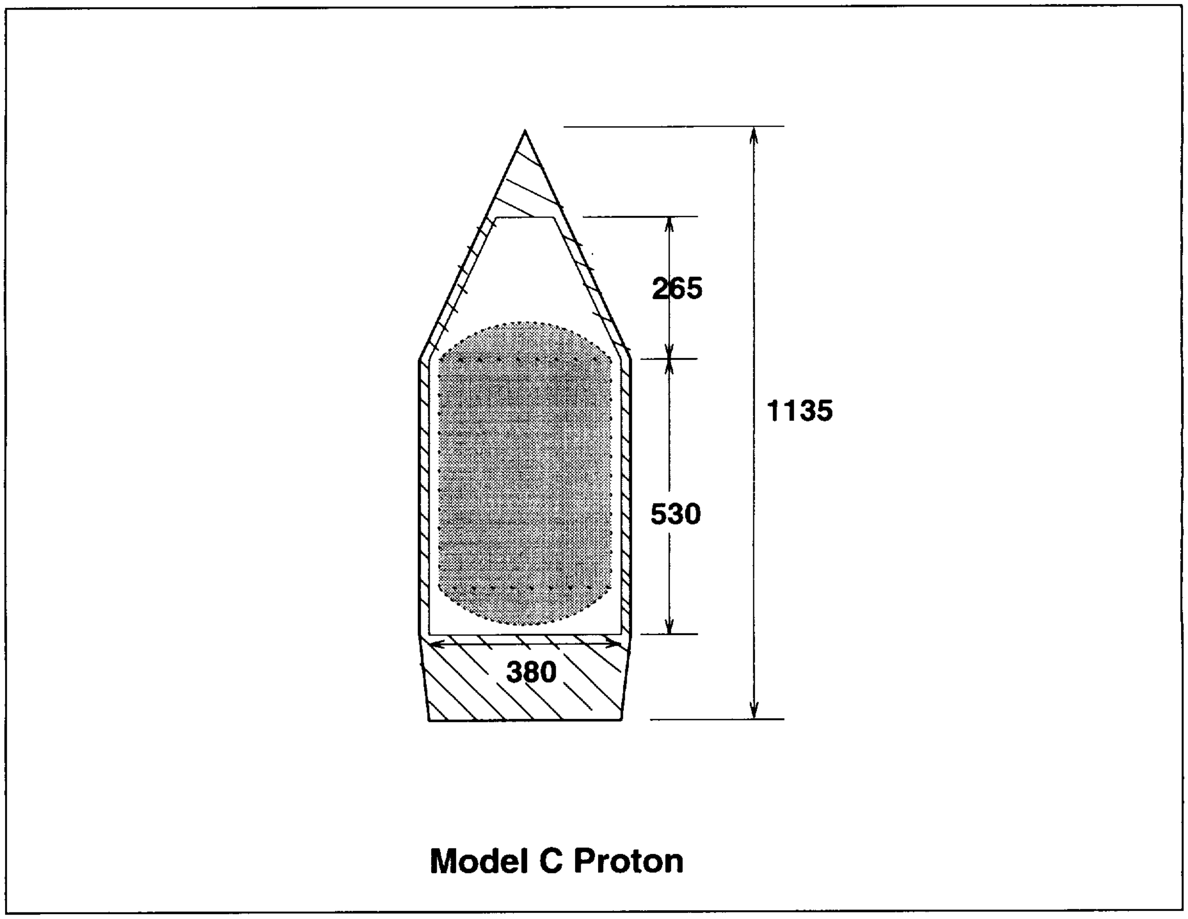

FIGURE 14.2 *Fuel Tank Launch Configuration.*

Rescue missions as well as other nearby (Mass Ratio < 3.5) crewed missions will not require the extra fuel tanks. The rendered image of a crewed configuration without extra fuel tanks is shown in Figure 14.5.

14.1.3 Teleoperated Missions

Before an uncrewed mission, the crew module will be detached from the base unit and stored in the VSF. If necessary, a mission specific docking adapter will be attached to the front of the vehicle in order to mate properly with the target asset. It is expected that this configuration, shown in Figure 14.6, will be used for the bulk of Freebird's missions.

For extended missions, or ones requiring large cargo transfer, extra fuel tanks will be added to the teleoperated vehicle. If fuel cannot be launched directly into a polar orbit for Freebird to utilize on a return trip, extra fuel tanks will be required for a polar mission. All GEO missions will require the extra tanks.

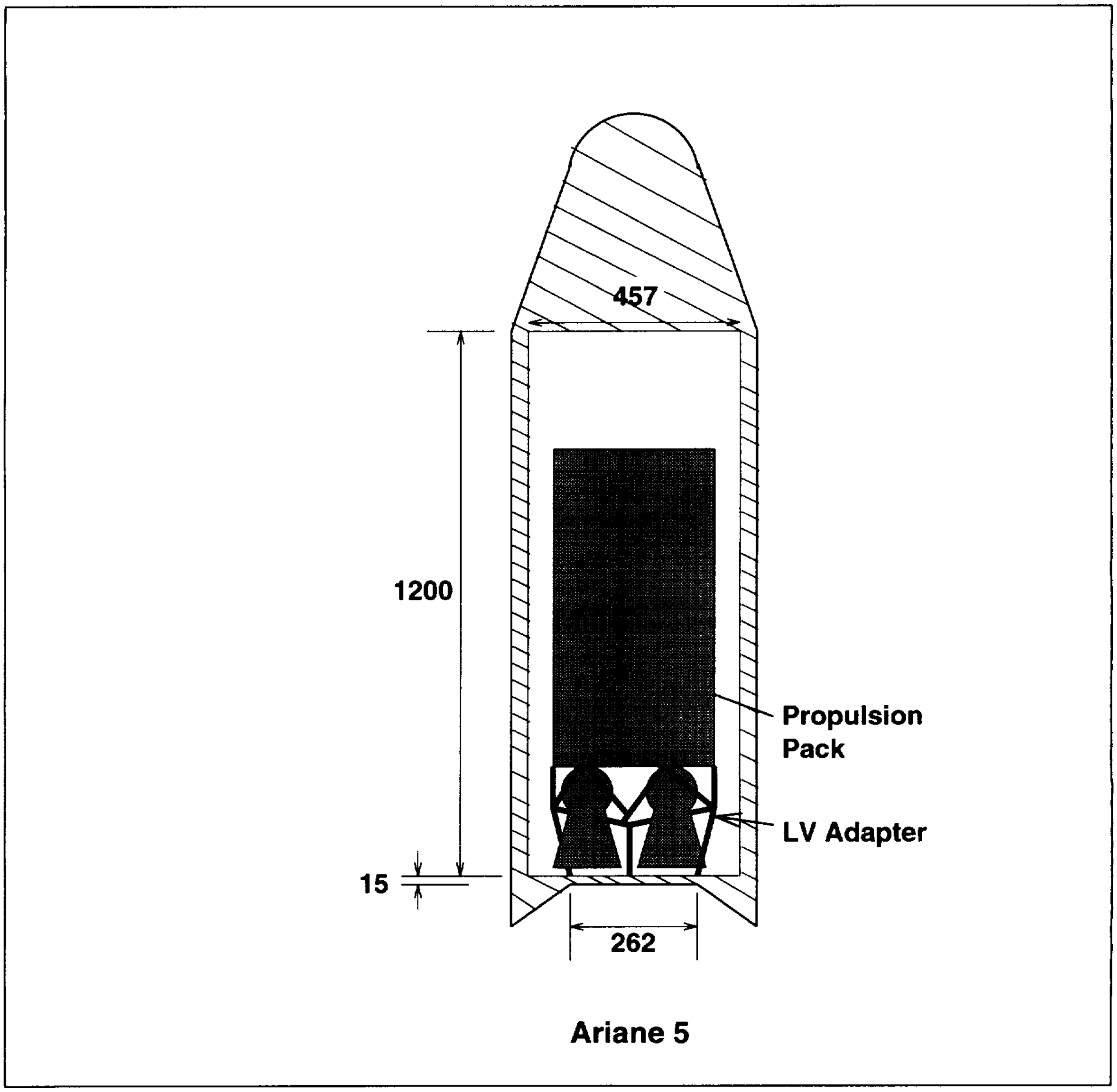

FIGURE 14.3 *Propulsion Module Launch Configuration.*

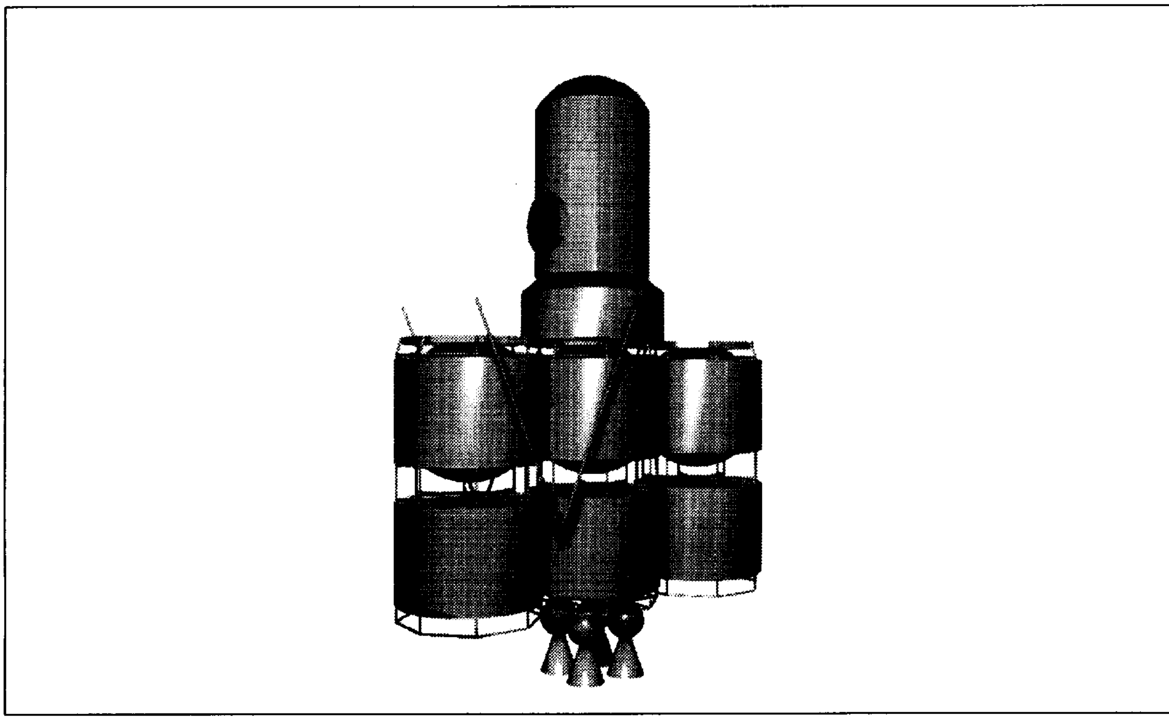

FIGURE 14.4 *Extended Crewed Mission Configuration (with Extra Fuel Tanks Added).*

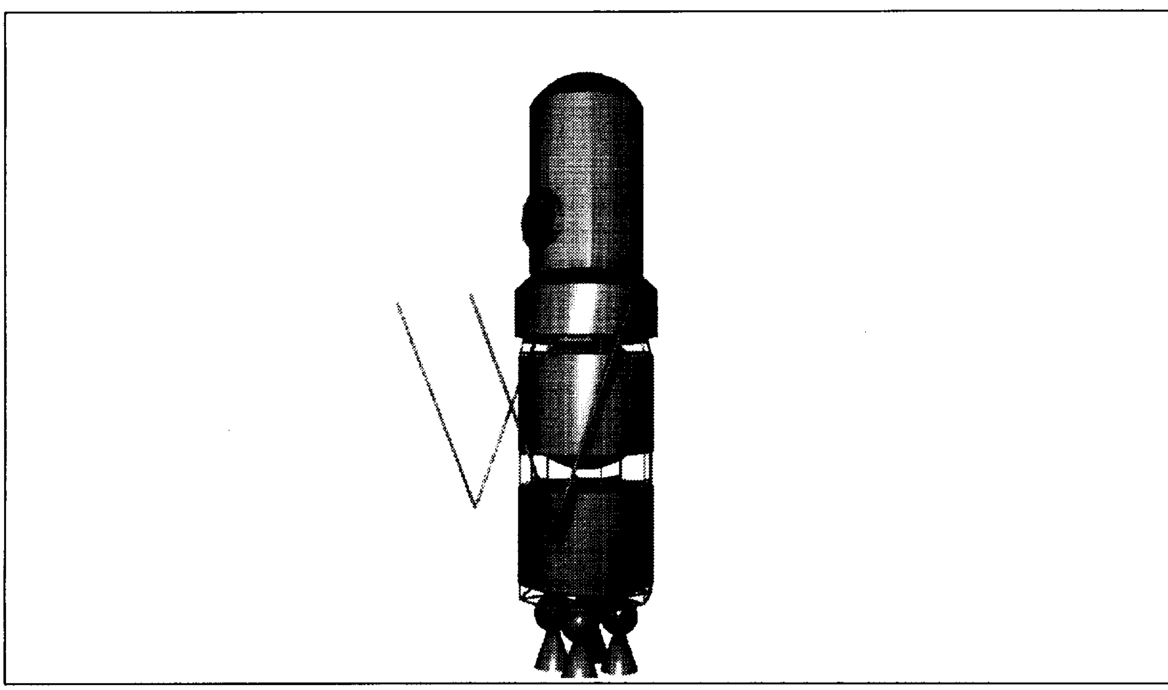

FIGURE 14.5 *Crewed/Rescue Mission Configuration (without Extra Fuel Tanks Added).*

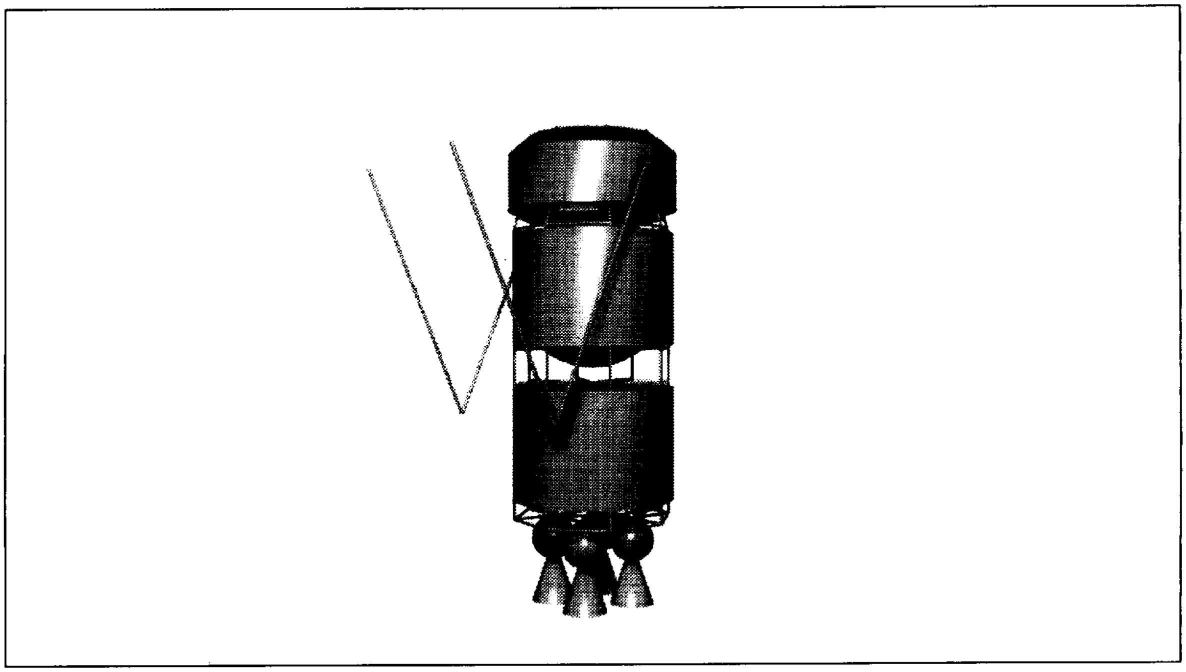

FIGURE 14.6 *Teleoperated Mission Configuration (without Extra Fuel Tanks Added).*

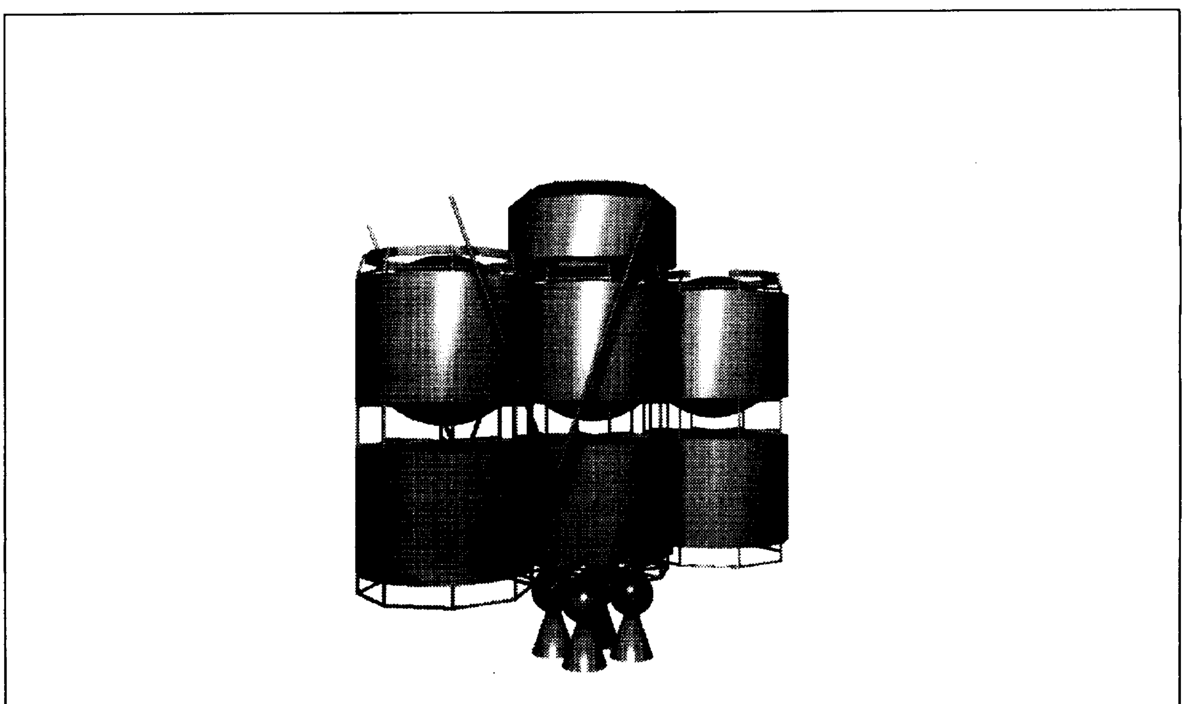

FIGURE 14.7 *Teleoperated Mission Configuration (with Extra Fuel Tank Added).*

14.2 Freebird Dynamics

To understand the control dynamics of Freebird, its center of mass and moments of inertia were computed. Once all of Freebird's components, masses, and volumes were known, the vehicle was configured for functionality and balance. Each component was modeled as a point mass. Whenever a component spanned a large volume, the center of mass was approximated by appropriately distributing the mass over the volume. The center of mass was calculated using the following formula for a discretized system like this:

$$\frac{\sum x_i m_i}{\sum m_i} \qquad \textbf{(Equation 14.1)}$$

where the index i ranges over all components. The center of mass was calculated for four different configurations: crewed, wet and dry; and teleoperated, wet and dry. Also, calculations were not done for Freebird in its extended mission configuration. Note that "dry" implies that main engine fuel was not included, but RCS and fuel cell fuel was included. Table 14.1 summarizes the center of mass coordinates of each module, and Table 14.2 summarizes the center of mass coordinates for the four configurations considered. Once the center of mass coordinates were known for each vehicle configuration, rough moments and products of inertia around that point were calculated for this discrete system. These values are summarized in Table 14.3. It is important to observe that I_{xz} and I_{yz} are non-zero. This means that the body axis system is not the principle axis system. However, these products of inertia are at least one order of magnitude smaller than the moments of inertia, indicating that the difference between body and principle axes is not large, and therefore not detrimental to vehicle control.

TABLE 14.1 *Approximate Coordinates for Module-Specific Centers of Mass.*

	Coordinates of the Center of Mass [m]		
	x	**y**	**z**
Crew Module	2.93	0	0
"Smartbox"	-2.47	0	0
Propulsion Pack	-7.19	0	0

TABLE 14.2 *Approximate Coordinates for Configuration-Specific Centers of Mass.*

	Coordinates of the Center of Mass [m]		
	x	**y**	**z**
Crewed, Dry	-2.5	0	0
Crewed, Wet	-3.9	0	0
Teleoperated, Dry	-5.1	0	0
Teleoperated, Wet	-4.7	0	0

TABLE 14.3 *Approximate Configuration-Specific Moments of Inertia About Respective Center of Masses.*

	Moments of Inertia [kg m^2]					
	I_{xx}	I_{yy}	I_{zz}	I_{xy}	I_{xz}	I_{yz}
Crewed, Dry	13,710	196,000	206,300	0	-758.2	-92.98
Crewed, Wet	13,710	248,600	258,800	0	-863.7	-92.98
Teleoperated, Dry	13,290	61,580	72,210	0	624.8	-92.98
Teleoperated, Wet	13,290	99,000	109,600	0	552.6	-92.98

14.3 Summary Budgets

The numbers for these budgets were derived by breaking down the vehicle into systems and components, and developing mass, power and cost numbers for each individual component. A contingency factor between 5% and 25% was included depending on technology maturity.

The total mass of the vehicle, with crew, is 8,731kg. Table 14.4 shows the mass distribution by system and module. This includes a full load of fuel for the RCS thrusters and the fuel cells, as well as 3 astronauts and a 5% system contingency. Appendix A contains a more detailed mass breakdown.

TABLE 14.4 *Mass Breakdown By Subsystem and Module.*

Subsystem	Base Unit Mass [kg]	Crew Module Mass [kg]
Structures, Materials & Mechanisms	1,812	1,711
Thermal Control	47	72
Guidance, Navigation & Control	173	---
Command, Communication, Control & Telemetry	78	---
Propulsion	2,256	---
Power	578	---
Environmental Control & Life Support	---	1,588
Subtotal	**4,944**	**3,371**
Overall System Contingency (5%)	**247**	**169**
Total Mass	**5,191**	**3,540**

The cost to develop, produce, and test Freebird are broken down by subsystem in Table 14.5. This figure includes hardware, integration and subsystem level verification and validation testing, but not system-level tests. The total program cost is $1,472,415,000, which includes costs for hardware, testing and integration, program management, launch, and operations and support. For a more detailed breakdown, see Chapter 12 and Appendix D.

TABLE 14.5 *Cost Breakdown By Subsystem.*

Subsystem	Hardware Cost [$k]	Subsystem Cost [$k]
Structures, Materials & Mechanisms.	115,725	336,432
Thermal Control	1,925	6,103
Guidance, Navigation & Control	18,903	81,125
Command, Communication, Control & Telemetry	8,781	28,865
Propulsion	29,525	125,312
Power	8,092	33,281
Environmental Control & Life Support	50,688	94,376
Total Component Costs	**233,639**	**705,494**

Power budgets are detailed in Chapter 4 (Power) and include stand-by power requirements, average power consumption, and mission power profiles. Individual component power requirements can be found in Appendix B.

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Part III: Pages 255-296
Operations

Chapter

1

Launch and Ground Support

The primary objectives set for the Freebird vehicle drove many of the considerations for selection of a launch vehicle. Different launch vehicles were chosen for the different modules in the Freebird system, due to the different geometries and deliverable masses in the design of each module.

1.1 Propulsion Pack and "Smartbox" Launch

The combination of the propulsion pack and the "smartbox" is the largest configuration of modules which must be launched. In addition to the initial launch of the propulsion pack and the "smartbox", an estimated four identical replacement propulsion pack modules will need to be launched into orbit over the system lifetime of ten years. Both the initial launch of the propulsion pack and "smartbox" combination, and the successive launches of replacement propulsion packs, will be on the same vehicle.

The propulsion pack module volume and mass were set by the mission objectives and propulsion system selection. Since both the volume and mass were relatively large, the number of possible launch vehicles for this configuration of modules was limited. The Ariane 5, the Titan IV, the space shuttle, and the Energia all met the necessary minimum mass and volume requirements, and were considered as possible launch vehicle choices.

1.1.1 Vehicle Choice

The combined propulsion pack and "smartbox" modules, designed to be launched attached and fully fueled with cryogenics, weigh approximately 20 000kg and have a volume of 171m^3. The Energia, which delivers 100 000kg to ISSA orbit, was ruled out as a prime or backup launch vehicle possibility due to a large, unusable mass margin. The space shuttle was ruled out as a primary or backup vehicle due to the danger to the shuttle crew from carrying cryogenic fuels as cargo in the payload bay.

The Ariane 5 and the Titan IV both have similar performance capabilities to the orbit of the International Space Station Alpha (ISSA), and meet the necessary volume and mass requirements (see Table 1.1). Either vehicle would be capable of placing the propulsion pack - "smartbox" combination or the propulsion pack module alone into the proper orbit. Since the performance capabilities are nearly equal, the Ariane 5 was chosen as the prime vehicle over the Titan IV because of its inexpensive launch cost ($90-$100 million FY90). The Titan IV was selected as a backup vehicle at a launch cost of $250-$300 million FY90. The use of a foreign launch vehicle produces significant cost savings. The Ariane 5 is currently in development, with a planned first launch in 1996 [2]. In the event that this vehicle does not achieve operational status in time to deliver the propulsion pack - "smartbox" combination or the propulsion pack module to orbit, the Titan IV will be used.

TABLE 1.1 *Parameters for Primary and Backup Vehicles.*

Vehicle	Titan III	Titan IV	Ariane 5	Proton C
Maximum Payload Volume[a] [m^3]	68	208	197	58
Maximum Mass to Orbit[b] [kg]	14,000	20,000	22,000	20,000
Maximum Axial Loads [g's]	+2.5, -5.0	+3.3, -6.5	+4.5	+6.0
Maximum Lateral Loads [g's]	+/- 1.7	+/- 1.5	N/A	+/- 3
Launch Cost[c] [$M]	130-160	250-300	90-100	35
Reliability	0.930	0.930	0.875	0.877
Country of ownership	U.S.	U.S.	ESA	Russia

a. Primary cylindrical volume only.
b. Circular orbit at 390km, 51.6° inclination.
c. Fiscal year 1990 (FY90) dollars.

1.1.2 Scheduling

Nominal manifesting information was not available for the Ariane 5. Nominal manifesting for a Titan IV launch occurs at least 33 months prior to the desired launch date. Approximately eight launches per year are expected for both these vehicles [1][2]. For more information on issues which concern contracting for a foreign launch vehicle, see Section 8.2.3 on page 290.

1.2 Crew Module Launch

The volume needed for the crew module drove the launch vehicle selection for this module. Once the needed volume was established at 60m^3, a list of candidate launch vehicles which matched the volume needed was assembled which included the Proton C and the Titan III (see Table 1.1).

1.2.1 Vehicle Choice

Since both the Proton C and the Titan III have similar performance capabilities (see Table 1.1), the launch cost became the deciding factor. The Russian-owned Proton C was chosen as the primary launch vehicle for its cost of $35 million FY90 per launch. The Titan III was chosen as the backup launch vehicle with a cost of $130-$160 million FY90.

With the dissolution of the Soviet Union, concerns about the integrity of launch support facilities for the Proton series of vehicles arose. These reports were entirely unfounded, and recent delegations to the launch site at Baikonour Cosmodrome have proved the facilities to be in satisfactory condition. However, concerns about the degrading infrastructure in the community of Leninsk surrounding the Baikonour facilities are quite valid. If the standard of living around Baikonour does not improve, the current trend of skilled workers leaving the region will continue. These workers maintain and operate the launch facility. A significant drop in the skilled labor force at Baikonour will have dangerous effects on the operation of the Proton launch system. [3]

1.2.2 Scheduling

Nominal manifesting for a Proton C launch occurs at least 18 months prior to the desired launch date. 32 Proton launches are expected yearly. For more information on issues surrounding contracting for a foreign launch vehicle, see Section 8.2.3 on page 290

1.3 Fuel Tank Launches

The drivers influencing selection of a launcher for the fuel tanks were cost and turnaround time for additional launches. In order to keep mission costs down, the launch vehicle selected had to be the least expensive cost option per unit of deliverable mass to orbit. In addition, several launches of fuel must occur within a few weeks in order to perform missions requiring extra fuel tanks. Thus, the launch vehicle chosen for the fuel tanks must be capable of a quick turnaround time for multiple launches.

1.3.1 Vehicle Choice

Taking into account of cost per unit of deliverable mass to orbit and turnaround time, the Proton C was selected as the prime launch vehicle for fuel tanks. At a cost of $1750 FY90 per kg of payload, it was the least expensive launch vehicle option. In addition, the Proton turnaround time of approximately one week is adequate for Freebird mission requirements.

The Titan III was chosen as a backup launch vehicle for its similar performance capabilities and fairing geometry as compared to the Proton C. However, the Titan cost per unit of deliverable mass to orbit is 400% more than the Proton C cost per unit of deliverable mass. Should the Proton become unavailable for fuel tank launches, the anticipated cost per mission will increase by at least 400%.

1.3.2 Scheduling

Nominal manifesting for a Proton C launch occurs at least 18 months prior to the desired launch date. The best turnaround time for a second Proton launch is approximately one week. 32 Proton launches are expected yearly, providing an ample supply for Freebird needs.

1.4 Crew Launches

Crew will be launched to Freebird aboard the space shuttle resupply and crew change-out missions to ISSA. These ISSA resupply missions are expected to be on the order of four to five per year. This provides ample opportunity to carry out the anticipated 1-4 crewed Freebird missions per year. For more information on the crew, see Chapter 5 on page 281.

1.5 Transportation to Launch Site

Regardless of where Freebird will be manufactured, or which site each module will be launched from, transportation will be required from the place of final component testing and evaluation to the launch site. Table 1.2

sums up the modes of transportation required for each launch site. The table is based on manufacture of Freebird modules in the U.S.

TABLE 1.2 *Transportation Modes from U.S. Manufacturers to Domestic and Foreign Launch Sites.*

Launch Site (Location)	Launch Vehicles Supported at Site	Preferred Payload Transportation Modes
Cape Canaveral (Florida, U.S.A.)	Titan III Titan IV	Rail and Barge
Guiana Space Center (Kourou, French Guiana)	Ariane 5	Barge
Baikonur Cosmodrome (Leninsk, Kazahkstan)	Proton C	Airplane

1.6 Ground Support

All Freebird mission operations will be controlled and monitored from the ground.

1.6.1 Freebird Mission Control

Freebird will need several console control stations in the new combined Shuttle and Space Station Mission Control Center at Johnson Space Center in order to handle mission operations. These console stations will be "soft" terminals, consisting of computer terminals and displays, which are easily and quickly configurable for each mission. At least one dedicated flight controller will be required for each Freebird mission.

For certain teleoperated missions in geostationary orbit where remote manipulator system (RMS) operations might be intensive, a long lag time on video downlink is undesirable. In order to alleviate this lag, a portable control console would be transported to a location directly under the geostationary asset. This would reduce communication lag time between the controller and the Freebird vehicle, making control of intricate RMS operations easier.

References

1. Isakowitz, Steven J. 1991. *International Reference Guide to Space Launch Systems.* Washington, DC: AIAA.

2. National Aeronautics and Space Administration, Jet Propulsion Laboratory. 1993. *Launch System Highlights for JPL Mission Planning.* JPL D-6936, Rev. C.

3. Alexander, Bretton and Dalby, A. Royce. 24 September 1993. *ANSER Report on Findings at Baikonur Cosmodrome.* Washington, DC: ANSER Corporation.

Chapter 2

Failure Modes

This chapter covers the failure modes which Freebird may encounter throughout its lifetime, as well as the recovery from these failures. Failures are occurrences which compromise any of the primary mission objectives. By examining failure modes and recovery scenarios, missions can be completed successfully. This is also an important part of assuring the reliability of the system [1].

2.1 Mission Failure Modes

Mission failure modes include all malfunctions of the Freebird vehicle or human error in operation which preclude mission completion. They can occur during any of the following: launch, docking, stand-by mode, fueling and refueling, or mission operations. Unforeseen complications during launch may disable or destroy Freebird. This possibility has been minimized by designing to the launch vehicle environment. Freebird will dock autonomously using the laser navigation system. If this system fails, docking can be accomplished by the manual or remote teleoperated system. If Freebird shuts down and cannot be restarted when in stand-by mode, no missions can be started or accomplished. Sufficient redundancy to prevent this occurrence has been incorporated. If the fuel tanks cannot be fueled or refueled, Freebird cannot execute the proper orbital maneuvers. During all mission operations, subsystem failures may make the primary objectives impossible. For this reason, a detailed list of failure modes for each subsystem was developed.

2.2 Subsystem Failure Modes

A chart listing detailed failures of all critical components is located in the chapter for each subsystem in Part II. Following is a list of references for all subsystem failure mode tables:

- Structures - Section 3.7
- Power - Section 4.5.3
- Propulsion - Section 5.4

- Thermal - Section 6.5
- Guidance, Navigation, and Control - Section 7.9
- Command, Communications, Control, and Telemetry - Section 8.6
- Environmental Control and Life Support Systems - Section 9.10

Each failure mode table lists the component, its failure, the effect of the failure, the prevention for or response to the failure, and the criticality of such a failure in each configuration of Freebird (crewed or teleoperated configuration). The failures can fall into one of four criticality categories, based on the severity of the failure. Table 2.1 lists the levels of mission criticality which can be assigned to failures.

TABLE 2.1 *Level of Mission Criticality*

Critical Level	Definition
1	Life threatening
2	Mission critical (mission aborted)
3	Non-mission critical
4	Telemetry critical

If a failure

1. Places the lives of the crew in jeopardy, then it is considered life threatening;
2. Affects the mission to the point that it must be aborted, then it is considered mission critical;
3. Affects the mission such that it is compromised but may still be completed, then the failure is considered non-mission critical; and
4. Has no meaningful effect on the mission or crew but can be detected, then it is considered telemetry critical.

This definition of failure criticality is the same as for Project Perseus [2]. An evaluation of each component's criticality is listed in Appendix C on page 309.

2.3 Failure Recovery

Failure prevention methods or recovery scenarios have been developed for each failure mode which is mission critical or life threatening. As mentioned above, these are listed in each subsystem's failure mode table. In most cases failures are prevented using three methods: fault avoidance, fault tolerance, and functional redundancy. Fault avoidance is designing a robust system, incorporating sizable safety margins, a rigorous testing program, and proven components. Fault tolerance is the use of redundant components to take over for a failed part and includes failure detection. Redundancy can be either full redundancy, partial redundancy, or internal redundancy. Functional redundancy is the ability to switch to another system to complete the functions of a failed component or system. All of these methods have been employed for Project Freebird.

In other situations, especially in the case of unexpected failures during mission operations, a recovery strategy can be formulated from the ground station and then implemented.

References

1. Dhillon, B.S. and Chanan Singh. *Engineering Reliability, New Techniques and Applications*. New York: John Wiley & Sons, 1981.

2. MIT Space Systems Engineering, Spring 1993, *Project Perseus-a Crew Return Vehicle for Space Station Freedom*. Massachusetts Institute of Technology, Cambridge, MA, 1993. Page 486.

Chapter

3

Mission Timelines

The mission scenarios that Freebird is most likely to encounter may be divided into several mission phases. In the pages that follow, various missions are described from the initial system power-up to the completion of the mission and the final system power-down. For each scenario, the mission phases are categorized into Pre-Mission Activity, Mission Activity, and Post-Mission Activity.

3.1 Missions in Low Earth Orbit

Based on the primary objective for Freebird of tending and repairing assets in LEO, the most common missions that Freebird is likely to encounter are those missions to various inclinations in LEO where space-based assets lie. Furthermore, the configuration for the mission (teleoperated or crewed) affects the type of activity that the vehicle will undergo.

3.1.1 Teleoperated Mission in Low Earth Orbit

Figure 3.1 is a sample timeline for a mission to tend and/or repair a space-based asset in low earth orbit with the vehicle in a teleoperated configuration. Should the vehicle not be in stand-by mode, the first phase of activity is to power-up to stand-by mode (Section 4.4.2 on page 279). For a teleoperated LEO mission, the vehicle would be fully assembled at this point, (consisting only of the base unit), and, as a result, would be ready for testing (Section 10.3 on page 208). Six hours is allotted for the Integrated Vehicle Test so that repairs may be made if any of the tests should fail. Once the vehicle successfully undergoes the Integrated Vehicle Test, it detaches from the VSF (where it is stored) and is refueled at ISSA if necessary. An estimated time of six hours is required to refill the propellant tanks. If the specific mission requires payload, the payload is attached to the base unit after refueling. This is the last phase of Pre-Mission Activity.

Mission Activity effectively begins when the RL10-A4 engines are fired. The length of time it would take to reach the desired position will vary from mission to mission. The corrective maneuvers required to make a rendezvous with the orbital asset possible (proximity operations) are included in the six-and-a-half hour estimate for the approach. Rendezvous and docking with the orbital asset, accomplished with the help of the Remote

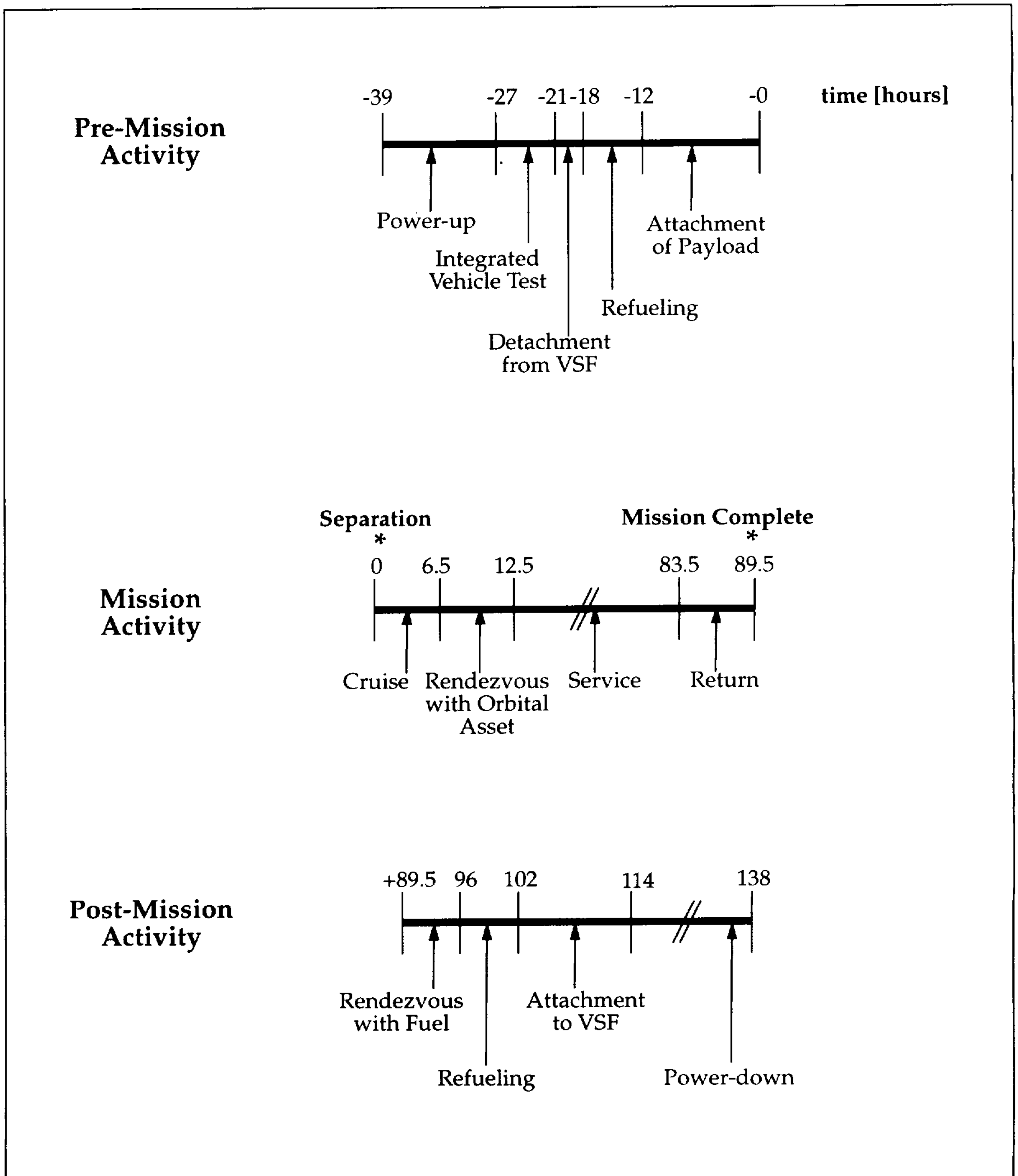

FIGURE 3.1 *Teleoperated LEO Mission Timeline*

Manipulation Systems (RMS), requires about six hours because of the complexity of the maneuver. In Figure 3.1, almost three days are designated for servicing the orbital asset. Clearly, the servicing time depends heavily upon the type of service or the type of mission. An extended servicing time is possible, but limited by several factors, including the amount of fuel that the vehicle is equipped with for maneuvering purposes, the available power, and the additional cost accumulated. The return trip completes the actual mission.

Once the mission is completed, the vehicle is refueled for the next mission. In the case that the refueling tanks are in orbit, launched from the ground to a position nearby ISSA, an additional rendezvous is required, and the refueling will therefore take over 12 hours. However, in the case that the fuel is obtainable at ISSA from the VSF, no additional rendezvous is required, and refueling would be preceded by the attachment of the vehicle to the VSF. Figure 3.1 shows the situation in which a rendezvous with the fuel is required. The attachment of the vehicle to the VSF follows the refueling. The final phase of Post-Mission Activity is power-down to stand-by mode (or a full power-down, which is shown in Figure 3.1.

3.1.2 Crewed Mission in Low Earth Orbit

Figure 3.2 is a sample timeline of a crewed mission in LEO. The timeline for a crewed mission in LEO is similar to the timeline for a teleoperated mission. The main constraint for this type of mission, however, is the 4-day design limit of maintaining an operable cabin environment for astronauts, based on the mission requirements. The sample timeline for the crewed mission to LEO begins by powering-up to stand-by mode. Once again, 6 hours is allowed for the Integrated Vehicle Test. After testing, the vehicle detaches itself from the VSF with the crew capsule already mounted onto the "smart box". If extra fuel tanks are required for a specific mission, they are attached at this point in time. Figure 3.2 allows 24 hours to rendezvous and attach two sets of fuel tanks. Up to this point, the procedure is teleoperated. A 12-hour period is designated for a rendezvous with ISSA to board the astronauts who will perform EVA.

Mission Activity is initiated by the separation of the vehicle from ISSA after ingress. After cruising for a length of time (which depends upon the destination position), Freebird accomplishes a rendezvous with the orbital asset. The service and return phases of activity complete the core of the Mission Activity. The Mission Activity is effectively the same for the two configurations (crewed and teleoperated). In a crewed configuration, however, an astronaut would perform EVA to repair a satellite which, for example, is not technologically advanced and cannot be repaired by the RMS.

In Figure 3.2, after returning to and undergoing a rendezvous with ISSA, the astronauts will have exhausted the 4-day operating envelope of their life support system. (The timeline presents generous allocations of time which allow for the limiting 4-day case to be met.) Note that almost three of the four days are allotted for the servicing of the orbital asset. Once the astronauts return to ISSA, the vehicle is refueled (remotely). Again, if the additional fuel is launched from the ground to a position slightly away from ISSA (case shown in Figure 3.2), then the vehicle will rendezvous with the fuel tanker and refuel prior to its attachment to the VSF and prior to power-down. If the vehicle is to be refueled at ISSA, then the vehicle would first attach itself to the VSF--a 12-hour activity--and then refuel afterwards.

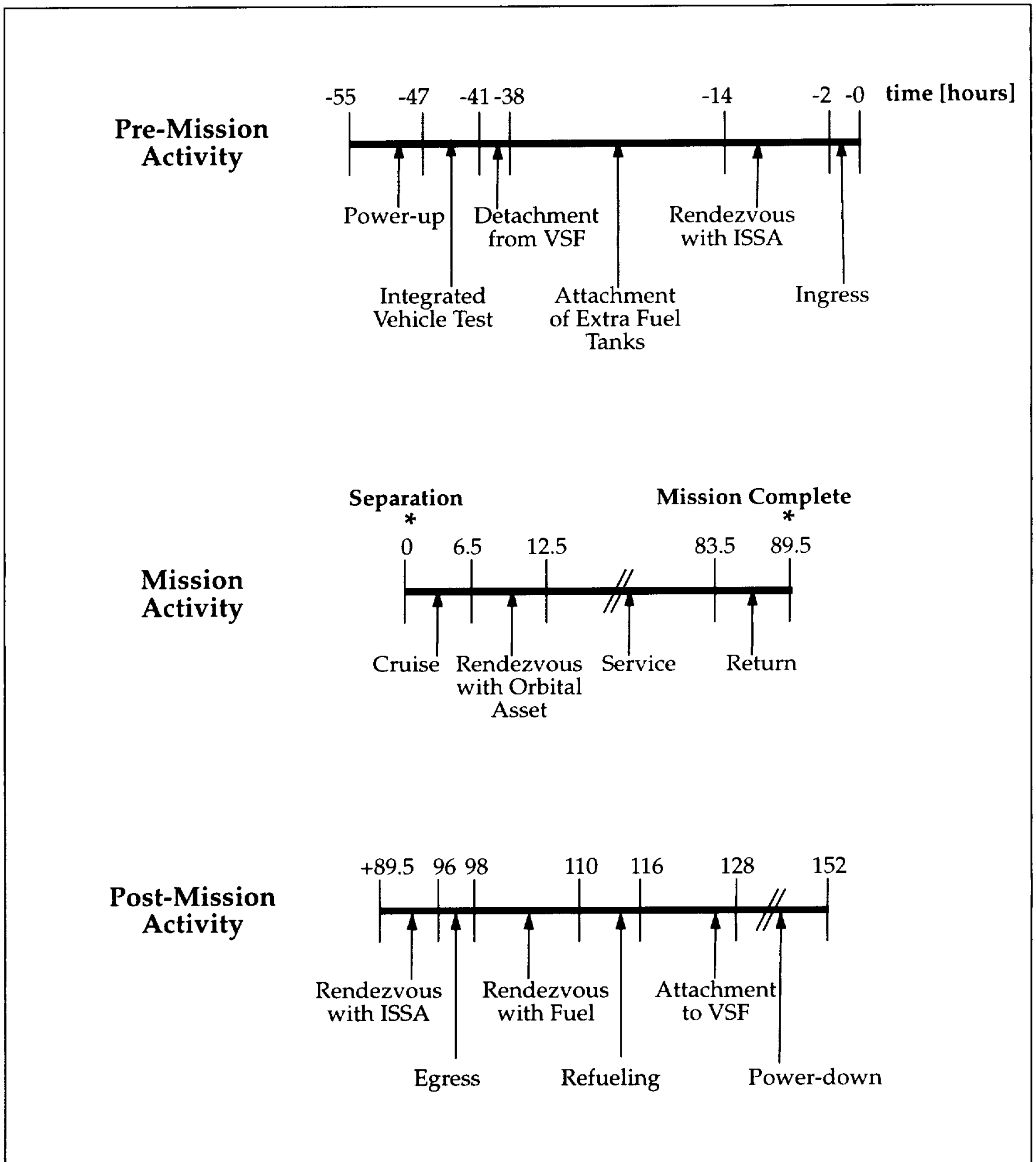

FIGURE 3.2 *Crewed Low Earth Orbit Mission Timeline.*

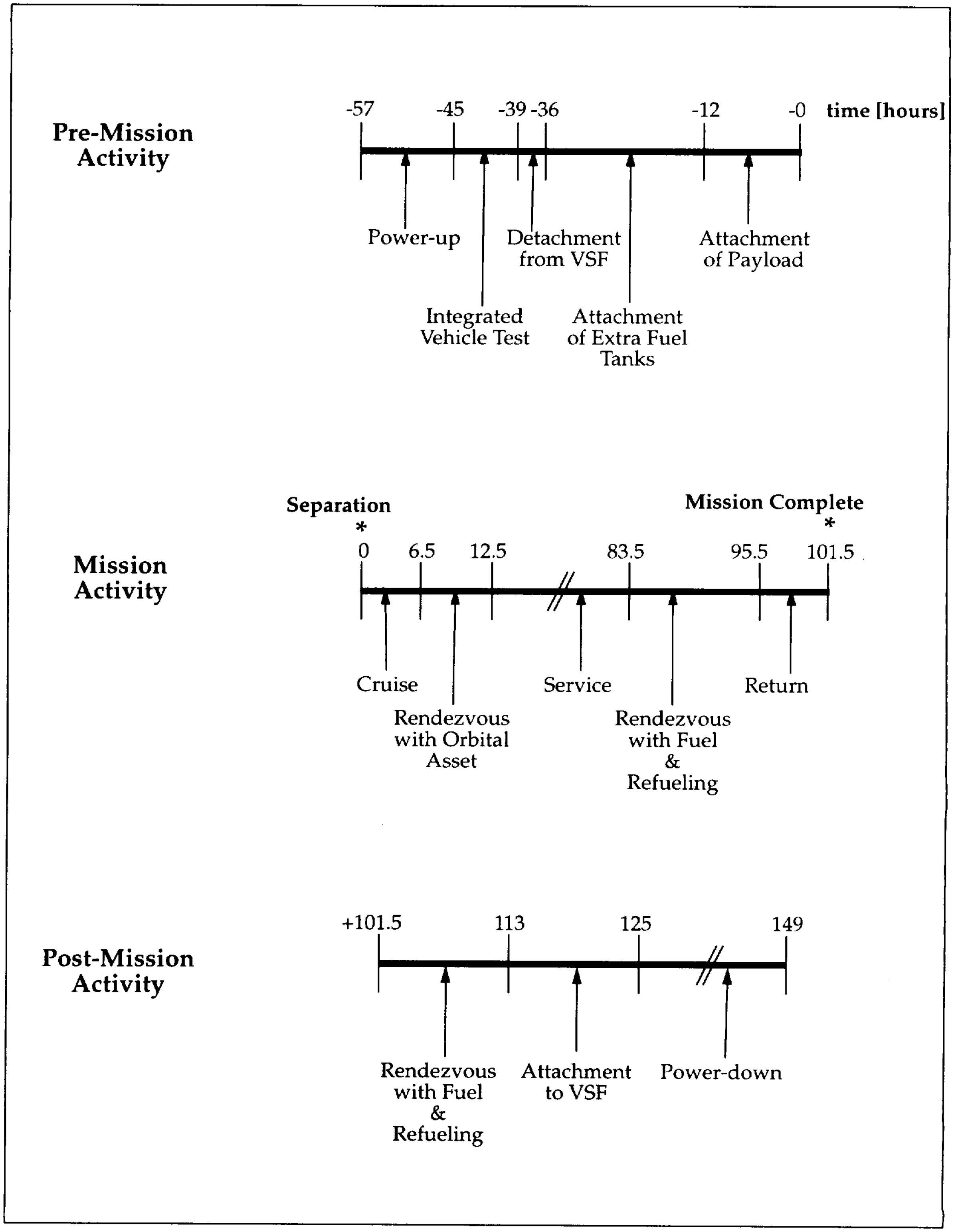

FIGURE 3.3 *Teleoperated Polar Mission Timeline.*

3.1.3 Teleoperated Mission to Polar Orbit

Figure 3.3 is a sample timeline of a teleoperated mission to service a space-based asset in LEO at a polar inclination. The vehicle could be equipped with fuel for a one-way mission and refueled for the return mission on orbit near the position of the orbital asset being serviced. This modified scenario, while increasing the mass fraction, greatly increases the complexity and risk of the mission. In Figure 3.3, 12 hours are incorporated into the Mission Activity segment of the timeline to allow for the additional rendezvous with the fuel tanker and refueling prior to the return trip. Although a polar mission is not a very likely scenario for Freebird, it is an option which shows the versatility of the system. A more typical representation of the teleoperated polar mission would simply omit the additional rendezvous in the Mission Activity segment of Figure 3.3 and would require that Freebird carry sufficient fuel to make a round-trip mission.

3.2 Missions in Geosynchronous Orbit

A mission to deploy a satellite in GEO equatorial orbit is a more challenging endeavor, but would be divided into mission phases similar to those given in Figure 3.4. Freebird must be prepared for a GEO mission in the same manner as for a LEO mission: the vehicle must be powered-up, tested, and detached from the VSF. From the VSF, the vehicle will effectively perform two rendezvous maneuvers to attach extra propellant tanks for the journey. Another 12 hours will be spent attaching the satellite to be deployed, or other cargo required. Then, the Pre-Mission Activity will be complete.

The Mission and Post-Mission Activity phases for a GEO mission are essentially the same as those for a teleoperated LEO mission, although the amount of time required for the approach and return depend on the mission and the specific mission destination.

3.3 Rescue Missions

One top-level requirement of Freebird is that it be capable of supporting a crew of 8 for 24 hours in an emergency situation. Out of this requirement arises the scenario for a rescue mission in which Freebird detaches from ISSA with the crew capsule and one crew member, performs a rendezvous with the Shuttle, for example, taking its 7 crew members aboard, and returns to ISSA. Figure 3.4 outlines the basic phases of Pre-Mission, Mission, and Post-Mission Activity which would be likely to be performed in a rescue mission. The first matter at hand is the declaration of the emergency. Once the emergency has been declared, the system would be powered-up from stand-by mode (which is not shown in Figure 3.4), the vehicle would be tested, and the best method of response to the emergency would be determined (e.g., route of travel). The astronaut would board the capsule to complete the Pre-Mission Activity.

The most notable aspect of the timeline in Figure 3.4 is the fact that the period of time that lies between the separation of Freebird from the VSF and the egress to ISSA is the maximum design limit of 24 hours. This time constraint allows a rescue time, once at the appropriate location, of over 10 hours.

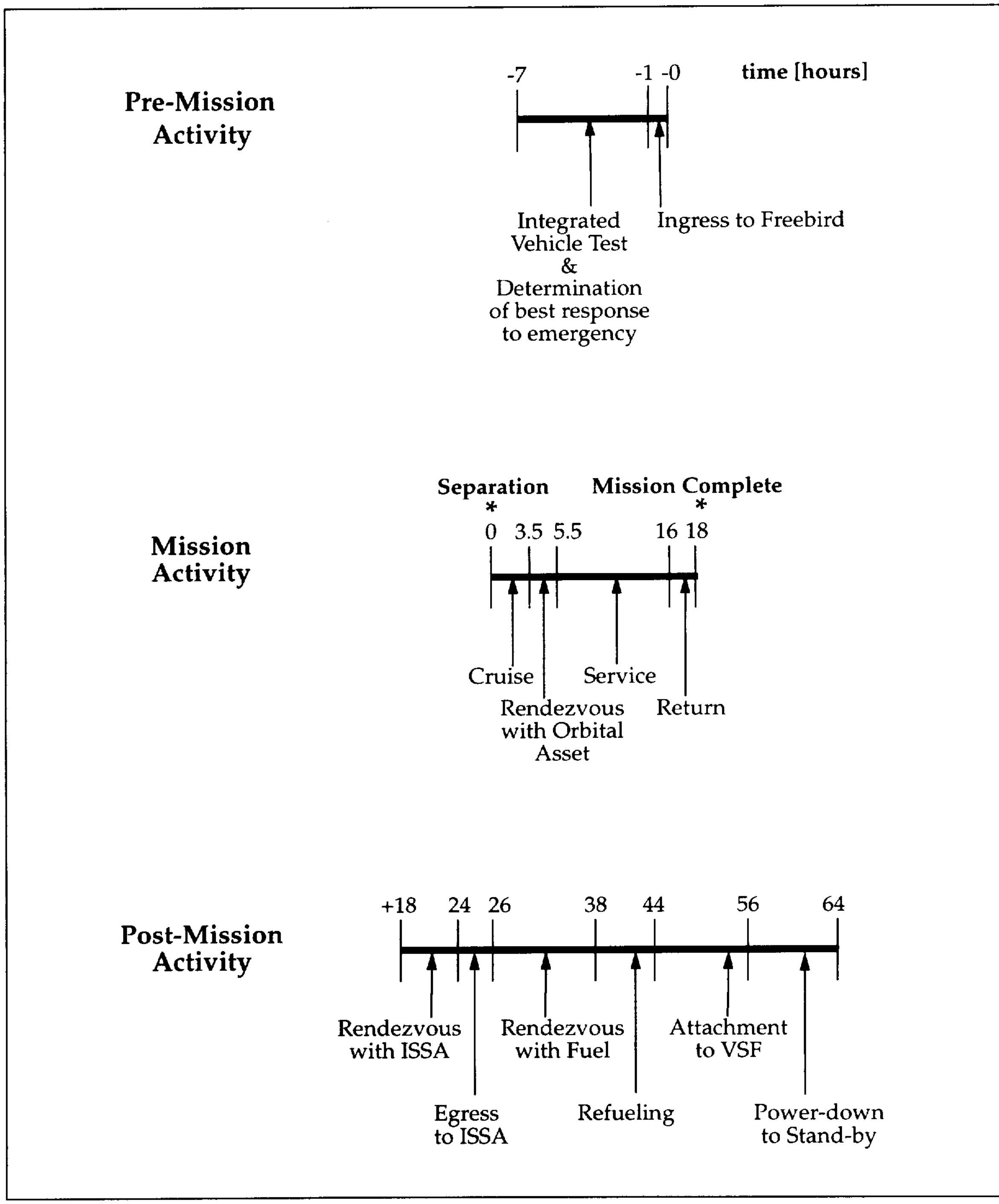

FIGURE 3.4 *Rescue Mission Timeline.*

Chapter 4

On Orbit Assembly and Maintenance

Critical issues for Freebird include on-orbit assembly methods and regular maintenance.

4.1 On Orbit Assembly

There are four major on-orbit manipulations which will need to be completed at various times during the vehicle lifetime: mating the crew module to the base unit, refueling the vehicle, attaching extra fuel packs, and replacing the propulsion packs. Mating and refueling will be required for nearly every mission. Replacing the propulsion packs will occur only once every twenty five missions.

Assembly will occur in a teleoperated manner. Instead of completely autonomous control, a ground-based human operator will manipulate the arms and thrusters for each rendezvous, docking and manipulation necessary.

4.1.1 Mating the Crew Module to the Base Unit

The crew module will be kept in storage at the Vehicle Storage Facility (VSF). The base unit will be remotely maneuvered to the correct attitude in the vicinity of the crew module. In order to assemble the full crewed vehicle, the crew module will be grappled by the base unit's Remote Manipulator System (RMS). In order to affect the mating, the RMS will be used to grapple the crew module and pull it onto the base unit. The interface is designed to allow for some error in the mating and to correct for minor misalignments. It is expected that the required accuracy will be similar to the docking with the International Space Station Alpha (a positional accuracy of 0.01m, velocity accuracy 0.01m/s, attitude to within 3°). Figure 4.1 shows the important steps of the process.

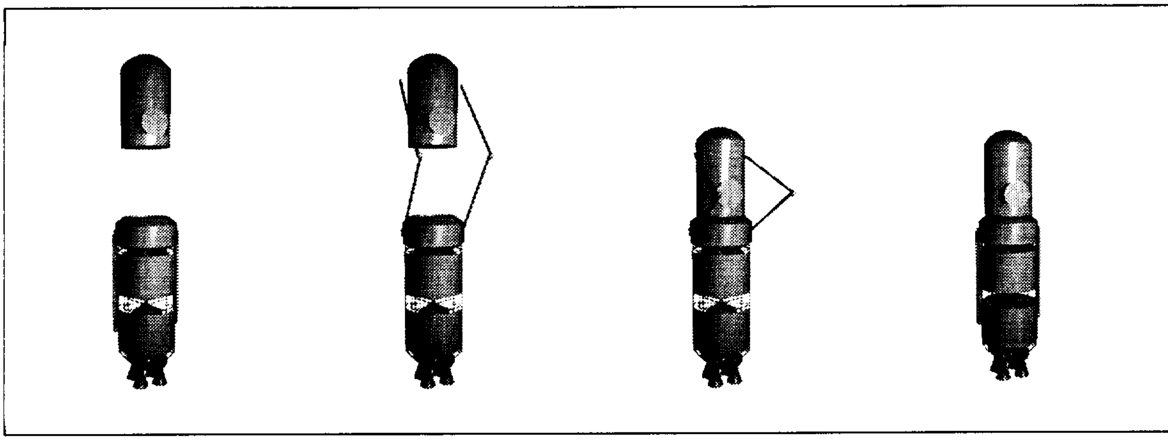

FIGURE 4.1 *Mating of the Crew Module and the Base Unit.*

4.1.2 Refueling the Freebird

Refueling will supply the Freebird with up to 18 000kg of fuel, enough for most nearby crewed and uncrewed missions. The first step in the refueling process will insert a fuel tank into an orbit nearby the ISSA on and the Freebird, or alternatively to wherever the Freebird needs to fuel.

Once the tank is in the orbit, Freebird will rendezvous with the tank and use the RMS for final docking maneuvers. The refueling pump will consist of a modification of the Thermal Venting System (TVS) that will remotely insert four fluid locks into the main tanks and fill them with fuel. Figure 4.2 shows the specific steps of this process.

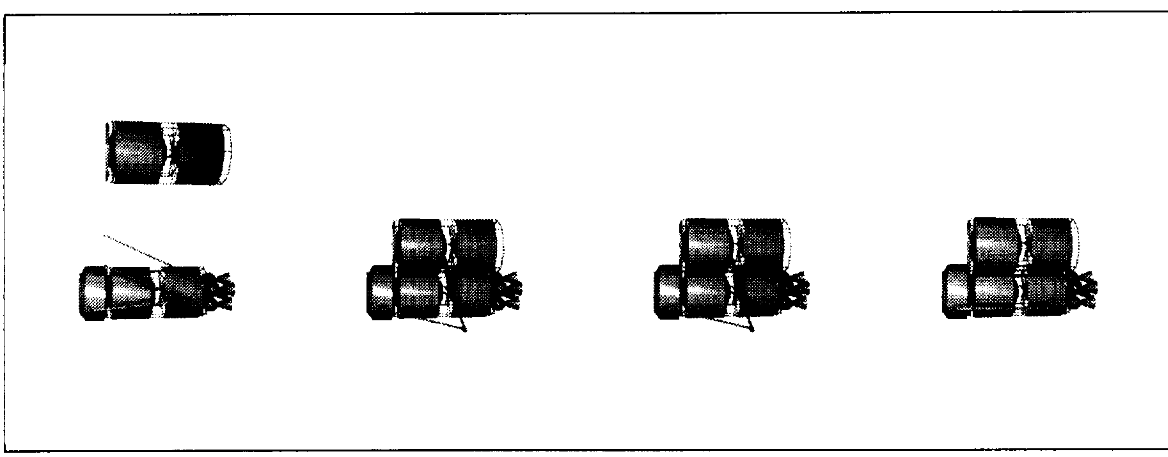

FIGURE 4.2 *Freebird Refueling Process.*

4.1.3 Attaching Extra Fuel Tanks

In a situation that requires Freebird to travel distances further than those attainable with a single full fuel tank, extra fuel tanks will be attached to Freebird. Unlike the refueling tanks, these are designed to travel with the vehicle and provide fuel along the transfer.

The Proton will be used to launch the fuel tanks. The specific launch configurations can be seen in Section 14.1.1 on page 243, Figure 14.2. Once the tank is in the orbit, Freebird will rendezvous with the tanks and use the RMS for final docking maneuvers. The fuel interface will be identical to the one used in refueling. Note that the arm configuration simplifies this operation slightly.

When the fuel has been expended (usually in GEO or polar orbit), the tanks are removed in a similar manner and moved clear of the vehicle with the RMS. The tanks can then be deorbited.

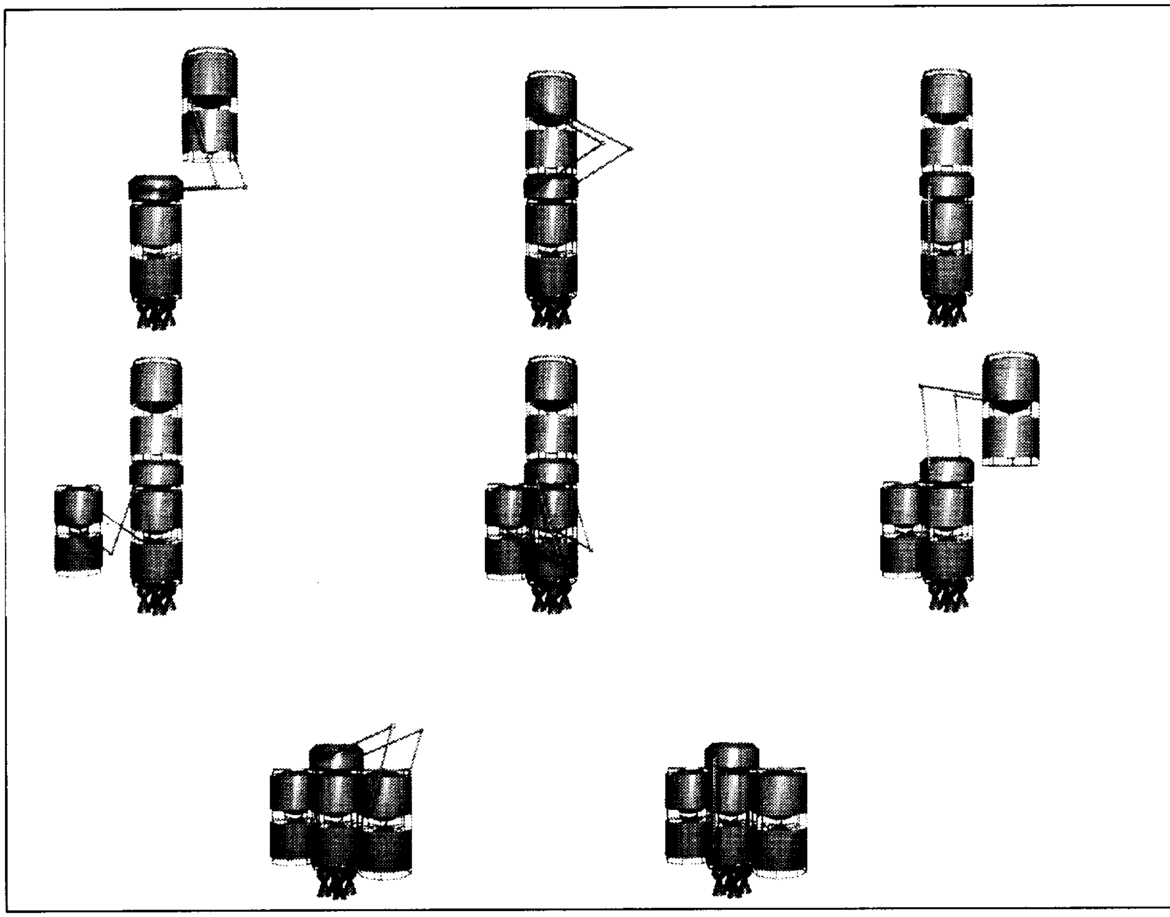

FIGURE 4.3 *Attachment of Extra Fuel Tanks to Freebird.*

4.1.4 Replacing the Propulsion Module

The most difficult on-orbit manipulation involves replacing the entire propulsion pack. The pack will need to be replaced when either the engines or fuel tanks have outlived their design life time, and to restore the state of the fuel tanks.

The Ariane V will be used to launch the fuel tanks. Launch configurations can be seen in section Section 14.1.1 on page 243, Figure 14.3. Once the new propulsion module is in the orbit, Freebird will rendezvous with the new pack and use the RMS for final docking maneuvers. The two propulsion packs will latch to each other as shown in Figure 4.4. The smartbox will then detach/unlatch from the expended propulsion module and use the RMS to move from the old module to the new module. Finally the "smartbox" will lower down and latch onto the new propulsion pack.

After checkout and verification of the interfaces, the old propulsion pack can be deorbited.

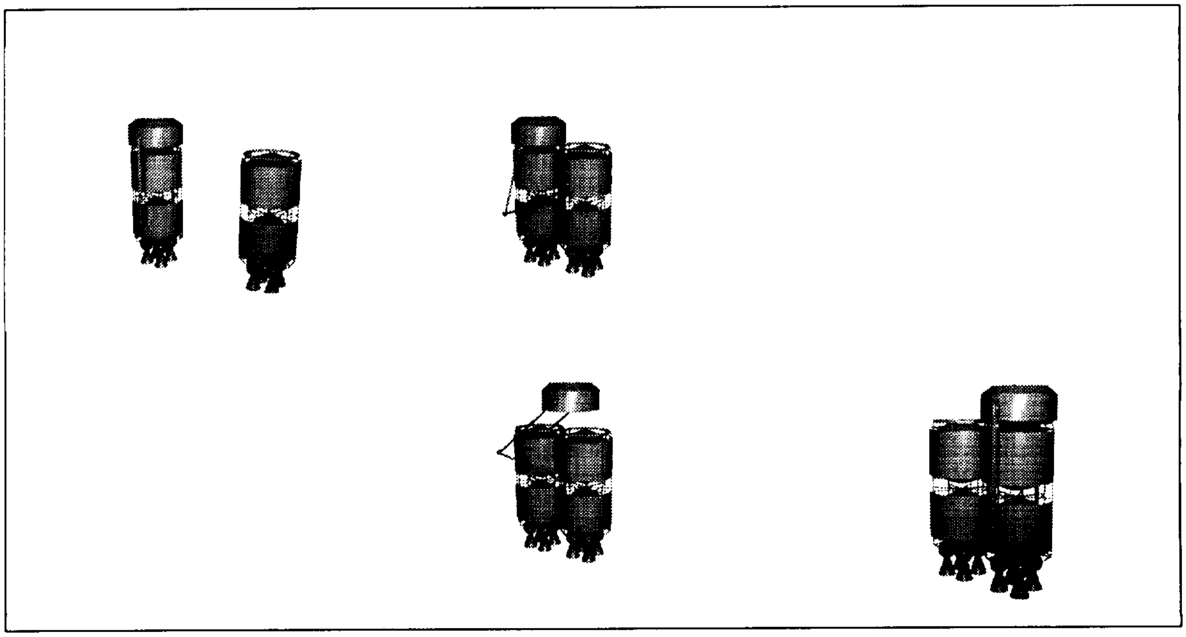

FIGURE 4.4 *Replacing the Propulsion Module.*

4.2 Maintenance

The vehicle will be maintained on-orbit to ensure safe and efficient performance for the duration of its 10 year lifetime. The maintenance needs of all components and subsystems must be met to enable the vehicle to perform under its intended mission scenarios. With proper maintenance, the vehicle will provide a safe environment for both the cargo and crew.

4.2.1 Maintenance Philosophy

The maintenance philosophy and approach is designed to maximize vehicle performance while minimizing cost. A preventive maintenance plan will be used for the vehicle and its modules. Under this system, maintenance is scheduled according to the results of failure analysis - therefore, components and subsystems will receive maintenance prior to failure. Cost is minimized by cutting back on the number of emergency repairs.

The vehicle will be maintained for the duration of its on-orbit life. Continued maintenance will help keep all subsystems functioning nominally, allowing the vehicle to be ready on a moments notice for rescue mission scenarios.

4.2.2 Maintenance Schedule

A maintenance schedule has been developed to ensure safe and proper vehicle performance during all missions. The schedule will account for the specific maintenance needs of each subsystem and will incorporate these needs in an orderly fashion to ensure efficient, low-cost maintenance.

Requirements

The vehicle's components and subsystems will be serviced or replaced according to their failure-analysis requirements. However, emergency maintenance using spare tools and parts will be available should the need arise.

The maintenance requirements of mature component and subsystem technologies are based upon the manufacturer's failure analysis. The requirements for less mature technologies have been developed during the design stage of the project.

- Power & Propulsion Systems: As part of the Power Pack, the RL10-A4 main engines must be replaced after 100 starts, or approximately every 25 missions. Assuming 8 missions per year, the engines must be changed evert 3 years. The main engine fuel tanks and the fuel cell tanks will be refilled following each mission.
- Environmental Control & Life Support System: To allow the astronauts to complete extravehicular activities (EVAs), the extravehicular mobility units (EMUs) will need to be serviced after each use. Expendable items (food, water, LiOH cannisters, activated charcoal, waste containers, and O_2) will be restocked after each mission.
- Command, Communications, Control, & Telemetry System: Regular diagnostics will be performed on the C^3T subsystem to ensure that it is functioning properly.
- Structure System: Because this subsystem was designed to survive the vehicle's 10 year lifetime it does not need traditional maintenance. However, regular visual inspections will be scheduled to verify that the structural integrity of the vehicle has not been compromised and that the interface connections are secure.
- Thermal System: This subsystem does not require traditional maintenance but will be visually inspected on a regular basis to ensure its integrity.
- Guidance, Navigation, & Control System: As part of the Power Pack, the reaction control system (RCS) maneuvering thrusters will be replaced after 25 missions, or approximately every 3 years.The RCS fuel tanks will be refueled after each mission.

Table 3.1 summarizes all of the subsystem maintenance requirements.

TABLE 3.1 *Subsystem Maintenance Requirements.*

Subsystem	Maintenance Requirements
P&P	RL10-A4 engines replaced every 25 missions; main engine and fuel cell tanks refueled after each mission
ECLSS	EMUs serviced after each use; expendables replaced after each mission
C^3T	Regular diagnostics
STRUCTURES	Regular visual inspection and interface verification
THERMAL	Regular inspection
GNC	RCS thrusters replaced every 25 missions; RCS tanks refueled after each mission

Schedule

Based on the individual subsystem requirements, a maintenance schedule has been developed and designed to minimize cost and time. Maintenance duties are combined where possible to allow for the demanding schedules of the crew and the Space Station personnel.

The inspections needed for the Thermal and Structures systems will be done in tandem - one maintenance team will perform an EVA to visually inspect the outer structure of the vehicle, while another team will perform the interface verifications. These inspections will occur before each mission. The RL10-A4 main engines and the RCS thrusters will be changed together with the vehicle's Power Pack. Refueling of the main engines, the maneuvering thrusters, and the fuel cells will also be done concurrently at the conclusion of each mission. While the refueling is taking place, another maintenance team will be removing waste items and replacing the expendable items on board the crew module. Finally, a series of diagnostics will be run every 24 hours to keep the C^3T subsystem in working order.

4.3 Maintenance Equipment

Specific equipment will need to be available to allow for proper vehicle maintenance. This equipment will primarily be stored at the Vehicle Storage Facility, and a small set will be stored on board the vehicle. The astronauts can meet emergency maintenance needs during a mission should they arise, but the bulk of maintenance will be performed at the Space Station.

4.3.1 Tools & Training

Astronauts will require the necessary tools and training to be able to meet the vehicle's maintenance needs. The tools needed to service or replace components and subsystems must be available at the Vehicle Storage Facility. A smaller set of tools must also be stored aboard the vehicle in case emergency repairs are required. On-board maintenance equipment will be kept to a minimum in order to satisfy vehicle mass constraints.

Both crew members and ISSA personnel must be trained to use these tools and to perform all maintenance procedures. All EVA maintenance tasks will require a team of two astronauts working together to complete a task. A team of two astronauts will not only decrease the time needed for maintenance duties, but will also provide the redundancy needed to ensure that the duties are performed correctly.

Subsystem Replacement Parts

Spare parts, components, and subsystems, must be available for replacement maintenance. These spares should be launched to orbit and kept in storage at the Vehicle Storage Facility. A few components will be kept on board the vehicle should the need for emergency repairs arise.

The spare subsystem parts kept aboard the vehicle will be limited to those needed to ensure the safety of the crew during manned missions. Should an emergency situation arise while the crew is on a mission, the astronauts will be able to maintain power, life support, and structural integrity with the on-board supplies. If a failure mode of a subsystem does not present a life-threatening situation for the crew, spare replacement parts for that subsystem will be stored at the Vehicle Storage Facility only. This maintenance arrangement guarantees the safety of the crew while minimizing vehicle mass.

A summary of the maintenance parts needed for each subsystem and their storage arrangements may be found in Table 3.2.

TABLE 3.2 *Spare Subsystem Parts.*

Subsystem	Spare Parts Aboard the Vehicle	Spare Parts At the Vehicle Storage Facility
P&P	Battery	Fuel Tanks, Fuel, RL10-A4 engines (Power Pack)
ECLSS	Food, Water, Gas Storage Tanks	Food, Water, Gas Storage Tanks
C^3T	None	None
STRUCT.	Interface Repair Equipment	Interface Repair Equipment
THERMAL	Radiation Tiles & Repair Equipment	Radiation Tiles
GNC	None	RCS thrusters (Power Pack)

4.4 Vehicle Storage

In an effort to reduce the weight of the vehicle, a storage facility was specified for Freebird. Using a storage facility enabled the shielding mass to be reduced by assuming that the vehicle will only need to endure the radiation and micrometeorite environment when it is actually preforming a mission. When the vehicle is not in use, it is stored inside the VSF. In addition to providing thermal and micrometeorite shielding, this facility will also provide the power consumed by Freebird in its stand-by mode.

Due to the limited docking space and power available on International Space Station Alpha (ISSA), it is recommended that the VSF also be designed to accommodate the thermal, and power needs of Freebird. A preliminary design similar to the VSF has been carried out by Martin Marietta[1].

4.4.1 Vehicles Storage Facility Location

Two options regarding the location of the VSF were considered: placement at the ISSA or a completely detached, free floating configuration. Locating the VSF at the station may be dangerous given the large amount of cryogenic fuel in storage aboard the vehicle. However, as a separate, free-flying structure, the VSF would require additional fuel to rendezvous with the station each time it had to perform a crewed mission. A free flying structure would also be more complicated since it would require its own thrusters to maintain orbit and attitude, a communications system, and some ground control.

The common berthing adapter (CBA) on the crew module will be used for docking with the VSF. This interface will provide Freebird with the power and communications interfaces required in the stand-by mode.

4.4.2 Stand-by mode

The purpose of the stand-by mode is to keep the navigation equipment and computer powered so that it does not have to be recalibrated every time the vehicle is to be used. This will allow quick response in an emergency situation. In addition, power can be drawn from the VSF instead of depleting the fuel cells store of oxygen and hydrogen. Most of the thermal control on-board Freebird will be passive, but the active temperature control system must be operational if the temperature of the critical components falls outside of the permissible range.

Preliminary calculations have determined that approximately 1kW of power is required to maintain the critical systems at 285°K. The computer and thermal venting system require less than 40W combined. Thus the driver of power during the stand-by mode is the temperature of the critical components.

References

1. Rohan Zaveri, Scott Geels, Erlinda Kiefel, Dan Uhlig and Benton Clark, "Assessment of a SSF Servicing Facility," *Proceedings of the International Space Engineering Conference III*, 1993.

2. Larson, Wiley J. and James R. Wertz, *Space Mission Analysis and Design*, Second Edition, Microcosm, Inc., 1992.

3. *Project Perseus*, 16.85 Space Systems Engineering, MIT, Spring 1993.

Chapter 5

Crew

Freebird can carry up to three crew on a nominal non-rescue mission. The most important issues surrounding the crew are selection, mission assignment and training.

5.1 Selection

The US astronaut corp and the Russian astronaut corp were considered as possible sources from which to draw Freebird crew. If crew were to be drawn from both countries, then two sets of new Freebird training facilities would need to be developed at each country's training complex. This would be costly. Therefore, crew should be drawn from only one country's pool of astronauts. Given the Russian discontent with making English the sole operating language, and taking into account that Freebird will most likely be manufactured by a US firm and thus having instruments most easily labeled for an English speaking crew, it was decided that the Freebird crew will be selected from among the US astronaut corp.

5.2 Mission Assignment

Following a precedent established in International Space Station Alpha (ISSA) crew selection and mission assignment, Freebird astronauts will be assigned to missions based on proven skill and past flight experience in extravehicular activity (EVA) and remote manipulation systems (RMS) operations. This will decrease the amount of training time necessary and increase the mission success probability.

5.3 Training

Crew training is important to the overall mission success. A skillfully trained crew with previous flight experience behind them can be the difference between meeting and falling short of a mission objective.

5.3.1 Philosophy

Training will be divided into two types: Advanced training and Increment training.

Advanced training will occur after an astronaut has completed the astronaut candidate training period, and become eligible for assignment to either a space shuttle or ISSA mission. Advanced training will be required for all astronauts who will eventually become Freebird crew, and is focused on providing astronauts with general skills necessary to operate the Freebird craft. These will include the following areas of instruction:

- Freebird housekeeping
- Maintenance procedures
- EVA procedures unique to Freebird
- RMS operations unique to Freebird
- Freebird rescue mission profiles for various spacecraft

Increment training will occur after an astronaut has been assigned to a specific Freebird mission. Increment training will be focused on providing astronauts with skills unique to a particular Freebird mission. The majority of increment training will include learning how to make specific repairs to a given satellite scheduled for maintenance on the mission to which an astronaut is assigned.

5.3.2 Scheduling

All Freebird training will run concurrently with shuttle and ISSA astronaut training. Increment training will commence at launch minus 14 (L-14) months, and will average 10 hours per week. This training will be integrated into the overall training flow for shuttle and ISSA missions which will involve the use of Freebird. Advanced training will commence at L-12 months, and will average an additional 10-20 hours per week depending on the nature of the training needed for the specific mission.

5.3.3 Instructors

The task of training Freebird astronauts would fall to the Spaceflight Training Division at Johnson Space Center. Instructors would be drawn from the current shuttle and ISSA instructor base. Freebird Basic Instructor Training and Certification guidelines would need to be established, and a new instructor base created.

5.3.4 Facilities

Existing as well as new training facilities will be utilized in preparing crews for Freebird missions. New facilities that would need to be constructed include:

- Full Freebird mock-up trainer
- Freebird RMS simulator (computer-simulated space environment)
- Part-task trainers

Existing facilities which would be utilized for Freebird training include:

- Weightless Environment Training Facility
- Precision Air-bearing Floor
- KC-135 Parabolic Flight Trainer
- Other facilities associated with EVA training

Chapter 6

Contingency

Contingency must be provided for so that the mission requirements will be met in the case of unforeseen or unpredictable occurrences. Contingency is added to the vehicle design, over and above the necessary margin of safety. Contingency must be considered so that chance or unforeseen events are accounted for. In fact, any part of the design process which uses some modeling technique will have inefficiencies in the conception. Although providing contingency for every mishap is difficult, percentage contingencies for appropriate parameters (which will be discussed below) add coverage to the design margins which have been incorporated into designs on the subsystem level.

6.1 Contingency Determination

The basis for contingency figures, like reliability data, is experience with existing or past systems. With more experience in dealing with a component, the chance of encountering unforeseen occurrences is less likely. For example, only a relatively small contingency would have to be provided for human systems on missions to LEO because extensive studies and testing have been performed for Space Shuttle and other crewed missions.

Contingency data also relies on the criticality of the design component or subsystem. For example, a high contingency should be provided for the link budget for Freebird, because if the system did not transmit the necessary power, it would be a critical failure. Also, the criticality of the power budget drives up its contingency. In the same manner, the mass budget will have a high contingency, because this is the first design stage for Project Freebird. The maturity of the design factors into the contingency in this way.

When components are purchased from outside source accurate specifications should be given and are generally more reliable than any predictions based on modeling or empirical equations. The propellant budget would require less contingency than the mass budget because the engines and fuel tanks are bought from outside sources. However, inefficiencies still need to be accounted for. Contingency must be provided for unexpected maneuvers or unknown maneuver accuracy.

6.2 Allocated Contingencies

Over and above all of the subsystem contingencies, which were applied at the component level, a system level contingency was added to Freebird's design. Table 6.1 summarizes the percentage contingencies for various budgets. The actual figures are based on the information presented above, as well as on previous space operations. Detailed summary budgets including contingencies are located in Appendix A on page 299, Appendix B on page 305, and Table 8.3 on page 161.

TABLE 6.1 *Allocated Budget Contingencies*

	Contingency
Mass Budget	5 to 25% [a]
Power Budget	5 to 25% [a]
Link Budget	4dB
Propellant Budget	5 to 17% [b]
System Budget	5%

a. Varies for specific component.

b. Depends on specific mission.

C-4.

Chapter 7

Operations, Maintenance and Support Costs

Operations, maintenance and support can be a considerable portion of any system's budget, particularly so when one considers the complicated nature of a space vehicle system. The budget can be broken down into two phases: initial costs to produce the required hardware and continuing costs which are often allocated on a yearly or per mission basis.

7.1 Delivery to Orbit

The space vehicle designed by the Freebird Project must be delivered to orbit. Because of the high cost of launch vehicles, orbital delivery is a significant cost factor. Fortunately multiple launch systems are available for use. The vehicle will be delivered in two separate launches. The crew module will be launched in the Proton C at a cost of $35 million; the base unit will be launched in the Titan IV at a cost of $300 million. Total launch costs are therefore estimated at $335 million.

7.2 Refueling

For each Freebird mission fuel must be delivered to the Space Station. Between one and three deliveries will be required depending on the mission. Each fuel delivery involves a single Proton C launch of approximately 20 000 kg of LH2 and LOX. Cost of delivery includes the launch cost, $35 million, and the estimated costs of the fuel and fuel tanks, $12 million. Therefore the estimated cost per refueling mission, not including orbital support, is approximately $50 million.

If Freebird is not utilized often it may become necessary to deliver an extra fuel tank to compensate for cryogenic fuel leakage/venting. If this is true, an additional refueling mission may be required at periodic intervals. To address that possibility an additional $50 million will be added to the expected yearly budget.

7.3 Maintenance and Repair

Throughout its lifetime, Freebird will require some degree of maintenance. Both scheduled and unscheduled maintenance can be costly.

7.3.1 Engine Replacement

The main engines for Freebird are currently qualified for 20 starts and a burn time of 3000 seconds. Money has been allocated to expand the qualification to 100 starts over 5.25 hours of burntime. Assuming Freebird will be flying only GEO missions (four burns per mission) as a worst case scenario, the vehicle must be refitted every 25 missions. The engines will need to be replaced approximately once every three years based on an expected eight missions per year. Therefore the entire booster pack will need to be replaced at least four times during Project Freebird's lifetime.

Replacement costs will be appreciable and must be taken into account. The booster pack hardware totals approximately $107 million including verification and validation. Delivery will most likely be carried out by the Proton C at a cost of $35 million. Therefore an engine replacement mission will total approximately $142 million. This yields a per mission allocated cost of $5 million, not including depreciation or possible increasing (or decreasing) cost of materials.

7.3.2 Unscheduled Repair

Additional maintenance and repairs may become necessary during the space vehicle's 10 year lifetime. While this possibility is important to consider it is impossible to quantify it at this time. A set of maintenance and repair materials will be kept at the station in the event preventative maintenance or repair is necessary and possible. Additional parts and modules may be delivered to the station should the need arise.

7.4 Support

Project Freebird will require extensive ground and orbital support. Some of this support will be continuous while some will be mission specific. Support costs include everything associated with Freebird after it is placed in orbit. Support costs can accumulate quickly and are usually a function of overall project complexity.

7.4.1 Ground Support

Ground Support includes the facilities, equipment, software, logistics, and management which keep Freebird in touch with Earth. These costs can often be modelled as a function of software costs which are a potential measure of system complexity. The estimated support budget for the ground includes communication access as well. By utilizing this model and including a complexity factor to account for the possibility of ground-based teleoperation a total up-front cost of approximately $160 million was derived. Continuation/operations cost can be expected to add $30 million/year (approximately 15% of initial cost). [1]

7.4.2 Orbital Support

Orbital support includes communications equipment and access to and through Space Station Alpha as a result of Project Freebird's existence. It also includes the estimated cost to train astronauts to perform Freebird missions (i.e. the cost to have experienced labor in orbit). This labor cost will exist independent of Project Freebird and must be carried by NASA.

Orbital Support also includes the existence of a Vehicle Storage Facility. Such a facility must be designed, constructed and delivered to maintain the vehicle's usefulness. The VSF is beyond the scope of this report but should not be ignored. For the purpose of a first cut cost estimation it can be expected to fall in the range of other powered but unmanned modules contained in the design of ISSA.

7.5 Summary of Operations, Maintenance and Support Budget

The total expected up-front costs of Operations, Maintenance and Support is $300 million. Cost/mission is approximately $58 million for a LEO mission and $162 million for GEO.Yearly prorated costs are estimated at $845 million. This number is based on an assumed eight missions/year: five (5) LEO, one(1) Polar, two(2) GEO. It also includes the possibility of an additional required refueling mission. (see Table 7.1)

TABLE 7.1 *Operations, Maintenance and Support Costs (All figures FY94 in millions)*

Operation	Initial Cost	Cost/Operation	Cost/Year
Vehicle Delivery to Orbit	$135	N/A	N/A
Refueling (LEO=1, GEO=3)	N/A	$50	$50
Engine Refit (once/25 missions)	N/A	$125 / 25	N/A
Ground Support	$160	N/A	$30
Orbital Support	unknown	N/A	N/A
Total LEO mission	N/A	$58	N/A
Total GEO/Polar mission	N/A	$162	N/A
Total Initial Cost	$300		
Total Yearly Cost:(8 missions/year)			$845

References

1. Larson, Wiley J., and Wertz, James R. (editors) 1992, Space mission analysis and design, 2nd ed.,Torranace California: Microcosm, Inc.

Chapter

8

Funding and International Political Concerns

A successful technical project must not only be backed with good engineering, but it must also contain innovative approaches to funding and operating in the current international political environment.

8.1 Funding

Freebird costs can be broken down into two areas: up-front costs, which total \$1.5 billion fiscal year 1994 (FY94), and yearly prorated operating costs of \$845 million FY94.

8.1.1 Up-front Costs

Up-front costs for the total design, production, testing, and launch of Freebird account for \$1.5 billion FY94 (see Section 12.4 on page 232). Since both commercial satellite manufacturers and the National Aeronautics and Space Administration (NASA) have vested interests in seeing Freebird built, these organizations will be expected to finance these aspects of the project. Freebird would serve as a model joint venture between government and industry, efficiently utilizing existing government research and testing facilities, and industry management and integration experience.

8.1.2 Operating Costs and Charges

Missions can be placed in three cost categories depending on the type of mission being contracted.

- Low Earth Orbit On-site Repair Mission
  Costs for this type of mission include the procurement and launch of fuel and expendables, and a fraction of yearly maintenance and operating costs for the Freebird system. These total approximately \$60 million FY94. A user of Freebird repair services will be charged approximately 50% of the worth of the asset being repaired. Only satellites with a net worth of greater than \$120 million will be considered. The user will be responsible for contracting with NASA for astronaut services.

- Polar Teleoperated Repair Mission and Geostationary Orbit Teleoperated Repair Mission

  Costs for this type of mission include the procurement and launch of fuel, as well as a fraction of yearly maintenance and operating costs for the Freebird system. These total approximately $165 million FY94. A user of Freebird repair services will be charged approximately 50% of the worth of the asset being repaired. Only assets worth $330 million or greater will be considered.

- Rescue Mission

  Costs for this type of mission include the procurement and launch of fuel and expendables, as well as a fraction of yearly maintenance and operating costs for the Freebird system. The total of these costs, as well as the respective additional charges that may be levied to users, will be determined as needed on a case by case basis.

8.2 International Political Concerns

Since Freebird will be a U.S.-owned space vehicle, and the U.S. is a signatory to the United Nations Outer Space Treaty (OST) of 1967, Freebird must conform to the provisions outlined in the OST. Relevant Treaty sections pertaining most directly to Freebird operations include Articles V, VI, VIII, IX, XI, and XIII. These Articles discuss the issues of astronaut protection, state responsibilities, benign motives, international cooperation, and sovereignty over launched objects [1].

8.2.1 Repairing Foreign Satellites

International agreements must be drawn up on a case by case basis for the repair of non-U.S. assets, in accordance with the OST; particularly Article VIII. Article VIII states that "A State party to the treaty on whose registry an object launched into outer space is carried shall retain jurisdiction and control over such object, and over any personnel thereof, while in outer space. Ownership of objects launched into outer space,...and of their component parts, is not affected by their presence in outer space, ... or by their return to earth" [1]. This implies that any object launched into space remains the property of the "nation group" which launched it. Thus, the U.S. would need to secure an official understanding between itself and the party owner of the satellite that will be repaired by U.S. astronauts in a U.S. vehicle, in order to avoid breaches of the OST.

8.2.2 Rescuing Stranded Astronauts

Though understandings will need to be secured in order to have Freebird repair a foreign-owned space asset, no such pre-arranged agreements would be necessary in the event of rescuing stranded foreign astronauts. Article V of the OST paves the way for such missions: "States parties to the treaty shall regard astronauts as envoys of mankind in outer space and shall render to them all possible assistance in the event of accident, distress, or emergency ... In carrying on activities in outer space, ... the astronauts of one State party shall render all possible assistance to the astronauts of other States parties" [1]. Hence, advance permission from state parties is not necessary for a rescue mission to assist foreign astronauts in an emergency situation.

8.2.3 Contracting for Foreign Launch Services

Today, the problems and tedious procedures involved in launching on a foreign launch vehicle are waning with the end of the Cold War. In order to contract a launch on a foreign launch vehicle, the appropriate joint venture marketing organization must be contacted. For launching the crew module and the extra fuel supply, the Proton C vehicle was chosen as the prime launch vehicle. Lockheed-Krunichev Enterprises (LKE) has marketing and contracting rights for the Proton series of launch vehicles. For launching the propulsion module and replacement propulsion modules, the Ariane 5 was chosen as the prime launch vehicle (see Chapter 1 on page 255). Arianespace, Inc. has the marketing and contracting rights for the Ariane series.

For launching U.S. space assets on foreign launch vehicles, it is first necessary to obtain an export license. This request for an export license initially must be approved by the Department of Commerce (DoC) and the Department of Defense (DoD). After DoC and DoD approval, the request goes to the Department of State, which ultimately grants the export license. In addition, the space asset must pass all current technology export regulations in order to secure the license.

References

1. Fawcett, J.E.S. 1984. *Outer Space: New Challenges to Law and Policy.* Oxford, Great Britain: Clarendon Press.

Chapter
9

Limitations and Future Growth

At this step in the design process, Freebird has inherent limitations to vehicle and mission performance. Safety considerations, the technology available, the space environment, and orbital dynamics are among the issues that impose restrictions on the design. The results of these issues are vehicle restrictions and a range of missions which is more narrow than that implied by the primary mission objectives. To minimize or overcome these limitations, the next level of design iterations should be performed. The future growth of Freebird depends upon modifying or extending the current design to preclude these limitations.

9.1 Range of Missions

Various missions cannot be accomplished within the operating envelope established for the primary mission objectives. The destination location, rescue mission considerations, the life support system design, and the potential for radiation damage act to restrict the range of missions that Freebird is anticipated to accomplish.

9.1.1 Destination Location

The location of, say, the satellite to be deployed (or, simply, the desired destination) relative to the location of International Space Station Alpha (ISSA), where the vehicle will be stored, can be a limitation to the range of missions possible. Depending on the destination location, a mission could require a large Δv, and, as a result, require excess fuel. Missions which would require excess fuel--more fuel than the attachable fuel tanks can provide--are generally not feasible. With this in mind, it has been determined that operations are limited to the envelope of a 2.8 mass ratio (see Section II, Chapter 1). Examples of missions which are generally not feasible are extended missions, missions to retrograde orbits, and equatorial missions in low earth orbit. One possibility of widening the range of possible missions is to send additional fuel to the location of the space-based asset being tended or repaired. In other words, an alternative option is to launch a fuel vessel from Earth to the destination location so that the vehicle can be fueled for a one-way trip prior to separation and then be refueled on orbit for the return trip. This approach, however, greatly increases the complexity of the mission because of the additional rendezvous with the fuel vessel. Certainly, modifications would have to be made in the current design to incorporate the possibility of the missions described above.

9.1.2 Rescue Mission Considerations

The secondary mission objective of Freebird requires the vehicle to be capable of rescue missions. Rescue operations, however, are restricted to inclinations between 34.6° and 78.6° because of mass ratio considerations (Section II, Chapter 1). Rescue operations would proceed with the standard fuel tanks and would thereby be limited. In addition, if Freebird is already in use--on another mission--it cannot be called upon for a rescue mission. To avoid such a situation, it might be advantageous to have two operating vehicles to increase the likelihood of availability for rescue purposes.

9.1.3 Life Support System

In non-rescue scenarios, Freebird's life support system is equipped to sustain 3 astronauts for 4 days. This design limit imposes restrictions on the types of missions that Freebird may accomplish. Within the scope of the current design, for example, a crewed lunar mission would not be possible. The amount of time a lunar mission would require would extend beyond the design limit established for maintaining the life support system. The most reasonable alternative to this limitation is to consider autonomous or teleoperated lunar missions, depending on the various tasks required.

9.1.4 Radiation

Carrying humans, a top level requirement of the system, has implications beyond the design limit for the life support system. Specifically, the potential for radiation damage to humans on Freebird missions is great (Section I, Chapter 2). Because astronauts are more readily adaptable than the Remote Manipulation Systems (RMS), say, or other robotic elements, Freebird repair missions would most often depend on astronauts to conduct repairs during EVA. While engaged in EVA, however, an astronaut would be exposed to levels of radiation dependent upon the inclination and altitude of orbit. Conducting an EVA on a mission to geosynchronous orbit, for instance, is simply not feasible because of the radiation environment. Astronauts in low earth orbit at polar inclinations would likewise be unable to withstand the radiation. Due to the harsh radiation environment that Freebird is likely to encounter, ECLSS, the environmental control and life support system, has set a maximum safe altitude of 600 km for EVA or crewed operations.

9.2 Vehicle Restrictions

Certain restrictions apply to the vehicle itself, rather than to the missions that the vehicle is capable of accomplishing. Specifically, the size of the payload that Freebird can carry is limited. While it is possible for Freebird to carry cargo, such as a small satellite to be deployed, the mass that Freebird can carry is heavily dependent upon the orbit that the satellite is to be transported to. Similarly, it is unlikely that Freebird will be conducting missions in which it retrieves satellites from orbit and ferries them back to ISSA for repairs. However, missions such as this are possible but are clearly limited by the mass of the asset and the inclination and altitude of its orbit. An alternative to this scenario would be for Freebird to attach a booster pack to the satellite to transport it to the desired destination.

The need to protect Freebird against micrometeorite debris and radiation damage has led to the incorporation of a Vehicle Storage Facility (VSF) into the design operations environment. Without the shielding of the VSF while not in operation, the lifetime of the vehicle is estimated to be only 2 years. Therefore, the VSF is a significant element of the design and should be further developed.

Another issue which will not necessarily limit vehicle performance but which should be addressed is fuel boil-off. The amount of cryogenic fuel that is expected to boil off is 2 - 5% per month (Section II, Chapter 5). Although this disadvantage could be avoided by using another type of fuel, the high specific impulse that cryogenic fuel yields is worth the amount of boil-off anticipated. Most of the missions that Freebird is likely to complete will not be severely limited by this effect. And, by maximizing the number of possible missions per year, the effect can be minimized.

A rather obvious vehicle restriction is the fact that Freebird was designed for a lifetime of 10 years. Some components were also designed to be replaced every 25 missions (approximately 3 years), such as the RL10-A4 engines and the Reaction Control System (RCS) thrusters (Section III, Chapter 6). Technological advancement and growth will most probably provide solutions or alternatives that will increase the efficiency and effectiveness of current endeavors and perhaps extend vehicle lifetime limitations.

Further design iterations or operational techniques should be performed to minimize or overcome these limitations. Alternatives which could make it possible to avoid many of these limitations are already available and are discussed above. Clearly, future research should be conducted to modify or extend the current design to preclude these limitations.

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Part IV: Pages 299-332
Appendices

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Appendix A

Detailed Mass Budget

TABLE A.1 *Component Contingencies and Masses*

Component	Subsystem	Quantity	Mass Each [kg]	Contingency	Adjusted Mass [kg]
Crew: Pressure Vessel	Structures	1	480	20%	576
Crew: Internal Structure	Structures	1	125	0%	125
Crew: Rad/Debris Shield	Structures	1	711.5	20%	853.8
Crew: Common Berthing Adaptor	Structures	1	130	20%	156
SB: Intermodule Interface	Structures	2	100	20%	240
SB: Remote Manipulator System	Structures	2	275	20%	660
SB: Primary/Truss	Structures	1	93.5	20%	112.2
SB: Internal	Structures	1	10	20%	12
SB: Rad/Debris Shield	Structures	1	218.3	20%	262
PP: Primary/Truss	Structures	1	216	20%	259.2
PP: Rad/Debris Shield	Structures	1	222	20%	266.4
Base: Heat Pipes	Thermal	1	15	10%	16.5
Base: Radiators	Thermal	2	0	10%	0
Base: Insulation	Thermal	1	3.3	10%	3.6

TABLE A.1 *Component Contingencies and Masses*

Component	Subsystem	Quantity	Mass Each [kg]	Contingency	Adjusted Mass [kg]
Base: Cold Plates	Thermal	11	0.5	10%	6.3
Base: Phase Change Devices	Thermal	1	2	10%	2.2
Base: Radiator Heaters	Thermal	2	.1	10%	0.2
Base: RCS Propellant Heaters	Thermal	64	0.1	10%	7
Base: Louvres	Thermal	20	0.1	10%	2.2
Base: Louvre Mechanical System	Thermal	4	2	10%	8.8
Crew: Insulation	Thermal	1	2.2	10%	2.4
Crew: Cold Plates	Thermal	7	0.5	10%	4.1
Crew: Coolant Pump	Thermal	2	2	10%	4.4
Crew: Coolant Piping	Thermal	1	10	10%	11
Crew: Radiator Heaters	Thermal	4	0.1	10%	0.4
Crew: Louvres	Thermal	20	0.1	10%	2.2
Crew: Louver Mechanical System	Thermal	4	2	10%	8.8
Crew: Air Recycling Unit	Thermal	1	35	10%	38.5
Crew: Radiator	Thermal	2	0	10%	0
Thermometer	Thermal	30	0	10%	0
DCS with Sun Fans	GNC	4	4.5	15%	20.9
Rendezvous Radar Transponder	GNC	2	10	25%	25
MIMU	GNC	2	5.1	5%	10.8
MANS	GNC	3	0	15%	0
MAPS	GNC	3	0	20%	0
Star Tracker	GNC	2	25	5%	52.5
LNS	GNC	2	20	30%	52
Docking Camera	GNC	2	5	20%	12
TDRSS Transponder	CCCT	3	6.3	5%	20
EVA Transponder	CCCT	2	3	20%	7.2
Power Amplifier	CCCT	3	5	20%	18
Parabolic Antenna	CCCT	2	2.5	20%	6
S-Band Omni Antenna	CCCT	2	1	20%	2.4
EVA Omni Antenna	CCCT	2	1	20%	2.4

TABLE A.1 *Component Contingencies and Masses*

Component	Subsystem	Quantity	Mass Each [kg]	Contingency	Adjusted Mass [kg]
Internal Communication Network	CCCT	2	1	20%	2.4
Harness	CCCT	1	10	20%	12
Processors (Harris)	CCCT	3	1.5	5%	4.7
Data Recording Unit	CCCT	1	2	20%	2.4
LOX Propellant Tank	Propulsion	1	79	20%	94.8
LH2 Propellant Tank	Propulsion	1	107	20%	128.4
Plumbing - Valves & Pipes	Propulsion	1	230	10%	253
Propellant Management Device	Propulsion	2	20	10%	44
Interfaces	Propulsion	2	30	10%	66
RL10A-4 Engine	Propulsion	4	168	5%	705.6
Thermal Venting System	Propulsion	2	30	20%	72
RCS Tanks, Valves, Pipes	Propulsion	1	150	10%	165
RCS Fuel	Propulsion	1	550	10%	605
Orbital Maneuvering Thrusters	Propulsion	2	10.3	5%	21.5
Primary RCS Thrusters	Propulsion	26	3.7	5%	101
Fuel Cell Powerplant	Power	2	77.1	20%	185.1
Power Conditioner	Power	2	25	20%	60
EPS Wire Harness	Power	2	50	10%	110
Fuel Cell Tank (H)	Power	1	34	10%	37.4
Hydrogen for Fuel Cells	Power	1	14.6	10%	16
Fuel Cell Tank (O)	Power	1	31.8	10%	34.9
Oxygen for Fuel Cells	Power	1	122.3	10%	134.6
EVA Tools	ECLSS	1	100	10%	110
Misc. Other Cargo	ECLSS	1	200	0	200
Extravehicular Mobility Unit	ECLSS	3	123.3	5%	388.4
EMU Batteries	ECLSS	10	4.4	5%	45.7
Atmosphere Control System	ECLSS	1	--	10%	0
- Cabin Air Storage System	ECLSS	1	68	10%	74.8
-Pressurized Oxygen Tank	ECLSS	2	17.5	10%	38.5

TABLE A.1 *Component Contingencies and Masses*

Component	Subsystem	Quantity	Mass Each [kg]	Contingency	Adjusted Mass [kg]
- Oxygen In Tanks	ECLSS	2	19	10%	41.8
- CO2 Removal System	ECLSS	1	10	20%	12
- LiOH Cannister	ECLSS	5	5	10%	27.5
- Sensors	ECLSS	11	0.5	10%	6.6
- Flow Valves	ECLSS	6	1	10%	6.6
-Cabin Fans	ECLSS	2	3	20%	7.2
Pressure Suit	ECLSS	1	10	20%	12
Waste Disposal System	ECLSS	1	5	20%	6
Food, Water, Containment System	ECLSS	1	80	10%	88
Portable O2 System	ECLSS	8	2	20%	19.2
Fire Detection Sensors	ECLSS	6	0.5	20%	3.6
Fire Extinguishers	ECLSS	4	4	20%	19.2
Medical Kit	ECLSS	1	2	20%	2.4
Human Interface Controls	ECLSS	1	20	20%	24
Internal Lights	ECLSS	4	1.3	20%	6
Cargo Bay Lighting	ECLSS	2	20	20%	48
Restraints, Seats	ECLSS	8	18.8	20%	180
Astronaut	ECLSS	3	70	5%	220.5
Total					**8,314.3**

TABLE A.2 *Subsystem Totals by Module*

Subsystem	Total Mass [kg]	Percentage of Total	Base Unit Mass [kg]	Crew Module Mass [kg]
Structures	3,522.6	42.3%	1,811.8	1,710.8
Thermal	118.8	1.4%	46.9	71.9
GNC	173.2	2.1%	173.2	--
CCCT	77.5	0.9%	77.5	--
Propulsion	2,256.3	27.1%	2,256.3	--
Power	578	7.0%	578	--

TABLE A.2 *Subsystem Totals by Module*

Subsystem	Total Mass [kg]	Percentage of Total	Base Unit Mass [kg]	Crew Module Mass [kg]
ECLSS	1,588	19.1%	--	1,588
Total	**8,314.4**		**4,943.7**	**3,370.7**

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Appendix B

Detailed Power Budget

TABLE B.1 *Component Average & Peak Power*

Component	Subsystem	Quantity	Average Power Each [Watts]	Average Power [Watts]	Peak Power Each [Watts]
Crew: Pressure Vessel	Structures	1	0	0	0
Crew: Internal Structure	Structures	1	0	0	0
Crew: Rad/Debris Shield	Structures	1	0	0	0
Crew: Common Berthing Adaptor	Structures	1	?	0	?
SB: Intermodular Interface	Structures	2	10	24	200
SB: Remote Manipulator System	Structures	2	800	1920	1200
SB: Primary/Truss	Structures	1	0	0	0
SB: Internal	Structures	1	0	0	0
SB: Rad/Debris Shield	Structures	1	0	0	0
PP: Primary/Truss	Structures	1	0	0	0
PP: Rad/Debris Shield	Structures	1	0	0	0

TABLE B.1 *Component Average & Peak Power*

Component	Subsystem	Quantity	Average Power Each [Watts]	Average Power [Watts]	Peak Power Each [Watts]
Base: Heat Pipes	Thermal	1	0	0	0
Base: Radiators	Thermal	2	0	0	0
Base: Insulation	Thermal	1	0	0	0
Base: Cold Plates	Thermal	11	0	0	0
Base: Phase Change Devices	Thermal	1	0	0	0
Base: Radiator Heaters	Thermal	2	0	0	0
Base: RCS Propellant Heaters	Thermal	64	?	0	?
Base: Louvres	Thermal	20	0	0	0
Base: Louvre Mechanical System	Thermal	4	0	0	10
Crew: Insulation	Thermal	1	0	0	0
Crew: Cold Plates	Thermal	7	0	0	0
Crew: Coolant Pump	Thermal	2	17.5	38.5	25
Crew: Coolant Piping	Thermal	1	0	0	0
Crew: Radiator Heaters	Thermal	4	0	0	11.3
Crew: Louvres	Thermal	20	0	0	0
Crew: Louver Mechanical System	Thermal	4	0	0	2.5
Crew: Air Recycling Unit	Thermal	1	50	55	75
Crew: Radiator	Thermal	2	0	0	0
Thermometer	Thermal	30	0	0	0
DCS with Sun Fans	GNC	4	5	23	5
Rendezvous Radar Transponder	GNC	2	30	75	75
MIMU	GNC	2	18	37.8	18
MANS	GNC	3	0	0	0
MAPS	GNC	3	0	0	0
Star Tracker	GNC	2	1	2.1	7.5
LNS	GNC	2	20	52	55
Docking Camera	GNC	2	25	60	50
TDRSS Transponder	CCCT	3	9.3	29.4	135
EVA Transponder	CCCT	2	?	0	20

TABLE B.1 *Component Average & Peak Power*

Component	Subsystem	Quantity	Average Power Each [Watts]	Average Power [Watts]	Peak Power Each [Watts]
Power Amplifier	CCCT	3	?	0	106
Parabolic Antenna	CCCT	2	0	0	0
S-Band Omni Antenna	CCCT	2	0	0	0
EVA Omni Antenna	CCCT	2	0	0	0
Internal Communication Network	CCCT	2	?	0	10
Harness	CCCT	1	0	0	0
Processors (Harris)	CCCT	3	?	0	36
Data Recording Unit	CCCT	1	?	0	2
LOX Propellant Tank	Propulsion	1	0	0	0
LH2 Propellant Tank	Propulsion	1	0	0	0
Plumbing - Valves & Pipes	Propulsion	1	?	0	?
Propellant Management Device	Propulsion	2	0	0	0
Interfaces	Propulsion	2	?	0	?
RL10A-4 Engine	Propulsion	4	?	0	?
Thermal Venting System	Propulsion	2	?	0	40
RCS Tanks, Valves, Pipes	Propulsion	1	?	0	?
RCS Fuel	Propulsion	1	0	0	0
Orbital Maneuvering Thrusters	Propulsion	2	?	0	?
Primary RCS Thrusters	Propulsion	26	0	0	58
Fuel Cell Powerplant	Power	2	0	0	0
Power Conditioner	Power	2	?	0	?
EPS Wire Harness	Power	2	0	0	0
Fuel Cell Tank (H)	Power	1	?	0	?
Hydrogen for Fuel Cells	Power	1	0	0	0
Fuel Cell Tank (O)	Power	1	?	0	1000
Oxygen for Fuel Cells	Power	1	0	0	0
EVA Tools	ECLSS	1	0	0	0
Misc. Other Cargo	ECLSS	1	0	0	0

TABLE B.1 *Component Average & Peak Power*

Component	Subsystem	Quantity	Average Power Each [Watts]	Average Power [Watts]	Peak Power Each [Watts]
Extravehicular Mobility Unit	ECLSS	3	0	0	0
EMU Batteries	ECLSS	10	0	0	20
Atmosphere Control System	ECLSS	1	--	0	0
-Cabin Air Storage System	ECLSS	1	?	0	?
- Pressurized Oxygen Tank	ECLSS	2	?	0	?
- Oxygen In Tanks	ECLSS	2	0	0	0
- CO2 Removal System	ECLSS	1	?	0	30
- LiOH Cannister	ECLSS	5	0	0	0
- Sensors	ECLSS	11	0.5	6	1
- Flow Valves	ECLSS	6	0	0	0
-Cabin Fans	ECLSS	2	30	72	60
Pressure Suit	ECLSS	1	0	0	?
Waste Disposal System	ECLSS	1	0	0	0
Food, Water, Containment System	ECLSS	1	0	0	0
Portable O2 System	ECLSS	8	0	0	0
Fire Detection Sensors	ECLSS	6	0.3	1.8	0.3
Fire Extinguishers	ECLSS	4	0	0	0
Medical Kit	ECLSS	1	0	0	0
Human Interface Controls	ECLSS	1	100	120	200
Internal Lights	ECLSS	4	7.5	36	50
Cargo Bay Lighting	ECLSS	2	0	0	250
Restraints, Seats	ECLSS	8	0	0	0
Astronaut	ECLSS	3	0	0	0
Total				**2552.7**	

Appendix C

Component Maturity and Criticality

TABLE C.1 *Component Maturity and Criticality.*

Component	Subsystem	Quantity	Maturity Level	Level of Criticality
Crew: Pressure Vessel	Structures	1	4	1
Crew: Internal Structure	Structures	1	4	3
Crew: Rad/Debris Shield	Structures	1	3	1
Crew: Common Berthing Adaptor	Structures	1	4	1
SB: Intermodular Interface	Structures	2	4	1
SB: Remote Manipulator System	Structures	2	4	3
SB: Primary/Truss	Structures	1	4	2
SB: Internal	Structures	1	4	2
SB: Rad/Debris Shield	Structures	1	3	2
PP: Primary/Truss	Structures	1	4	1
PP: Rad/Debris Shield	Structures	1	3	1
Base: Heat Pipes	Thermal	1	8	1
Base: Radiators	Thermal	2	8	

TABLE C.1 *Component Maturity and Criticality.*

Component	Subsystem	Quantity	Maturity Level	Level of Criticality
Base: Insulation	Thermal	1	7	
Base: Cold Plates	Thermal	11	8	
Base: Phase Change Devices	Thermal	1	8	3
Base: Radiator Heaters	Thermal	2	8	3
Base: RCS Propellant Heaters	Thermal	64	8	1
Base: Louvres	Thermal	20	4	2
Base: Louvre Mechanical System	Thermal	4	7	1
Crew: Insulation	Thermal	1	8	
Crew: Cold Plates	Thermal	7	8	
Crew: Coolant Pump	Thermal	2	7	1
Crew: Coolant Piping	Thermal	1	7	2
Crew: Radiator Heaters	Thermal	4	8	1
Crew: Louvres	Thermal	20	4	2
Crew: Louver Mechanical System	Thermal	4	7	1
Crew: Air Recycling Unit	Thermal	1	7	1
Crew: Radiator	Thermal	2	8	1
Thermometer	Thermal	30	8	2
DCS with Sun Fans	GNC	4	7	2
Rendezvous Radar Transponder	GNC	2	4	2
MIMU	GNC	2	8	2
MANS	GNC	3	7	2
MAPS	GNC	3	2	2
Star Tracker	GNC	2	8	2
LNS	GNC	2	2	2
Docking Camera	GNC	2	8	3
TDRSS Transponder	CCCT	3	8	2
EVA Transponder	CCCT	2	8	2
Power Amplifier	CCCT	3	7	1
Parabolic Antenna	CCCT	2	7	2
S-Band Omni Antenna	CCCT	2	7	1
EVA Omni Antenna	CCCT	2	7	2
Internal Communication Network	CCCT	2	7	1

TABLE C.1 *Component Maturity and Criticality.*

Component	Subsystem	Quantity	Maturity Level	Level of Criticality
Harness	CCCT	1	7	1
Processors (Harris)	CCCT	3	8	1
Data Recording Unit	CCCT	1	8	4
LOX Propellant Tank	Propulsion	1	4	1
LH2 Propellant Tank	Propulsion	1	4	1
Plumbing - Valves & Pipes	Propulsion	1	7	2
Propellant Management Device	Propulsion	2	7	2
Refueling Interfaces	Propulsion	2	6	2
RL10A-4 Engine	Propulsion	4	8	1
Thermal Venting System	Propulsion	2	6	1
RCS Tanks, Valves, Pipes	Propulsion	1	8	1
RCS Fuel	Propulsion	1	8	
Orbital Maneuvering Thrusters	Propulsion	2	8	1
Primary RCS Thrusters	Propulsion	26	8	1
Fuel Cell Powerplant	Power	2	4	1
Power Conditioner	Power	2	7	2
EPS Wire Harness	Power	2	7	1
Fuel Cell Tank (H)	Power	1	4	1
Hydrogen for Fuel Cells	Power	1	8	1
Fuel Cell Tank (O)	Power	1	4	1
Oxygen for Fuel Cells	Power	1	8	1
EVA Tools	ECLSS	1	8	2
Extravehicular Mobility Unit	ECLSS	3	8	1
EMU Batteries	ECLSS	10	8	3
Atmosphere Control System	ECLSS	1	7	1
Cabin Air Storage System	ECLSS	1	7	2
Pressurized Oxygen Tank	ECLSS	2	7	1
Oxygen In Tanks	ECLSS	2	8	1
CO2 Removal System	ECLSS	1	7	1
LiOH Cannister	ECLSS	5	7	1
Sensors	ECLSS	11	8	2

TABLE C.1 *Component Maturity and Criticality.*

Component	Subsystem	Quantity	Maturity Level	Level of Criticality
Flow Valves	ECLSS	6	8	1
Cabin Fans	ECLSS	2	8	1
Pressure Suit	ECLSS	1	7	1
Waste Disposal System	ECLSS	1	8	4
Food, Water, Containment System	ECLSS	1	8	2
Portable O2 System	ECLSS	8	8	1
Fire Detection Sensors	ECLSS	6	8	3
Fire Extinguishers	ECLSS	4	8	1
Medical Kit	ECLSS	1	8	3
Human Interface Controls	ECLSS	1	7	2
Internal Lights	ECLSS	4	7	3
Cargo Bay Lighting	ECLSS	2	7	3
Restraints, Seats	ECLSS	8	8	1

Appendix D

Detailed Costs and Cost Modelling

Top down and bottom up cost modelling were used to estimate costs for Project Freebird. Below is an example of both models and the detailed cost approximations which were produced for Project Freebird.

D.1 Cost Models

Both cost models are based on a library of previous space missions both manned and unmanned. As with any model, the results are subject to interpretation within certain degrees of accuracy. By utilizing both models, a comparison was established to compensate for model inaccuracy.

D.1.1 Top Down Modelling Approach

The top down model is based heavily on the USAF Unmanned Space Craft Model, Fifth Edition [Fong, 1981]. This model was modified for inclusion in Space Mission Analysis and Design [Wertz & Larson, 1992] by Robert Fong. It was further modified through consultation with Professor Stan Weiss at MIT to account for increased costs of manned spaceflight and alterations in design methodology.

Initially a series of Cost Estimation Relationships were applied to the preliminary design to produce hardware budgets for the subsystems. It was found Project level costs (RDT&E, Project Management, Ground Operations and Support, etc..) could be described with a relatively high degree of accuracy as a function of total vehicle costs. As the vehicle design progressed the top down model was discarded with relation to the flight vehicle. However, the primary relationships between vehicle cost and program level costs were maintained.

Table D.1 presents the top down relationships which were developed to cost model Project Freebird.

The top down estimate has been modified as the design has progressed to reflect the changing vehicle cost estimate produced by the Lockheed model.

TABLE D.1 *Cost Relationshiops Between Vehicle Cost and Program Level Costs*

Program Level Group	Cost Relationship
RDT&E (inc. V &V)	equal to hardware cost
V & V	~ .50*(RDT&E)
Program Management	~ .25*(Hardware + RDT&E + Integration)
Ground Support	~ .33*(Hardware + RDT&E + Integration)

D.1.2 Bottom Up Modelling Approach

The bottom up modelling approach used for Project Freebird is based on the modelling approach used by Lockheed to price the Orbital Transfer Vehicle (OTV) which is under design. It differs from the top down approach in that it attempts to price individual sub-systems utilizing relationships between the hardware, prototype construction and design and development. It is similar to the CER approach found in Wertz & Larson but is designed specifically for manned missions. Most importantly it differs in that it allows for component breakdown within the sub-systems. Each component can be listed seperately and priced out accordingly. When a quoted price is available it is used instead of the cost/weight relationship for the subsystem. Table D.2 on page 314 presents the Lockheed model and the Cost Relationships which use relationships between component factors, such as weight, and cost.

TABLE D.2 *Lockheed OTV Cost Model provided by Steve Moran of Lockheed*

PROGRAM ELEMENT	ENG. HRS. DEV. UNIT, QUAL. UNIT & PROGRAM MANAGEMENT	PROTO-TYPE MANU-FACTURE	PRODUCTION FIRST UNIT (T1)
Total Program			
Program Management	.04*Sum(Hardware + Software Development)		.04*Sum(Hardware Production)
System Engineering	.08*Sum(Hardware + Software Development)		.08*Sum(Hardware Production)
Flight Vehicle (FV)			
Structure	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$4K/lb
Thermal Control	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$8K/lb
Avionics Hardware	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$52K/lb
Flight Software	$260K/1000 Source Lines of Code	none	none
Telemetry, Tracking & Command	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$33K/lb
Power Gen. & Dist. Propulsion	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$4K/lb
Axial	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$9K/lb

TABLE D.2 *Lockheed OTV Cost Model provided by Steve Moran of Lockheed*

PROGRAM ELEMENT	ENG. HRS. DEV. UNIT, QUAL. UNIT & PROGRAM MANAGEMENT	PROTO-TYPE MANU-FACTURE	PRODUCTION FIRST UNIT (T1)
Attitude Control	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$7K/lb
Environ. Control & Life Support Systems	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$32K/lb
Crew Accomodations	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$6K/lb
Orbital Support Equip-ment (on FV)	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$13K/lb
Integration & Assembly	.25*Sum(Hardware + Software Development)		.25*Sum(Hardware Production)
Orbital Support Equipment (on FV)	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$13K/lb
Flight Support Equipment	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$5K/lb
Ground Support Equipment	.1*Total FV T1	included	Included in Develop-ment Cost
Verification Software	$35K/1000 Source Lines of Code	none	none
Ground Operations			
Hardware	3.3*T1*(%New Design + 0.2)/1.2	1.5*T1	$5M/facility
Structures	3.3*T1*(%New Design + 0.2)/1.2	none	$100/sq.ft.
Software	$35K/1000 Source Lines of Code	none	none
System Test	.13*Sum(Hardware + Software Development)		.13*Sum(Hardware Production)

To further modify the cost model to represent price-control management strategies the possibility of component protoflight was added where feasible. Protoflight components are prototype components which need not be tested to failure. There is an additional cost equal to 30% of the hardware cost to facilitate the additional non-intrusive testing and analysis as well as the additional integration costs which must be accomplished for all protoflight materials. However, a new component need not be produced saving the cost of a new unit. The net result is a savings equal to 70% of component hardware cost. This savings is represented in the protoflight savings column.

D.2 Detailed Cost Estimate

Following is a detailed listing of costs for Project Freebird. To produce these estimates a combination of the top down and the bottom up approaches was employed. The top down estimate influenced the treatment of Program Level costs. The bottom up approach was relied upon to price the space vehicle hardware, development, testing and integration.The estimate produces a final vehicle-only cost (including project management) of approximately $1 billion.

Note: Total Component Cost includes all cost of component involved in producing a single subsystem. Production, RDT&E, and V&V are also included.

TABLE D.3 *Program and Management Costs*

Subsystem Component	Actual/ Quoted 1st Unit Cost [$K]	System Factor Qty. or Units [kg]	First Unit Cost [T1] [$K]	Prototype Cost [$K]	Proto-flight Savings [$K	Develop-ment Cost [$K]	% New Design	Total Component Cost $K
Total Program			9,268			19,582		
Program Management			9,268			39,164		
System Engineering						191,150	10%	
Management Cost			18,536			249,896		
Total Management Cost								**268,432**

TABLE D.4 *Structures and Mechanisms Subsystem Components*

Subsystem Component	Actual/ Quoted 1st Unit Cost [$K]	System Factor Qty. or Units [kg]	First Unit Cost [T1] [$K]	Prototype Cost [$K]	Proto-flight Savings [$K]	Develop-ment Cost [$K]	% New Design	Total Component Cost [$K]
Crew: Pressure Vessel		576	7,619	11,429	5,333	6,286	10%	20,000
Crew: Internal Structure		125	1,653	2,480	1,157	1,364	10%	4,340
Crew: Rad/debris Shield		853.8	11,294	16,941	7,906	15,529	30%	35,858
Crew : Common Berthing Adaptor		156	3,439	5,159	2,407	2,837	10%	9,028
P&P Primary/Truss		259.2	2,286	3,429	1,600	1,886	10%	6,000
P&P: Rad/Debris Shield		266.4	3,524	5,286		4,845	30%	13,655
SB: Primary/Truss		112.2	989	1,484		1,088	20%	3,562
SB: Intermodular Interface		240	6,349	9,524		17,460	80%	33,334
SB: Rad/Debris Shield		262	3,466	5,198		4,765	30%	13,429
SB: Internal		12	106	159		87	10%	352
SB: Rmt. Manip. System	75,000	660	75,000	112,500	52,500	61,875	10%	196,875
Structures Totals		**3522.6**	**115,725**	**173,588**	**70,904**	**114,820**		**336,432**

TABLE D.5 *Thermal Control Subsystem Component Costs*

Subsystem Component	Actual/ Quoted 1st Unit Cost [$K]	System Factor Qty. or Units [kg]	First Unit Cost T1] [$K]	Prototype Cost [$K]	Proto-flight Savings [$K]	Develop-ment Cost [$K]	% New Design	Total Component Cost [$K]
Base: Heat Pipes		16.5	218	327		120	0%	666
Base: Insulation		3.6	63	95		35	0%	194
Base: Cold Plates		6.3	83	125		46	0%	254
Base: Phase Change Devices		2.2	29	44		16	0%	89
Base: Radiators Heaters		0.2	3	4		1	0%	8
Base: Louvers		2.2	39	58		96	70%	193
Base: Louver Mech. System		8.8	155	233		128	10%	516
Base: RCS Heaters		6.4	85	127		47	0%	258
Thermometer	2	0	2	2	1	1	0%	4
Crew: Insulation		2.4	42	63		23	0%	129
Crew: Cold Plates		4.1	54	81		43	0%	237
Crew: Coolant Pump		4.4	78	53		19	0%	108
Crew: Coolant Piping		11	194	291		107	0%	592
Crew: Heat Pipes		5	88	132		49	0%	269
Crew: Heaters		0.4	5	8		3	0%	16
Crew: Louvers		2.2	39	58		96	70%	193
Crew: Louver Mech. System		8.8	155	233		128	10%	516
Crew: Air Recycling Unit		38.5	679	1,019		373	0%	2,071
Thermometer	1	0	1	2	1	1	0%	3
Thermal Totals		**118**	**1.925**	**2,887**	**2**	**1,293**		**6,103**

TABLE D.6 *ECLSS Subsystem Component Costs.*

Subsystem Component	Actual/ Quoted 1st Unit Cost [$K]	System Factor Qty. or Units [kg]	First Unit Cost T1] [$K]	Prototype Cost [$K]	Proto-flight Savings [$K]	Develop-ment Cost [$K]	% New Design	Total Component Cost [$K]
EVA Tools	100	110	100	150	70		0%	180
Atm. Control Unit	10,000	0	10,000	15,000	0	13,750	30%	38,750
-Pressurized O2 Tank		38.5	included	included	included	included	0%	0
-Oxygen in Tanks		41.8	included	included	included	included	0%	0
-CO2 Removal System		12	included	included	included	included	0%	0
-LiOH Canister		27.5	included	included	included	included		
-Sensors		6.6	included	included	included	included		
-Flow Valves		6.6	included	included	included	included		
-Cabin Air Storage System		74.8	included	included	included	included	0%	0
-Cabin Fans		7.2	included	included	included	included	0%	0
Pressure Suit		12	847	1,270	593	698	10%	2,222
Waste Disposal System		6	423	635	296	233	0%	995
Extravehicular Mobil Unit	10,000	3 units	30,000	0	0		0%	30,000
-EMU Batteries		10 units	100	included	included	included.	0%	100
Food/Water Containment System		88	776	1,164	543	427	20%	1,824
Portable O2 System		19.2	1,355	2,032	948	745	0%	3,183
Fire Detection System	200	0	200	300	140	110	0%	470
Fire Extinguishers		12	212	317	148	116	0%	497
Medical Kit		1.2	21	32	15	12	0%	50
Emergency System		24	1,693	2,540	1,185	931	0%	3,979
Human Interface Controls		24	1,693	2,540	1,185	1,397	10%	4,444
Lighting	200	0	200	300	140	110	0%	470
Restraints/Seats		180	1,587	2,381	1,111	873	0%	3,730
InternalLighting		6	423	635	296	233	0%	995
Cargo bay Lighting		48	1,058	1,587	741	582	0%	2,487
ECLSS Totals		**632.4**	**50,688**	**30,882**	**7,412**	**20,217**		**94,376**

TABLE D.7 *Guidance, Navigation & Control Subsystem Component Cost*

Subsystem Component	Actual/ Quoted 1st Unit Cost [$K]	System Factor Qty. or Units [kg]	First Unit Cost T1] [$K]	Prototype Cost [$K]	Proto-flight Savings [$K]	Develop-ment Cost [$K]	% New Design	Total Component Cost [$K]
DCS with Sun Fans		20.9	2,396	3,594	1,677	1,977	10%	6,289
Rendezvouz Radar Transponder		25	2,866	4,299	2,006	6,305	0%	11,464
MIMU		10.8	1,238	1,857	867	681	0%	2,910
Star Tracker		52.5	6,019	9,028	4,213	3,310	0%	14,144
LNS		52	5,961	8,942	4,173	16,393	80%	27,124
Docking Camera		12	423	635	296	233	0%	995
Flight Software: MAPS		40 Klines	none	none		10,400	40%	10,400
Flight Software: MANS		30 Klines	none	none		7,800	10%	7,800
GNC Totals		**173.2**	**18,903**	**28,355**	**13,232**	**47,099**		**81,125**

TABLE D.8 *Communications, Command, Control & Telemetry Subsystem Component Cost*

Subsystem Component	Actual/ Quoted 1st Unit Cost [$K]	System Factor Qty. or Units [kg]	First Unit Cost[T1] [$K]	Prototype Cost [$K]	Proto-flight Savings [$K]	Develop-ment Cost [$K]	% New Design	Total Component Cost [$K]
TDRSS Transponder	6,000	19.9	6,000	9,000	4,200	3,300	0%	14,100
EVA Transponder	1,000	7.2	1,000	1,500	700	550	0%	2,350
Power Amplifier	250	18	250	375	175	137	0%	587
Parabolic Antenna		6	437	655	306	480	20%	1,266
S-Band Omni Antenna		2.4	175	262	122	192	20%	506
EVA Omni Antenna		2.4	175	262	122	192	20%	506
Internal Communication Network		2.4	127	190	89	70	0%	298
Harness	100	12	100	150	70	55	0%	235
Processors (Harris)		4.7	342	513	239	188	0%	804
Data Recording Unit		2.4	175	262	122	96	0%	410
Flight Software: Control		30 Klines	none	none		7,800	10%	7,800
C^3T Totals		**77.4**	**8,781**	**13,169**	**6,145**	**13,061**		**28,865**

TABLE D.9 *Power Subsystem Component Cost*

Subsystem Component	Actual/ Quoted 1st Unit Cost [$K]	System Factor Qty. or Units [kg]	First Unit Cost[T1] [$K]	Prototype Cost [$K]	Proto-flight Savings [$K]	Develop-ment Cost [$K]	% New Design	Total Component Cost [$K]
Fuel Cell Powerplant		185.1	4,897	7,345	3,428	13,466	80%	22,281
Power Conditioner		60	1,587	2,381	1,111	4,365	80%	7,222
EPS Wire Harness		110	970	1,455	679	534	0%	2,280
Fuel Cell Tank (H)		37.4	330	495	231	181	0%	775
Fuel Cell Tank (O)		34.9	308	462	215	169	0%	723
Power Totals		**537.4**	**8,092**	**12,138**	**5,664**	**18,716**		**33,281**

TABLE D.10 *Propulsion Subsystem Component Cost*

Subsystem Component	Actual/ Quoted 1st Unit Cost [$K]	System Factor Qty. or Units [kg]	First Unit Cost[T1] [$K]	Prototype Cost [$K]	Proto-flight Savings [$K]	Develop-ment Cost [$K]	% New Design	Total Component Cost [$K]
RL10A-4 Engines	12,000	705.6	12,000	18,000		13,200	20%	43,200
Secondary RCS Thrusters		21.5	758	1,138		417	0%	2,313
Primary RCS Thrusters		101	3,563	5,344		3,919	20%	12,825
RCS Tanks, Valves, Pipes		165	1,455	2,183		1,200	10%	4,838
Plumbing, Valves, Pipes		253	3,347	5,020		1,841	0%	10,207
Thermal Venting System		72	2,540	3,810		6,286	70%	12,635
LOX Propellant Tank		94.8	1,254	1,881		1,035	10%	4,169
LH2 Propellant Tank		128.4	1,698	2,548		1,401	10%	5,647
Refueling Interfaces		66	2,328	3,492		5,762	70%	11,582
PMD		44	582	873		640	20%	2,095
Engine Requalification						30,000		
Propulsion Totals		**1363.8**	**29,525**	**35,623**	**0**	**60,164**		**125,312**

TABLE D.11 *Total Costs*

Subsystem Component	Actual/ Quoted 1st Unit Cost ($K)	System Factor Qty. or Units (kg)	First Unit Cost [T1] ($K)	Prototype Cost ($K)	Proto-flight Savings ($K)	Develop-ment Cost ($K)	% New Design (%)	Total Component Cost ($K)
Orbital Support Eqt. (on FV)			0	0	0	0		
Total Vehicle Hardware Cost			233,639	293,641	103,359	278,573		
Integration & Assembly			58,410	none		69,643		

Theoretical First Unit [T1] Vehicle Only:	**$292M**
Development and Delivery of 1 Vehicle:	**$936M**
D&D of 1 Vehicle utilizing protoflight:	**$833M**
D&D of 1 Vehicle Including Mgt. Cost:	**$1,105M**

TABLE D.12 *Support Costs*

Subsystem Component	Actual/ Quoted 1st Unit Cost ($K)	System Factor Qty. or Units (kg)	First Unit Cost[T1] ($K)	Prototype Cost ($K)	Proto-flight Savings ($K)	Develop-ment Cost ($K)	% New Design (%)	Total Component Cost ($K)
Support								
Orbital Support Eq. (off FV)			0	0	0			
Launch Costs (1 ArianneV & 1 Proton-C)		135,000						
Ground Support Equipment			included	included		29,664	10%	
Verification Software		200 Klines				7,000	N/A	
Total Support Cost		135,00	0			36,664		
Ground Operations								
Hardware		3 facili-ties	30,000	none		33,000	20%	
Structures		400,000 sq. ft.	40,000	none		22,000	0%	
Software		1000 Klines	none	none		35,000		
Total Ground Cost			70,000	none		90,000		
System Test			30,373			32,297		666,639 cost not vehicle
								259,334 Ground, Operations & Support
Total Development						507,178		222,670 round costs

TABLE D.12 *Support Costs*

Subsystem Component	Actual/ Quoted 1st Unit Cost ($K)	System Factor Qty. or Units (kg)	First Unit Cost[T1] ($K)	Prototype Cost ($K)	Proto-flight Savings ($K)	Develop-ment Cost ($K)	% New Design (%)	Total Component Cost ($K)
Total Protoype Mfr.				296,641				38,900 Ground Ops Mainte-nace/year
Total Theoretical First Unit			392,421					
Total Management Cost	272,304							
PROGRAM COST (Devel-opment, Prototype, First Unit, Support & Manage-ment):								1,500,186

Refueling Cost/Tank	**52,273**
Booster Pack	**138,909**
Cost/Mission LEO	**57,829**
Cost/Mission GEO	**162,375**

Total Cost up to and including launch	**$1,500M**
First Year Budget + vehicle cost	**$2,298M**
10 Year Budget + vehicle cost	**$9,311M**

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Appendix E

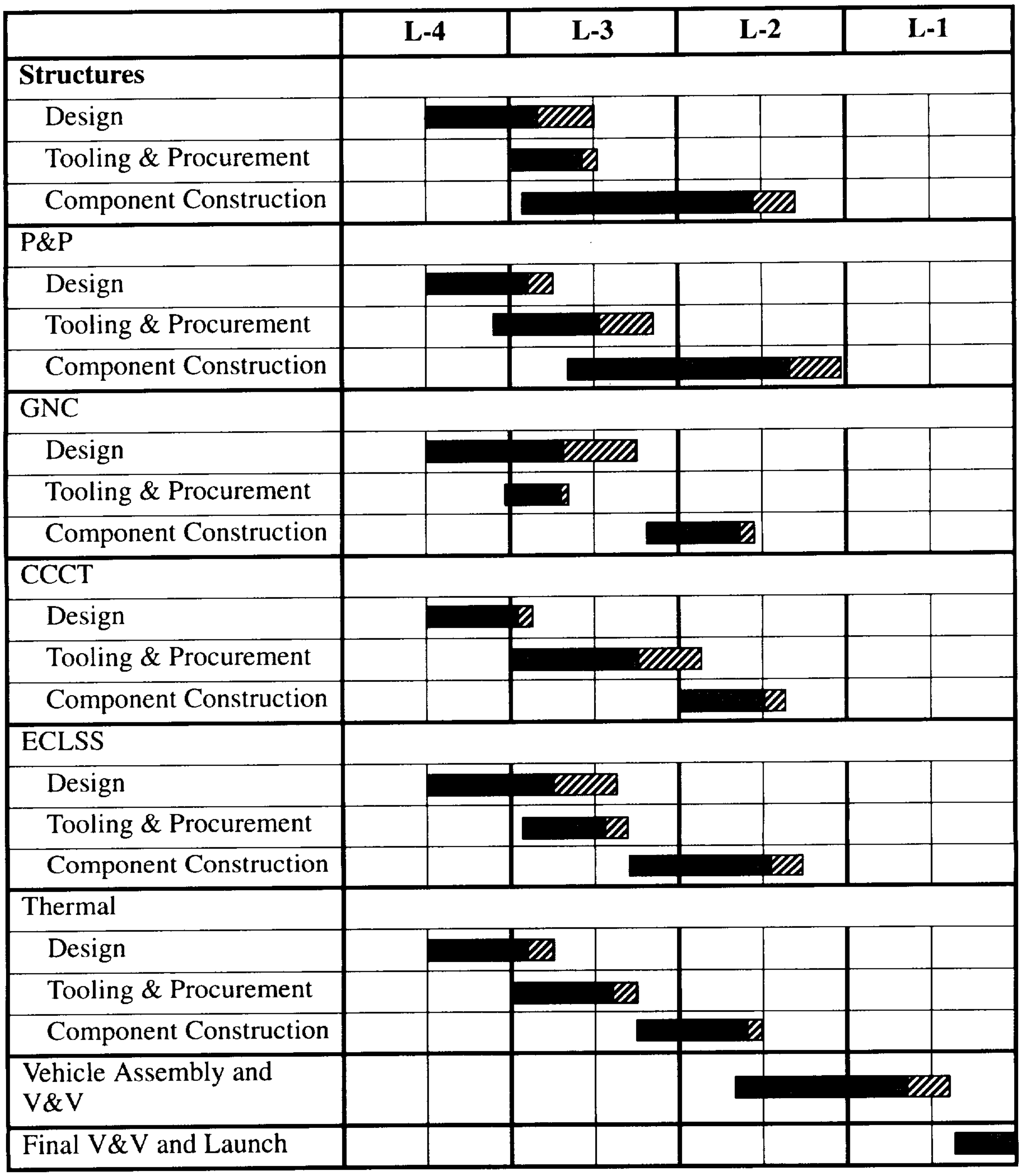
Master Schedule

The framework for the master schedule was assembled by analyzing the design, procurement and construction time for similar systems. These analogous times were then modified by a time contingency factor as outlined in the chart below. Special consideration was given to new industry trends such as the coordination of design, tooling and manufacturing. For example, the propulsion systems is designed to use the currently available RL-10A engines but re-qualified for 100 starts. Consequently, the procurement category was rated with a high (30%) time contingency to allow for the requalification process.

The master schedule shows the indicated subsystems times and includes the appropriate contingency time. The scale across the top is in years till launch.

Low - 10% Med - 20% High - 30%

	Design	Procurement/Tooling	Construct
Structures	High	Low	Med
P & P	Med	High	Med
Thermal	Med	Med	Low
GNC	High	Low	Low
C^3T	Low	High	Low
ECLSS	High	Med	Med



Contingency time

Appendix F

Acronym List

TABLE F.1 *Project Freebird Acronyms and Explanations.*

Acronym	
A	Acceleration Test
AC	Acoustic Test
ACRV	Assured Crew Return Vehicle
ADCS	Attitude Determination and Control System
BFO	Blood Forming Organs
BI	Burn-in Test
C^3T	Command, Communication, Control, and Telemetry
CBA	Common Berthing Adapter
CCTV	Closed Circuit Television
CES	Conical Earth Sensor
CMG	Control Moment Gyro
CO_2	Carbon Dioxide
C&P	Cost and Producibility
DCS	Dual-Cone Scanner
DOF	Degrees of Freedom
DSN	Deep Space Network
ECLSS	Environmental Control and Life Support System

TABLE F.1 *Project Freebird Acronyms and Explanations.*

Acronym	
EER	End Effector Retainer
EMC	Electromagnetic Compatibility Test
EMPT	Extended Mission Propellant Tank
EMU	Extravehicular Mobility Unit
EPS	Electrical Power System
EVA	Extravehicular Activity
F	Functional Test
FCP	Fuel Cell Powerplant
FDIE	Failure Detection Isolation and Estimation
FSS	Flight Support Station
g	Acceleration due to gravity
GCR	Galactic Cosmic Ray
GEO	Geosynchronous Orbit
GNC	Guidance, Navigation, and Control
GPS	Global Positioning System
IMU	Inertial Measurement Unit
IPS	Image Processing System
IR	Infrared
ISSA	International Space Station Alpha
JPL	Jet Propulsion Laboratory
L	Leak Test
LC	Life Cycle Test
LDS	Laser Docking System
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LNS	Laser Navigation System
LOX	Liquid Oxygen
LSS	Life Support System
LiOH	Lithium Hydroxide
MANS	Microcosm Autonomous Navigation System
MAPS	Microcosm Autonomous Planning System
MDB	Mesh Double Bumper
MF	Mass Fraction
MIMU	Miniature Inertial Measurement Unit
MIPS	Millions of Instructions per Second

TABLE F.1 *Project Freebird Acronyms and Explanations.*

Acronym	
MIT	Massachusetts Institute of Technology
MLI	Multi-Layer Insulation
MMH	Monomethyl Hydrazine
MMU	Manned Maneuvering Unit
MSS	Multi-Shock Shield
NASA	National Aeronautics and Space Administration
N_2H_4	Nitrogen Tetroxide
O_2	Oxygen
OST	Outer Space Treaty
OTS	Off the shelf
OTV	Orbital Transfer Vehicle
P	Burst Pressure Test
P&P	Power and Propulsion
PCU	Power Control Unit
PF	Protoflight Test
PIT	Process Improvement Team
PLSS	Portable Life Support System
PRT	Performance Refinement Team
PS	Pyro Shock Test
PTU	Pan and Tilt Unit
Prox Ops	Proximity Area Operations
RCS	Reaction Control System
RDT&E	Research, Development, Test and Evaluation
RFC	Regenerable Fuel Cell
RLG	Ring Laser Gyroscope
RMS	Remote Manipulation System
RRS	Rendezvous Radar Set
RV	Random Vibration Test
SSA	Space Suit Assembly
STS	Space Transportation System (Space Shuttle)
SV	Sine Vibration Test
TB	Thermal Balance Test
TC	Thermal Cycling Test
TDRSS	Tracking and Data Relay Satellite System
TQM	Total Quality Management

TABLE F.1 *Project Freebird Acronyms and Explanations.*

Acronym	
TRM	Technology Risk Management
TV	Thermal Vacuum Test
TVS	Thermal Venting System
VSF	Vehicle Storage Facility
WBS	Work Breakdown Structure

Appendix G

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