University of Illinois
at
Urbana-Champaign
Aeronautical and Astronautical Engineering

AAE 241
Aerospace Vehicle Design, Spacecraft Section
Final Project Reports

Volume II
Project Groups 6 through 8

May 1989

LiBERTy

Servicing FREEDOM with LiBERTy

FINAL DESIGN REPORT

FOR

LiBERTy - LOGISTICS BUS AND EMERGENCY RESCUE TRANSPORT

UNIVERSITY OF ILLINOIS

AEROSPACE VEHICLE DESIGN COURSE -- AAE241 GROUP #6

2 MAY 1989

Contributing Authors:

Mission Management, Planning, and Costing Subsystem	John P. Hedrick(*) [redacted]
Structures Subsystem	Rommel C. Villalobos [redacted] Rommel C. Villalobos
Power and Propulsion Subsystem	Richard C. Gianvecchio [redacted]
Attitude and Articulation Control Subsystem	Dan J. Quitno [redacted] Dan J. Quitno
Command and Data Control Subsystem	Dave J. Quinn [redacted]
Environmental Control and Life Support Subsystem	Ernst P. Jahensch [redacted]
Reentry and Recovery Subsystem	Jeffrey C. Berg [redacted] Jeffrey C Berg

(*) denotes project leader

TABLE OF CONTENTS

Abstract....................iii

Introduction....................v

Mission Management, Planning and Costing Subsystem....................1-1

Structures Subsystem....................2-1

Power and Propulsion Subsystem....................3-1

Attitude and Articulation Control Subsystem....................4-1

Command and Data Control Subsystem....................5-1

Environmental Control and Life Support Subsystem....................6-1

Reentry and Recovery Subsystem....................7-1

ABSTRACT

LiBERTy was designed to meet the requirements listed in the request for proposal for a logistics resupply and crew emergency return system for Space Station Freedom. Logistics resupply of the Space Station will take place every 90 days as outlined in figure 1-1. Two vehicles will be required to carry up the necessary payload. A Titan IV expendable launch vehicle (ELV) utilizing the upgraded solid rocket motors (SRMU) will be used to launch each vehicle up to the Station.

The structure for LiBERTy will utilize materials and technology similar to those used in the production of the Space Shuttle. The overall mass for LiBERTy's structure is approximately 2560 kg, and the total mass for the structure and other subsystems is 6085 kg. LiBERTy is 12.2 meters tall, and is 4.3 meters wide at the aft section. All the materials used have been proven and should not pose any problems.

A bi-propellant system using monomethylhydrazine and nitrogen tetroxide was used for propulsion. 4 smaller scaled OMS engines will be used to provide the necessary Δv for reentry only.

The power system uses state-of-the-art Lithium Bromine Complex cells to provide necessary power for all other subsystems. PCU's control and distribute this power.

The actuators are 16 gas thrusters with 90 Newtons of force each. Nitrogen gas is the fuel used in the thruster. Two sun sensors and one star scanner provide feedback on the orientation of LiBERTy. A range finder and optoelectronic sensors are used for docking with the Space Station. Manual labor will be employed for payload loading and unloading.

S-band command link through TDRSS with the ground Station and a direct S-band link to the Space Station will be provided. Remote control is maintained through these links except for an automated rendezvous and docking procedure, and the emergency evacuation commands from the crew. Cooperation with the SAR team begins upon return and supplies a low-power UHF system for crew voice communication.

LiBERTy can support 8 crew members for up to 10 hours. An active atmosphere control system supplies O_2 and eliminates CO_2. A passive temperature control system, using wax panels and blankets, ensures a comfortable environment.

The module separates from the Space Station and maneuvers to acquire the proper attitude for a deorbit burn. The Δv is chosen after a landing sit is determined. Maintaining an entry flight path angle of -1.5°, LiBERTy will experience a maximum of 4 g's and thermal temperature of 1648°C. Parachutes are deployed to decelerate the module prior to splashdown. Impact g-forces will be less than 10 g's experienced briefly (0.2 sec). Naval Search and Recovery will move in to help the crew members and recover LiBERTy to begin processing for another flight.

The total cost for 4 LiBERTy vehicles will be about $1.3 billion.

INTRODUCTION

The 1990s will see many adventurous endeavors in space, and none will prove to be more exciting than the construction of Space Station *Freedom*. *Freedom* is an international effort between the United States and its cooperating partners designed to have a longer lifetime, higher reliability, and lower maintenance requirements than previous manned space flight mission. *Freedom* will also provide a base of operations for scientific experimentation, research and development, and manufacturing. It will be a way point for exploratory missions to the Moon and the planets.

The Station will need to be periodically resupplied with experiments, hardware, and consumables. Similarly, contingencies must be made for the removal of waste material and other equipment for return to earth. The use of a space vehicle designed specifically for this mission -- a Logistics Resupply Module (LRM) -- would greatly facilitate these tasks as well as minimize the use of the Space Shuttle for this role.

In addition to the logistic resupply of *Freedom*, there must be a plan for the emergency return of crew members to earth from the Space Station. The crew emergency return vehicle (CERV) must provide life support and thermal protection for crew members during reentry. The CERV must also return safely to earth as close to immediate medical facilities as possible.

The goal of this report is to conceptually design a vehicle capable of fulfilling both the roles of a logistics module and crew emergency return vehicle in response to the Request for Proposal for a Logistics Resupply and Emergency Crew Return System for Space Station *Freedom*. The Logistics Bus and Emergency Rescue Transport vehicle (LiBERTy) will use proven and reasonable technologies to offer the most cost-effective and reliable support of the Space Station for years to come. The request for proposal (RFP) was distilled to obtain and understand all of the requirements that this final design report must fulfill. The major requirements include:

1. The system will consist of three primary components: logistics resupply capsule(s), Space Station docking adaptor, and orbital transfer propulsion system.

2. The following subsystems are identified for all the purposes of system integration. The derived requirements for each subsystem are also listed

a) Mission, Management, Planning, and Costing (MMPC)
b) Structures (STRUC)
c) Power and Propulsion (PPS)
d) Attitude and Articulation Control (AACS)
e) Command and Data Control (CDC)
f) Environmental Control and Life Support (ECLSS)
g) Reentry and Recovery (RRS)

3. The system's components and payload will be delivered to orbit on an expendable launch vehicle. The extent of shuttle support should be identified and minimized. Vehicle components must be able to be returned to earth in the Space Shuttle bay.

4. Nothing in the system's design should preclude it from performing several possible missions, carrying vastly different payloads to the Space Station.

5. The system will have a design lifetime of six years, but nothing in its design should preclude it from exceeding this lifetime.

6. The vehicle will use the latest advances in artificial intelligence where applicable to enhance mission reliability and reduce mission costs.

7. All vehicle components will operate under positive Space Station control at all times.

8. The design will stress simplicity, reliability, and low cost.

9. For cost estimating and overall planning, it will be assumed that four logistics resupply modules will be built. Three will be flight ready while the fourth will be required for use in an integrated ground test system.

MISSION MANAGEMENT, PLANNING, AND COSTING SUBSYSTEM

JOHN P. HEDRICK

INTRODUCTION

The mission planner is responsible for the integration of all other subsystems, the successful completion of the mission, and necessary ground support and logistics. The requirements distilled from the RFP relating to MMPC include:

1. Identification of payloads
2. Integration of payloads into transport module
3. Launch vehicle selection
4. Δv calculation (this will be discussed in section 7, Reentry and Recovery)
4. Mission support
5. Mission timeline
6. Mission costing
7. Integration with other subsystems

The purpose of this section is to detail the design of the LiBERTy vehicle as it pertains to mission management. Figure 1-1 depicts the method of attack used in the design process. The logistics resupply and crew emergency return mission profiles of LiBERTy will discussed, and the major components, options, and trades are addressed. Lastly, problem areas encountered during this conceptual design, as well as recommendations will be discussed.

LOGISTICS RESUPPLY

Space Station concepts being considered by NASA depend heavily on the Space Shuttle acting as the work horse in Space Station resupply. Since the shuttle will be supporting other programs in addition to the Station, however, there are likely to be limitations on its availability to fulfill all of the required logistics resupply for Freedom. LiBERTy, in its role as a reusable unmanned logistic resupply module, will supplement the Space Shuttle, and thereby solve the availability problem. The resupply mission of LiBERTy can then be divided into six key phases: payload identification; expendable launch vehicle (ELV) selection; orbit injection; terminal phase; docking; and unloading and storage.

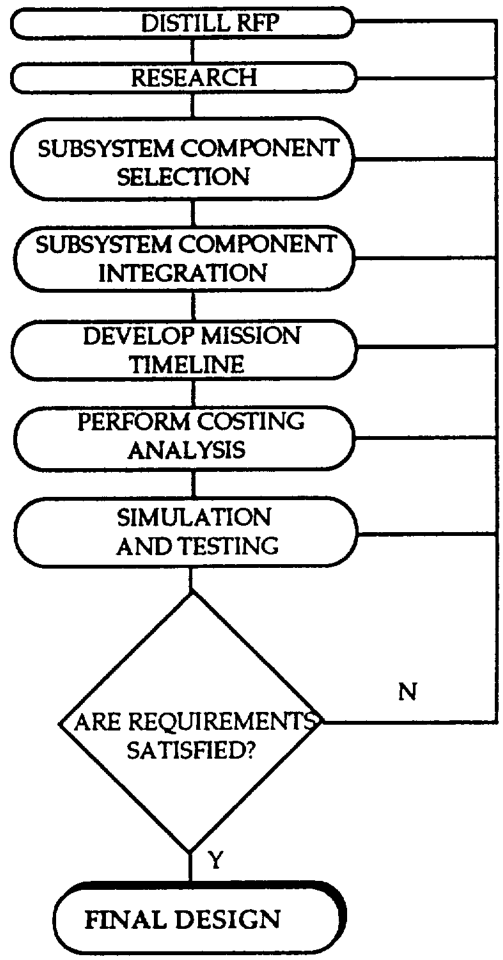


Figure 1-1. MMPC Method of Attack

Table 1-1. 90 Day Total Logistics Requirements

Mass

	Pressurized Crew/Sta	(kg) Customer	Unpress. Crew/Sta	(kg) Customer	Fluids Crew/Sta	(kg) Customer	Propellants Crew/Sta	(kg) Customer	Totals (kg)
Up	4148.56	4954.14	513.01	4152.18	360.61	365.14	45.36	1681.92	16220.92
Down	3497.99	4757.39	513.01	4152.18	0.0000	173.73	0.000	0.00	13094.30

Volume

	Pressurized Crew/Sta	(m^3) Customer	Unpress. Crew/Sta	(m^3) Customer	Fluids Crew/Sta	(m^3) Customer	Propellants Crew/Sta	(m^3) Customer	Totals (m^3)
Up	14.78	13.92	4.53	32.64	0.45	0.50	0.57	1.68	69.06
Down	11.50	13.75	4.53	32.64	0.00	0.17	0.00	0.00	62.59

PAYLOAD IDENTIFICATION

Present Space Station concepts focus on a 90-day logistics resupply period. Table 1-1 shows the requirements for one 90-day resupply period. These mass and volume totals are the main design drivers for LiBERTy, and they were given to the Structures Analyst as guidelines in sizing the vehicle.

The up and down totals for both mass and volume are then divided into the individual needs of the crew/station and customers of the Station for pressurized and unpressurized materials, fluids, and propellants. The largest item cited for resupply, a dewar flask, is guaranteed to fit through the 127 cm x 127 cm hatch. The totals for fluids includes both gases and liquids. Although the internal volume must accommodate the totals, 69.06 m^3 up and 62.59 m^3 down, the sizing of individual gas and liquid bottles are not accounted for. It should be noted that the data in Table 1-1 reflect those weights and volumes for the 90-day logistics resupply payload alone; Environmental Control and Life Support Subsystem requirements for the CERV mission are not considered here.

ELV SELECTION

The mass of the payload is also one of several design drivers for the selection of an expendable launch vehicle. The weight of the fully loaded vehicle must be within the payload capacity of the chosen ELV with some margin to allow for any possible future growth in system weight. Another factor in ELV selection is whether or not LiBERTy will fit within the launch vehicle's fairings. The ELV must also be able to achieve low circular orbits at the same inclination as the Space Station, 28.5°. Launch sites at Cape Canaveral Space Launch Center (CCAFS) are desirable, because the inclination of CCAFS is 28.5°, and this means no plane changes are required to reach the Station. The Space Station orbit varies with the solar cycle from 290 km to 430 km. A standard circular orbit for ELVs is 407 km x 407 km; this would allow the ELV to place LiBERTy as close to the Station as possible.

Several ELVs were researched, and four were selected for consideration; Delta, Atlas Centaur, Titan III, and Titan IV. Figure 1-2 shows the comparisons of the launch vehicles, the necessary design considerations, and the final selection. The major factor in the selection process is payload capacity. The total mass for a 90-day resupply period is 16,220.02 kg (see Table 1-1). The estimated mass of each subsystem was obtained, and from this the mass of

ELV	Orbit (km)	Inclination (Launch Site)	Payload Fairing (m)	Payload Capacity (kg)	Success Rate (%)	Vehicle Selection
Delta	407km x 407km	28.5° (CCAFS)	2-9	2500	98%	
Atlas-Centaur	407km x 407km	28.5° (CCAFS)	12	5900-7200	83%	
Titan III	407km x 407km	28.5° (CCAFS)	6-9	15000	96%	
Titan IV	407km x 407km	28.5° (CCAFS)	7	18000	98%	xxxxxxxxxx

*note: The Titan III and Titan IV success rates are projected numbers, and have yet to be verified by actual flight testing.

Figure 1-2. ELV Comparison

LiBERTy without payload was calculated to be approximately 6,085 kg. The total mass to be put into orbit every 90 days is then about 22,305 kg. None of the ELVs is capable of carrying this size payload, and it therefore becomes necessary to split the payload into two separate vehicles; this will be discussed in further detail later. Assuming two launches per 90 days, the payload for one ELV would be approximately 14,200 kg. Adding a factor of 20% to allow for any future growth in system weight then raises the payload to about 17,000 kg.

The only ELV capable of lifting this payload is the Titan IV equipped with upgraded solid rocket motors (SRMUs), which is expected to be operational by 1994. Even though the Titan III could handle the payload, an increase in the expected mass of just 900 kg would render it unusable; therefore, it would not be reasonable to make the Titan III the main launch vehicle. Also, the price for the Titan III is the same as the Titan IV ($110 million), and the fairing sizes on the Titan III are much smaller. For these reasons, the Titan IV was decided upon as being the main ELV for LiBERTy.

ORBIT INJECTION

Once the ELV was decided upon, the intent was to place LiBERTy as close to the Station as possible. The calculations for Titan IV Δv's and trajectory considerations are too lengthy and complex to be considered here, but it is assumed the 407 km x 407 km parking orbit is sufficient to place LiBERTy close enough to the Space Station so only a small portion of our redundant fuel supply is needed to perform the orbit changes to bring LiBERTy within sensor acquisition range of the Station. Past data and Soviet experiences with their on-orbit stations suggest lock-on from a "friendly" station can be achieved out to nearly 35 km.

TERMINAL PHASE AND DOCKING

Once LiBERTy has sensor lock-on from the Station, control of its operations will pass from ground control to complete positive control of all subsystems by the Space Station. Sensors on board LiBERTy will aid in targeting it into docking range of the Station. The actual docking phase of the flight starts at about 90 meters away from the Station (see Reference 1-2). During this phase, LiBERTy will be slowing down its rate of closure on the Station, and it will begin to orientate and maneuver into position for docking. The entire procedure will be engineered for automatic rendezvous and docking. The latest advances in artificial intelligence will be used, wherever possible, to ensure a safe docking requires no human intervention. The terminal and docking phases also denote the change in direct control of LiBERTy's systems from the mission planner to both the AACS and CDC subsystems. A more detailed analysis of this phase as well as the unloading of the cargo will be discussed in Sections 4 and 5 of this report.

UNLOADING/STORAGE

Once a secure link has been established, unloading of cargo can begin; the exact method in which this takes place will be discussed in later sections. When all the cargo is unloaded, LiBERTy will be prepared for storage until it is loaded up with waste materials and experiments or called upon to perform an emergency crew return.

CREW EMERGENCY RETURN

The key aspects of an effective emergency return mission is its ability to provide maximum protection for the crew during reentry and splashdown, minimize the amount of loiter time needed to assess an appropriate landing site, and to land as close to emergency medical facilities as possible. The emergency crew return mission of LiBERTy falls mainly under the control of the Environmental Control and Life Support Subsystem in conjunction with the Reentry and Recovery Subsystem during the reentry phase. However, mission management does play a role in assessing how many crew members LiBERTy will be able to support and also in determining the mission length and Δv requirements.

CREW ACCOMADATIONS

The ECLSS Analyst performed several trades and comparisons early on in the design process to determine LiBERTy's crew size. Coordination with the ESCLSS Analyst was needed to determine how many vehicles would be required and the number of crew members each would be designed for to insure sufficient protection to all eight Space Station crew members. One requirement of any CERV mission is that if one crew member has to be returned to Earth another must accompany him; there will never be fewer than two people brought back. Also, the Space Station can no longer remain operational if it is less than 50% manned. Therefore, if it is necessary to bring back five people, the remaining three must return as well. A one vehicle system was dismissed early on, because it did not provide any means of emergency return for those crew members that remained on the Station. For reasons outlined in Section 6, the choice was narrowed down to supporting either six or eight persons.

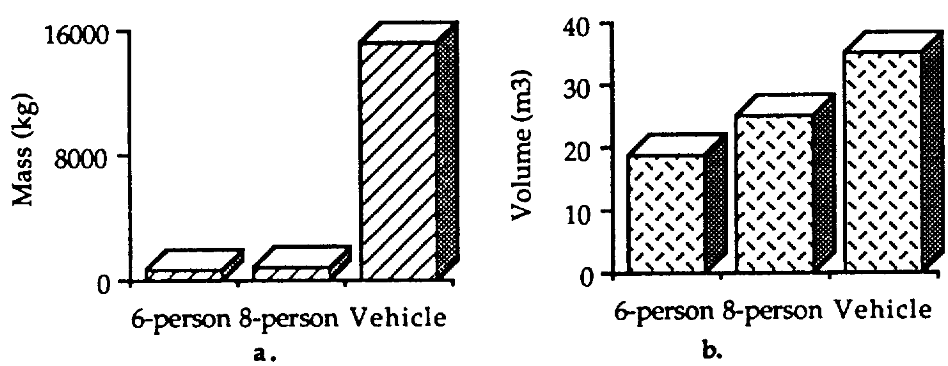


Figure 1- 3 Comparison of possible configurations to total vehicle mass for a 32 hour period (a) mass and (b) volume

Figures 1-3a and 1-3b outline the trades performed to determine crew size. The numbers represent the amount of life support needed for a 32 hour period for both a 6-person and 8-person vehicle. This is the worst-case scenario for life support, and as will be discussed later, LiBERTy was designed for a shorter time period. Since the difference between supporting six persons and eight is negligible, and such a small fraction of the total vehicle mass is devoted to ECLSS, the decision was made to equip LiBERTy with enough life support to to sustain a crew of eight. Section 6, Environmental Control and Life Support Subsystem, explains this more fully.

REENTRY

Operations between the release of LiBERTy from the Space Station and recovery were planned in conjunction with the RCS Analyst. The method of reentry and recovery is the same regardless of whether LiBERTy is returning in the LRM or CERV configuration. The LiBERTy will splashdown in the ocean, and then will be picked up by a naval rescue and recovery team (see Section 7). Water landing was decided on for the initial study, because it was considered the minimum cost solution. Consideration of land recovery, which would be necessary to land LiBERTy closer to medical facilities, will be proposed for follow on studies. Presently, there are only two sites which provide the necessary naval support of a splashdown return; just off of Hawaii and Cape Canaveral, Florida.

Research done on the SCRAM vehicle that is currently being considered by NASA to perform just the CERV mission shows the time from departure at the Space Station to touchdown is slightly less than three hours, and the total Δv's requirement for reentry, as determined in Section 7, is .15 km/s (see Reference 1-3). This includes a maximum 90 minute loiter time to complete one orbit. Prior to departure, LiBERTy's on-board computer systems will check all available data relating weather and atmospheric conditions at possible landing sites and chose the optimal target. Once targeting has been completed, LiBERTy will release from the Space Station and prepare for orbit and descent. The entry flight path angles, and maximum g loading during splashdown were calculated by the RRS Analyst to ensure the safety of both man and machine.

A second possibility for reentry is to have LiBERTy brought back to Earth via the Space Shuttle. This can only be accomplished if the Shuttle is in the vicinity of the Space Station or able to arrive in time. The Structures Analyst has designed LiBERTy so it will fit within the shuttle cargo bay as per the requirement listed in the RFP. The use of the shuttle, however, must be kept to a minimum for cost effectiveness. As mentioned earlier, the availability of the shuttle may not exist in times of emergency. Therefore, LiBERTy should be returned in the shuttle only in cases where an injured crew member or a delicate experiment could not stand the reentry and splashdown.

The ECLSS Analyst was contacted to determine the time length for which the on-board life support systems were to be sized for. Basing the decision on reentry times for the SCRAM, and adding a safety factor of 4 to guard against long loiter times (which, for example, could occur due to poor weather at the targeted landing site), LiBERTy was designed to provide enough life support to maintain an eight person crew for ten hours.

PRE-LAUNCH OPERATIONS

The pre-launch operations of LiBERTy and the processing requirements at the Kennedy Space Center (KSC) will be kept to a minimum. KSC operations will mainly include the readying of the Titan IV ELV for launch, servicing of LiBERTy's subsystems, pumping fuel to the on-board propulsion system and the installation of the flight batteries. Further operations will be done at facilities other than KSC. These "off-line" operations will include:

1. Propulsion subsystem purge, pressurization and leak checks
2. ECLSS check
3. RRS installation
4. Power generation and distribution check
5. Loading of logistics resupply cargo
6. Crew Accommodations and medical equipment stowage

ON-ORBIT OPERATIONS

Once the logistics cargo has been unloaded from LiBERTy, it will be prepared for storage for the remainder of its time docked to the Space Station, and handled like any attached payload. Periodic automated checks of LiBERTy must be made to assure all subsystems are functioning as expected. Maintenance of LiBERTy while attached to the Space Station will be done only when necessary and cost effective, and will consist of the replacement and testing of components in a shirt-sleeve environment. For systems critical to flight capability, sufficient instructions will be provided to the Space Station crew to attempt necessary repairs. Spare parts will be provided on the Space Station.

When LiBERTy is ready to depart from the Space Station, whether in the LRM or CERV configuration, all subsystems will go through a checkout. LiBERTy will wait at the Space Station - if this is possible - to minimize loiter time. In cases of emergency, however, there may not be time to perform the needed activities. The crew should be able to board within a few minutes, and then final separation procedures will occur automatically or manually from LiBERTy. Detailed analysis of reentry and recovery will be given in Section 7.

MISSION TIMELINE

Research and development of the LiBERTy project should begin immediately upon approval of this conceptual design if LiBERTy is to be operational in the same time frame as Space Station Freedom. The use of off-the-shelf hardware and technologies present before 1995 will cut down much of the initial research, and the emphasis can be focused on development. Closed-loop testing and simulating of each the subsystems should be done several months in advance of the first launch. Simultaneously, the vehicle shell and reentry and recovery mechanisms will be subjected to drop tests and recovery procedures to ensure those systems are working optimally.

Figure 1-4 is a sample mission timeline detailing the use of LiBERTy over a one year period. Day zero on the timeline at the top of figure 1-4 denotes the first day of each 90-day resupply period. Two Titan IV launches will be required every 90-days to transport the required payload. To ease the docking and unloading procedures at the Station as well as the launch preparations on the ground, the launches are staggered over a period of several days. The 13 day preparation period for the first Titan IV will begin on day 75 of the 90-day period. Launch of the ELV will occur on day 88, and unloading will begin as soon as LiBERTy is securely docked. At this time, preparations for the return of one of the LiBERTy vehicles already attached to the Station will begin. Departure from the Station will occur in two days on day 90. Meanwhile, preparations for the second Titan IV launch will begin on day 79 with the subsequent launch occurring on day 92; four days after the initial launch. The departure of the other vehicle (if two were docked at the Station) will occur on day 94. The values for the recycling and refurbishment periods were assumed to be long enough periods of time, but no there is no data on similar missions or experience to gage

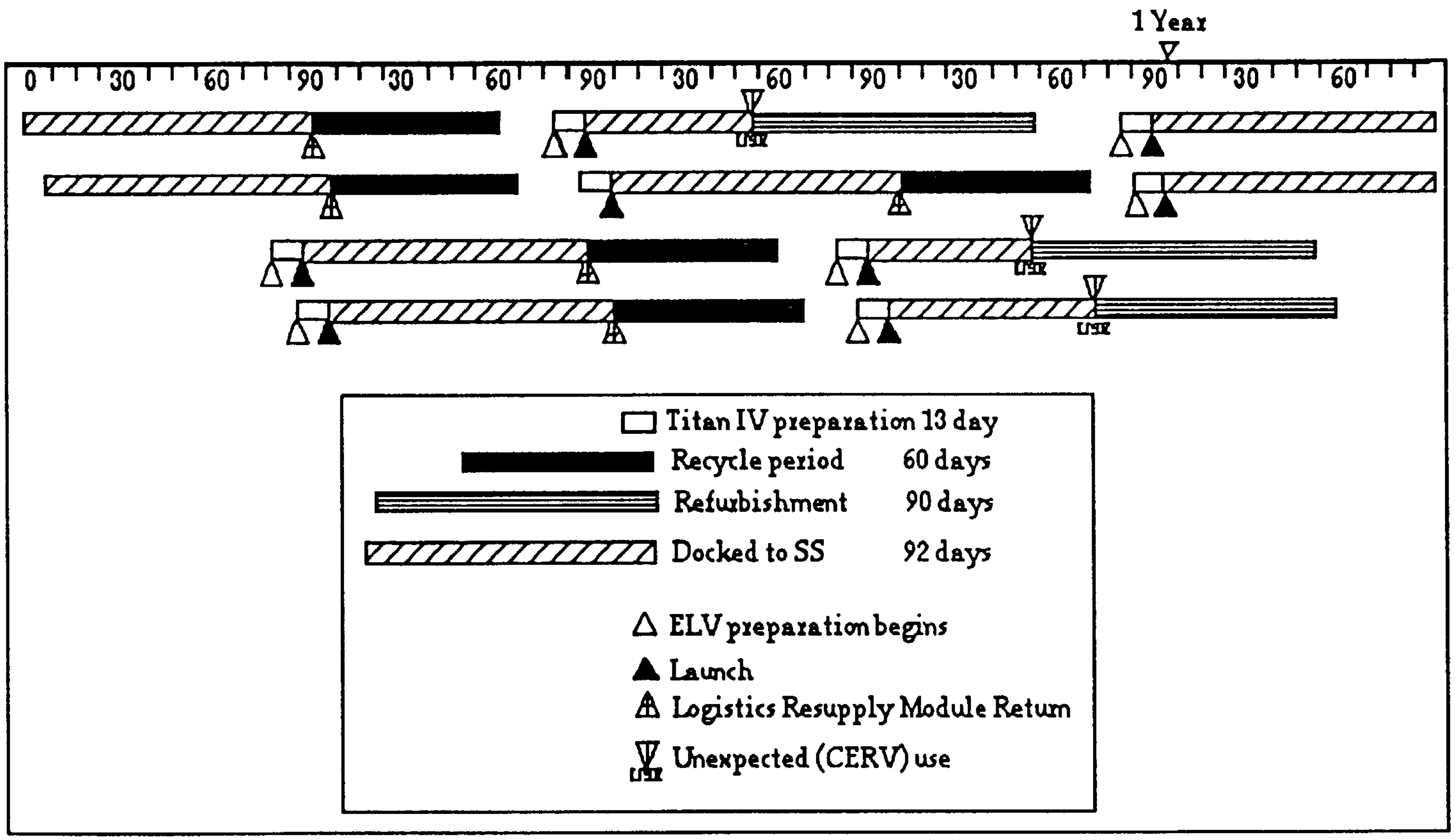


FIGURE 1-4. MISSION TIMELINE

the correctness of these assumptions. Using these periods, however, it can be seen that each vehicle will be able to meet its next launch date; even if it has to be refurbished after an emergency crew return. In case one of the vehicles does have to leave orbit earlier than expected due to an emergency crew return, the other vehicle will be able to transport the remaining station crew in cases of emergency without the need of sending up another LiBERTy before the next scheduled launch date.

The RFP requires "the system will have a design lifetime of six years, but nothing in its design should preclude it from exceeding this lifetime." To accomplish this, a system of Fault and Redundancy Management (FRM) should be incorporated into the design phase. The goal of FRM is to ensure a system's function is available at all times through redundant systems. This is done through hardware redundancy - identical hardware replication - and functional redundancy - similar functional capability by another system. Requirements of FRM to assure full system coverage include the determination of the minimum redundancy levels needed by each subsystem.

MISSION COSTING

Table 1-2. Cost Breakdown By Subsystem (in millions of FY 89 $)

SUBSYSTEM	WEIGHT (kg)	DDTE COST (million $)	PROD COST (million $)	TOTAL COST (million $)
STRUC	2720	$57.8	$24.1	$81.8
RRS	680	$32.7	$10.4	$43.1
PPS	1920	$1.1	$136.4	$137.5
AACS	170	$158.8	$18.9	$177.7
ECLSS	530	$0.8	$29.3	$30.0
CDC	65	$11.5	$32.4	$43.9
TOTALS	6085	$262.7	$251.4	$514.1

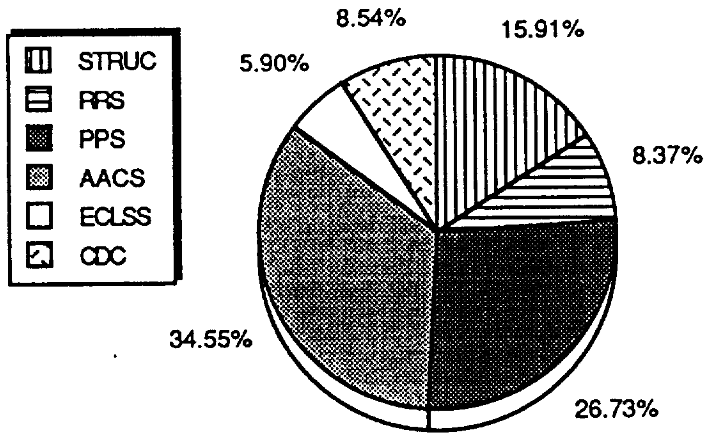


Figure 1-5. Total Cost of 1 LiBERTy Vehicle by Subsystem
($514M Fy 89)

Table 1-2 outlines the cost of one LiBERTy vehicle broken down by subsystem. The values were determined using the NASA Cost Estimating Relationship (CER) equation for the Space Station and future manned planetary missions. There was some difficulty in placing certain systems into one of the subsystems listed in the CER breakdown, so there were some assumptions made. The Reentry and Recovery Subsystem was considered, for costing purposes to fall under environmental protection. Also, since a closed-loop ECLSS system was not used, the ECLSS weights were included in the propulsion system; it was felt the manufacturing and technologies in producing propulsion tanks and life support tanks was similar enough to justify this move for costing purposes. The reaction control system (RCS) fuel requirements were specified by the AACS Analyst, and then added to the propulsion weights. Next, the design, development, testing, and engineering (DDTE) and production costs were calculated for all of the subsystems listed in Figure 1-5. In determining the costs of the seven subsystems outlined in the RFP, the docking module costs were added to the structures subsystem, thermal protection was lumped into ECLSS, and RCS and electrical power were included in the propulsion subsystem. The cost of four LiBERTy vehicles (three operational vehicles and one for integrated ground testing) is $1.3 billion dollars (FY 89); see Appendix 1A for calculations.

INTEGRATION WITH OTHER SUBSYSTEMS

The Mission management subsystem required constant interaction with the other subsystems. Material selection was discussed with the Structures Analyst to find materials were at once cost effective and provided adequate safety and structural efficiencies. Once the ELV was chosen, the Structures Analyst needed know the maximum payload capacity of the Titan IV for sizing purposes. Selection of the propulsion and electric systems were discussed with the PPS Analyst to ensure the fuel was cost effective and easily handled. Δv requirements for the mission were worked out with the help of the RRS Analyst. As explained earlier in the paper, many considerations required the interaction of the mission planner and the ECLSS and RRS Analysts. The CDC Analyst needed to know when controls should to be implemented. Communication distances were also a big concern for the CDC Analyst. This information proved helpful in determining antenna size and control system pointing requirements.

PROBLEM AREAS

The fact that none of the ELVs studied was capable up transporting both LiBERTy and the total 90-day logistics resupply created some difficulty in formulating the mission timeline. The additional $500 million required to launch ten Titan IV's may not prove to be cost effective. Also, limitations on payload mass also led to the decision to forego the requirement of an attachable orbital transfer propulsion system; this will be discussed further in Section 3.

System costing proved to be difficult, because LiBERTy's subsystems were hard to fit into the CER categories. Some trade-offs and assumptions had to be done to get all of the subsystems costed. In addition, even though one of the spacecraft is to serve as a ground test system, costing was done as if all four spacecraft were to be used for space flight. The calculated costs give a good, rough estimate into some of the costs that will be incurred, but they should not be taken as actual figures.

APPENDIX 1A

Cost Estimating Relationship (CER) calculations:

Cost = $A\,(WGT_{ss})^{B}\,(1.2)$ where: A = CER coefficient
B = CER exponent
WGT_{ss} = Subsystem weight (kg)
(1.2 is the escalation index from FY 84 to FY 89 dollars)

Table 1-3. Cost Estimating Relationships by Subsystem (in millions of FY 89 $)

SUBSYSTEM	WEIGHT	DDTE A	DDTE B	PRODUCTION A	PRODUCTION B	DDTE COST	PROD COST	TOTAL COST
STRUCTURES	2560	1.76	0.42	0.49	0.44	$57.0	$18.6	$75.6
ENV. PROT.	680	1.76	0.42	0.49	0.44	$32.7	$10.4	$43.1
DOCKING MODULE	160	0.45	0.06	0.49	0.44	$0.7	$5.5	$6.2
ELECTRICAL POWER	420	0.57	0.04	0.58	0.78	$0.9	$77.4	$78.3
CDC	65	7.81	0.05	0.58	0.92	$11.5	$32.4	$43.9
RCS	120	0.1	0.11	0.88	0.55	$0.2	$14.7	$14.9
PROPULSION	1500	0.1	0.11	0.88	0.55	$0.3	$59.0	$59.2
AACS	50	4.57	0.86	0.52	0.49	$158.6	$4.2	$162.8
THERMAL CONTROL	210	0.35	0.05	0.42	0.39	$0.5	$4.1	$4.6
ECLSS	320	0.1	0.11	0.88	0.55	$0.2	$25.2	$25.4
TOTALS	6085					$262.7	$251.4	$514.1

DDTE = Design, Development, Testing, and Engineering

Total Cost for 4 LiBERTy vehicles = $COST_{DDTE} + 4{*}COST_{PROD}$
$= \$262.7 + 4{*}(251.4) = \boxed{\$1.3 \text{ billion (FY 89)}}$

ANNUAL OPERATING COSTS

In addition to the cost of each LiBERTy vehicle, each Titan IV will cost $110 million. According to figure 1-4, 10 ELVs will be launched during a year. Therefore, $1.1 billion will be spent annually on ELVs alone. This figure, however, does not consider all the costs that will be incurred during the year. Taking these factors into consideration then gives an estimated six year lifetime cost of the project of $8 billion dollars.

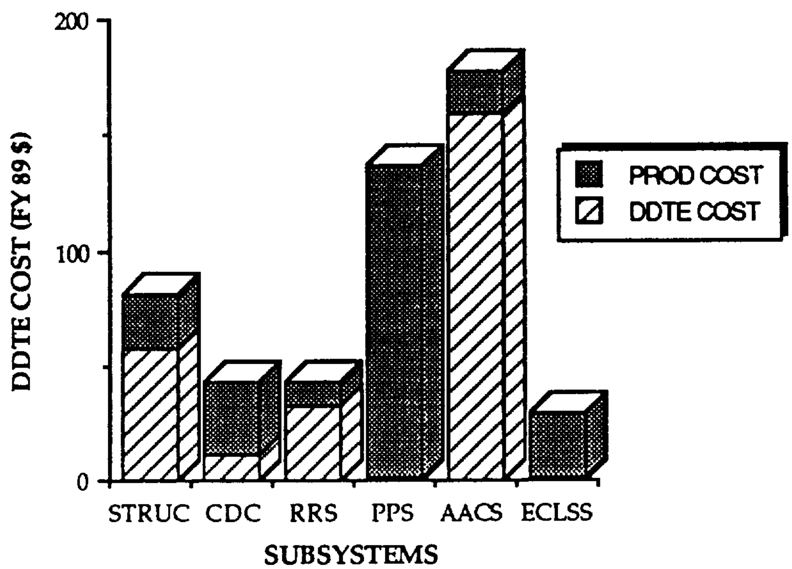


Figure 1-6. DDTE and Production Costs by Subsystem

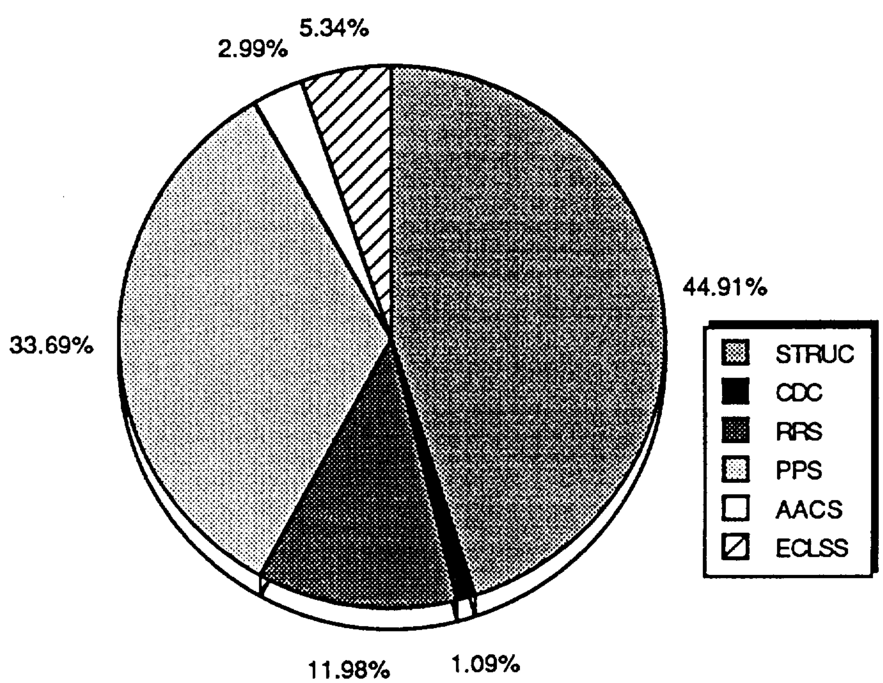


Figure 1-7. Total Weight Break Down By Subsystem
(6085 kg)

REFERENCES

1–1. *ACRC-CERV Option Reference Concept* (JSC-23264). Johnson Space Center, National Aeronautics & Space Administration. 15 November 1988.

1-2. Avvento, Gennaro J. *Reusable Unmanned Automated Resupply Freighters for Space Station Operations*. AIAA Technical Paper 87-0687. March 1987.

1–3. *Conceptual Design Study of a Crew Emergency Return Vehicle* (JSC-23103). Advanced Programs Office, JSC, NASA. August 1988.

1–4. Gerlach, Ronald H. (CERV Office, JSC, NASA). Phone conversation. April 1989.

1–5. Koepke, Andrew (Department of Aeronautical & Astronautical Engineering, University of Illinois). Conversations. Spring Semester 1989.

1–6. Lembeck, Michael F. (Department of Aeronautical & Astronautical Engineering, University of Illinois). Aerospace Vehicle Design Course -- AAE 241 lectures and conversations. Spring Semester 1989.

1–7. "Space Shuttle News Reference". NASA. 1978.

STRUCTURE SUBSYSTEM

Rommel B. Villlalobos

Introduction

The Structures Subsystem is responsible for the physical support of all other subsystems during each mission and during the lifetime of the module.

Requirements

The requirements from the Request for Proposal are as follows:

1. provide thermal control for all components
2. provide the structure necessary for all the components
3. locate all components (utilizing INERT)
4. protect against micrometeorite impact/radiation
5. materials should have a lifetime of at least six years
6. design will stress simplicity, reliability, and low cost and use technology available prior to 1995

Conceptual Design

The method of attack for completing the subsystem was not very complex. A derived requirement for the vehicle is the ability to be returned within the Shuttle bay. Knowing the Shuttle bay has a 15 ft. diameter, the design could not exceed a maximum of 14 ft. The left-over space will be utilized for structure to support the vehicle.

The layout of the components was a series of compromises. Each subsystem component must be positioned in such a way as to achieve the best inertial configuration. In addition, they had to be situated to achieve optimal flotation ability. Using a program called INERT (Reference 2–4), this task was carried out.

Protection against micrometeorite impact/radiation was taken into consideration but not deeply explored. Radiation should not be a problem, but micrometeorites will be. To protect against impacts, a double-wall configuration was chosen.

The fabrication of materials employed will use present-day technology. If new technology is developed by 1995 to manufacture and form materials, it could be utilized.

Studies

During the course of the design process, numerous studies were made to determine which configurations, materials, and components would be best. Studies which compared capsules vs. flared cylinders, materials vs. mass/cost, safety vs. mass, and strength vs. mass/cost were conducted.

The first study was the analysis of capsules vs. flared cylinders. After reading different NASA reports on CERV configurations, it was found NASA preferred using capsules. However, volume is a major drawback. The capsule's flaw is that it would not be large enough to accommodate the volume required for logistics. This is the main reason a flared cylinder was chosen above the capsule. A cylinder has adequate volume and can also accommodate enough crew members for an emergency return from space, if necessary.

Another study conducted was materials vs. mass/cost. It was started by looking at the materials utilized in the Shuttle. The reinforced carbon-carbon and heat tiles were found to be the best for our purposes during reentry. In addition, it was discovered the aluminum used was very good. Titanium was looked into for use in the structure but was quickly ruled out because of its high cost. Eventually, it was decided the materials used on the Shuttle would be best. They have all been tested and proven in space.

A third study is safety vs. mass. Spalling was a major factor to consider. Also, it can be easily seen that a dual wall structure is far superior to the single wall configuration. A further explanation of the dual wall configuration will be found in the Technical Approach/Skin subsection.

A final study conducted was strength vs. mass/cost. At this point, the materials which had good strength-to-weight ratios were examined. As was previously stated, titanium and aluminum were explored. Aluminum was chosen for its lower cost. Its strength-to-mass ratio is perfect for LiBERTy.

Technical Approach

As previously stated, it was decided the materials used in the Shuttle were the best that could be found. The Shuttle uses materials which have tested and proven in space, are readily available, and are easy to use. These are important factors because the RFP stressed simplicity and reliability. Figure 2–1 shows LiBERTy's over-all design.

Inner Structure

Aluminum-2024 was chosen for the skeleton because it has a good strength-to-weight ratio and its relatively inexpensive. Also, it had a relatively low mass per unit volume. A layout of the struts is shown in Figure 2-2. The vertical and horizontal struts are made of Al-2024 and a cross section of the struts can be seen in Figure 2-3. The model accompanying this report also shows how the struts are placed.

Skin

The skin is directly attached to the skeleton in the usual manner, using rivets. The first layer of the skin is similar to the one used on the Shuttle. The skin is thin aluminum, 1.5 cm thick. Nomex felt is attached to this using RTV 560 adhesive. The felt is treated to make it water-proof and to provide thermal protection below 371° C. When the skin and thermal tiles are implemented, they act like a dual-wall. This is advantageous because micrometeorite impacts will not cause problems. The adhesive acts similar to cork or other fillers in wall structures.

Thermal Tiles

Along with the Nomex felt and aluminum skin, reinforced carbon-carbon and high- and low- temperature reusable tiles will be used for further thermal protection. Reinforced carbon-carbon (RCC) is placed at the tip of the nose, the area which experiences the most heat during reentry. RCC can withstand temperatures passed 1260° C. The high-temperature reusable surface insulation (HRSI) is used for the lower surfaces and most of the nose. This is the area which experiences the most heat during reentry. These tiles can withstand temperatures up to 704° C. The low-temperature reusable surface insulation (LRSI) is applied to the upper surfaces because it experiences lower temperatures during reentry. It can withstand temperatures up to 649° C. The area directly behind the berthing ring will have no tiles attached to it. Instead, the Nomex felt will be used but its thickness will be increased, from 1.5 cm to 5 cm, for better protection. In this area are located sensors and blow off doors for the parachute. The configuration of the skin and heat tile and placement of the tiles can be found in figure 2-4.

Implementation

Taking all of the components for the structure and arranging them, total weight for the structure alone is approximately 2500 kg (see appendix 2B). Figure 2-1 shows the overall layout of the module. Also included in the figure is the placement of some components (others have been omitted for clarity). Figure 2-5 is a more detailed layout of the batteries, computer, avionics, and life support. A more detailed explanation of each of these components can be found in their respective subsystem sections.

Figure 2-6 shows the area of the berthing ring and the components directly behind it. The berthing ring used is similar to the one used on the SCRAM. It can be seen in Figure 2-7. Along with the berthing ring, a flotation device is placed along side to ensure the hatch is above the water after splashdown. The SARSAT beacon, antennas, star scanner, and parachute packs are shown. Each of these components are explained further in their respective subsystem sections.

In Figure 2-8, the propulsion system and other sensors are shown. Also, an escape hatch will be installed for emergency egress. The hatch will have explosive bolts and will be used only in extreme emergencies. The propulsion system uses MMH as its propellant and are stored in space Shuttle APU tanks. The propulsion system and the coordinating tanks are further explained in the power and propulsion section. The TDRSS-link antenna is also located in this area. It will be mounted on a telescoping-swivel arm. A further explanation of the TDRSS-link can be found in the command and data control section.

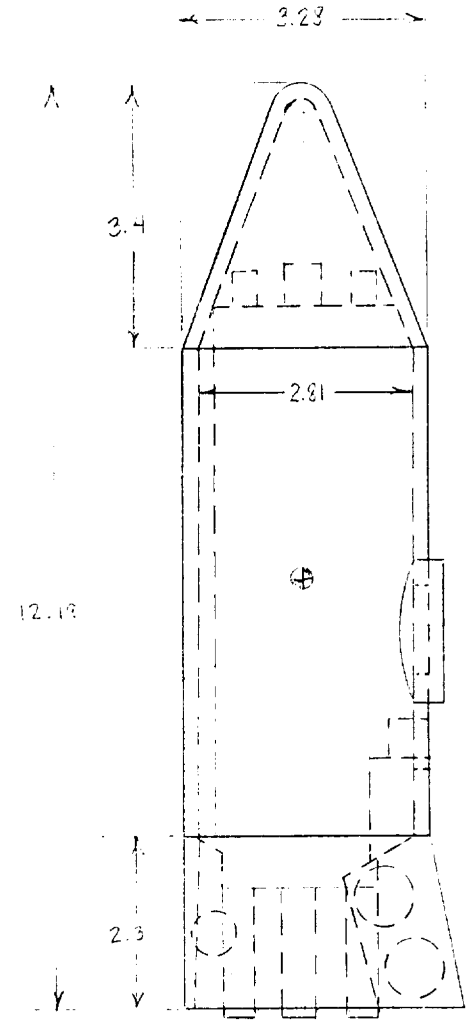
Figure 2-9 shows a rough concept on how the seats for the emergency return will be implemented and stored. These seats are similar to the ones used on the SCRAM. The seats will be collapsed and stored in the rear of the module. The specifications on the seats can be found in the life support section.

With all the components and different elements in place, we used the INERT program to compute the module's overall inertia and locate its center of gravity. The INERT program gave the following inertia matrix (see appendix 2B). In addition, the center of gravity is located at 6.645 m away from the nose. This is very valuable information during reentry and especially during rendezvous with the space station.

Conclusion

During the entire mission, the structure subsystem will provide all the necessary support for the components. As the mission continues, the structure must still withstand the landings, launches, and space station docking for at least six years. Therefore, the components chosen have all be proven to be able to handle such a mission. In conclusion, it must be realized that this is only a conceptual design for a logistics module and emergency return vehicle. Further study into the project must be made in order for any module to completely fulfill it. In the future, LiBERTy must be refined to better meet the specifications that could be introduced.

Appendix 4A
(all measurements in meters)



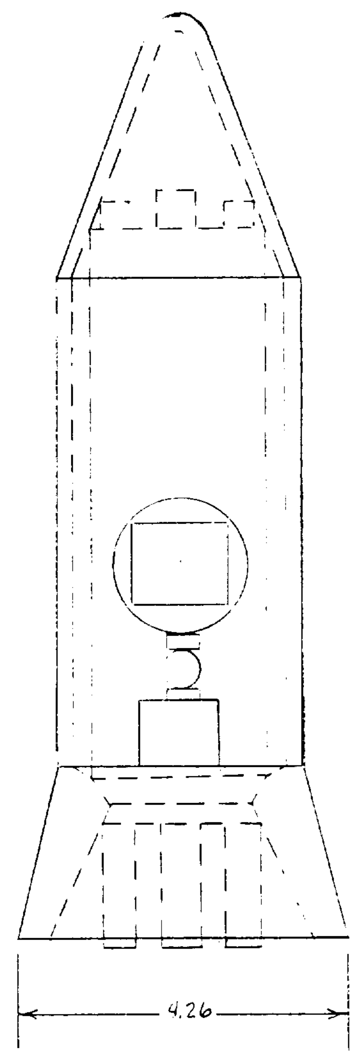
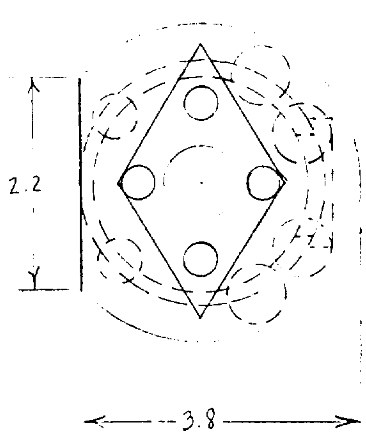


Figure 2-1. Overall Design Layout



* **Some components have been omitted for clarity**

ORIGINAL PAGE IS
OF POOR QUALITY

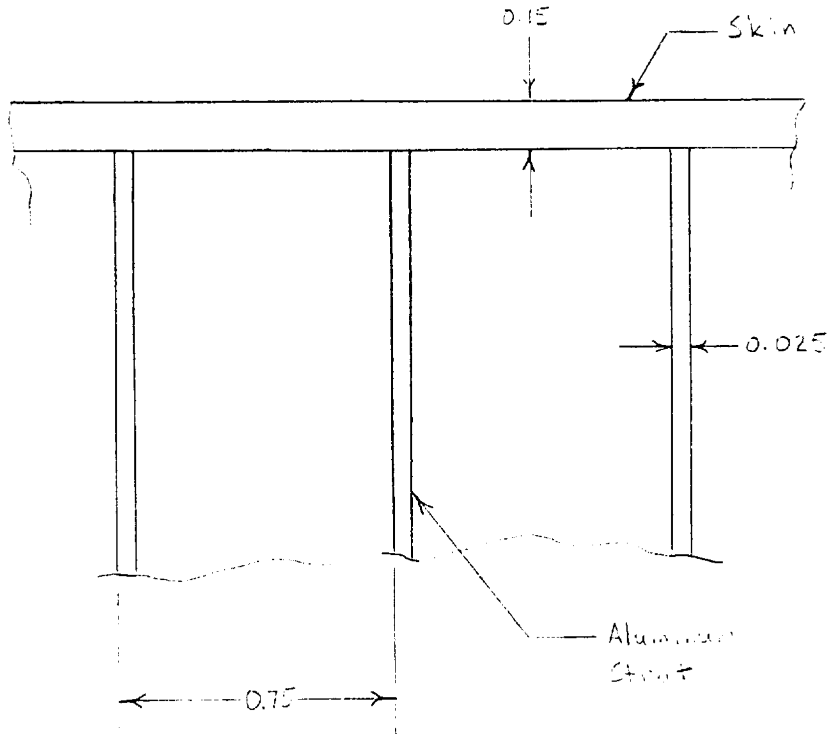


Figure 2-2. Layout of Struts

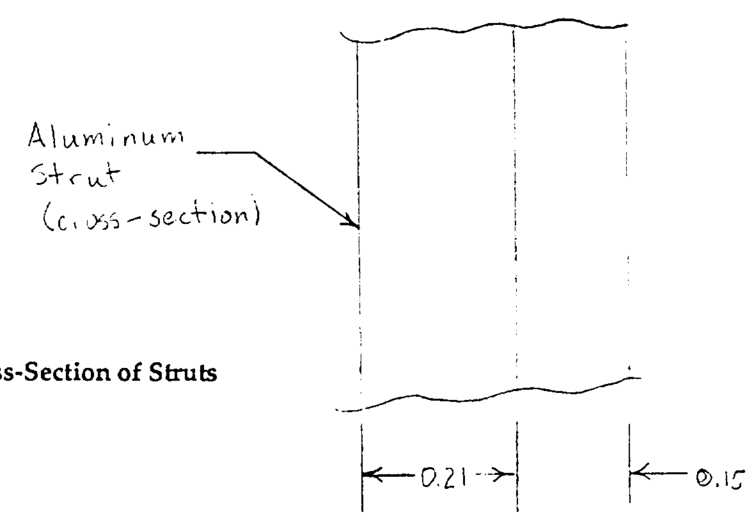


Figure 2-3. Cross-Section of Struts

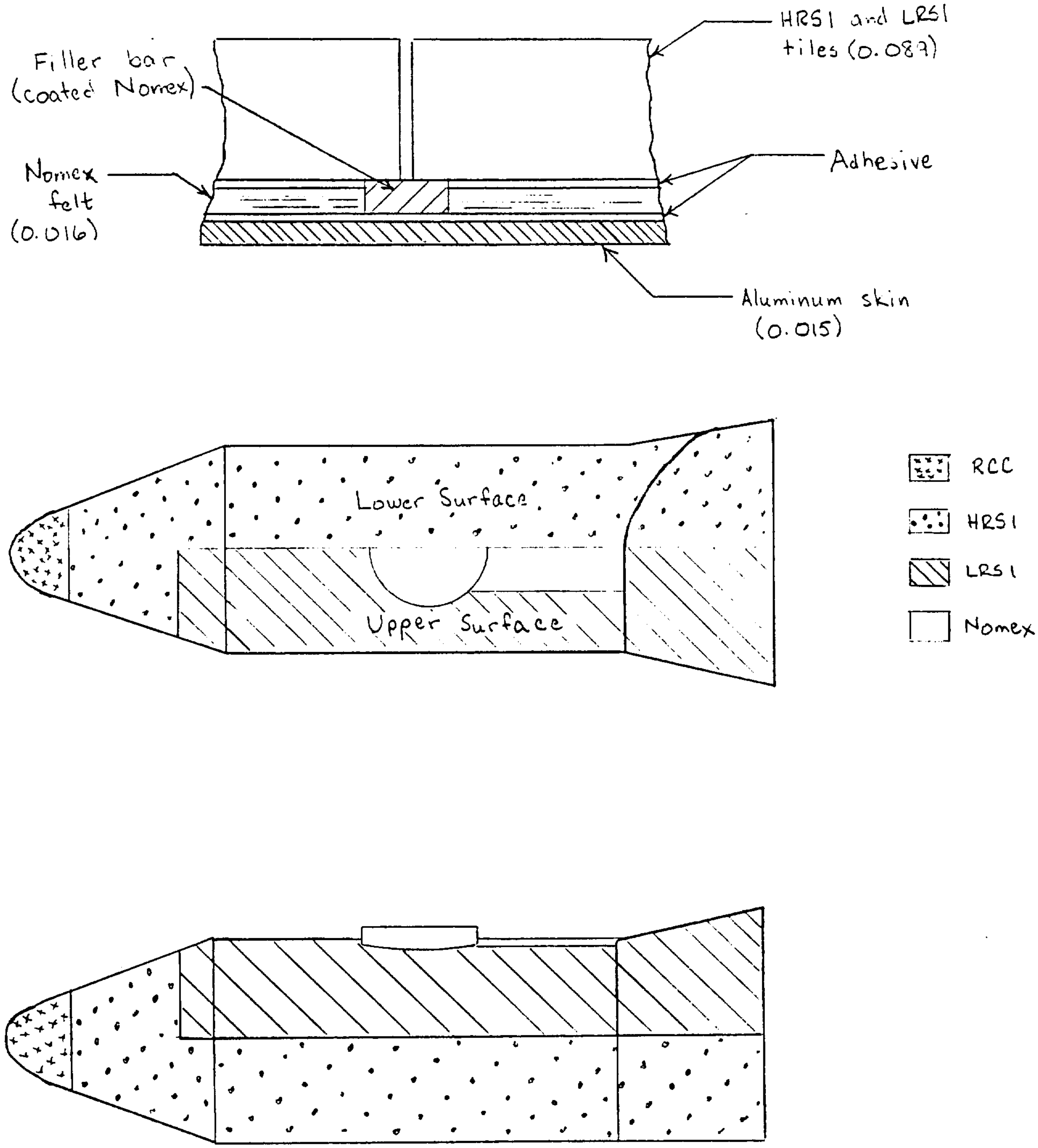


Figure 2-4. Skin configuration and Tile Placement

* Some components have been omitted for clarity

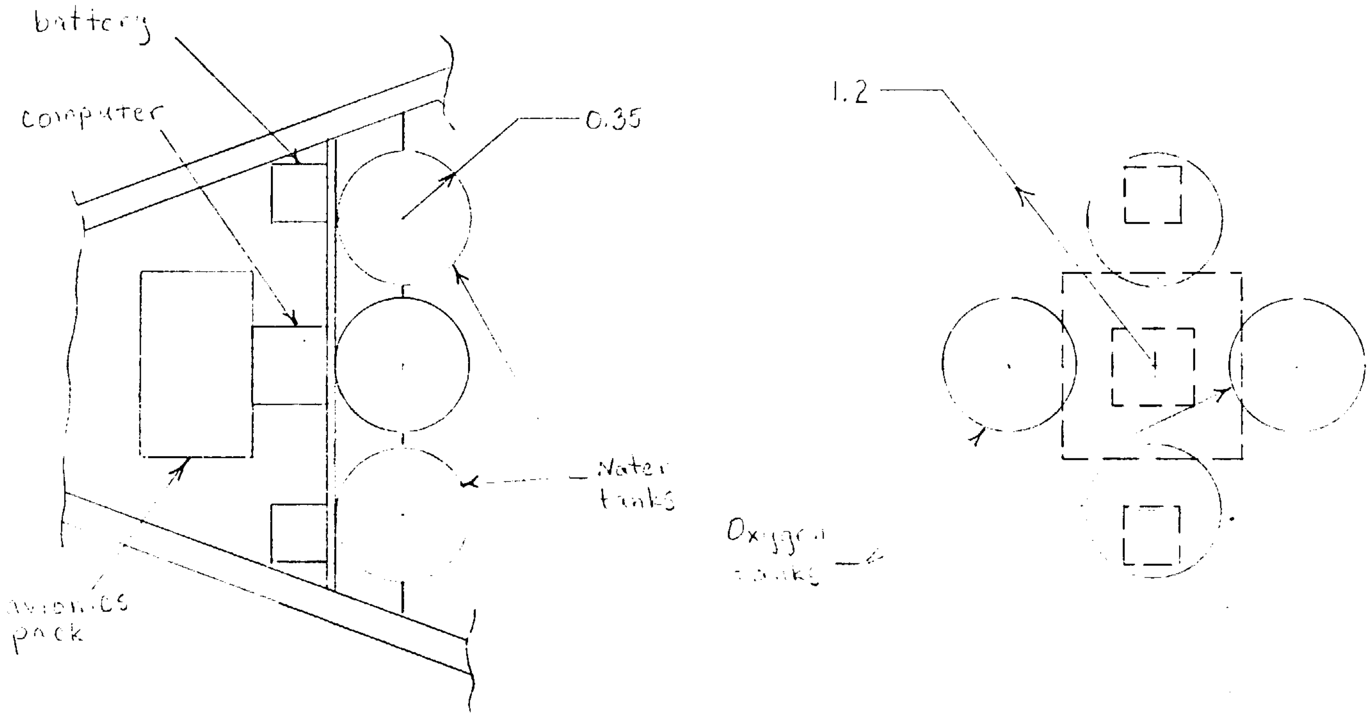


Figure 2-5. Detail of Batteries, ECLSS, Computer, and Avionics

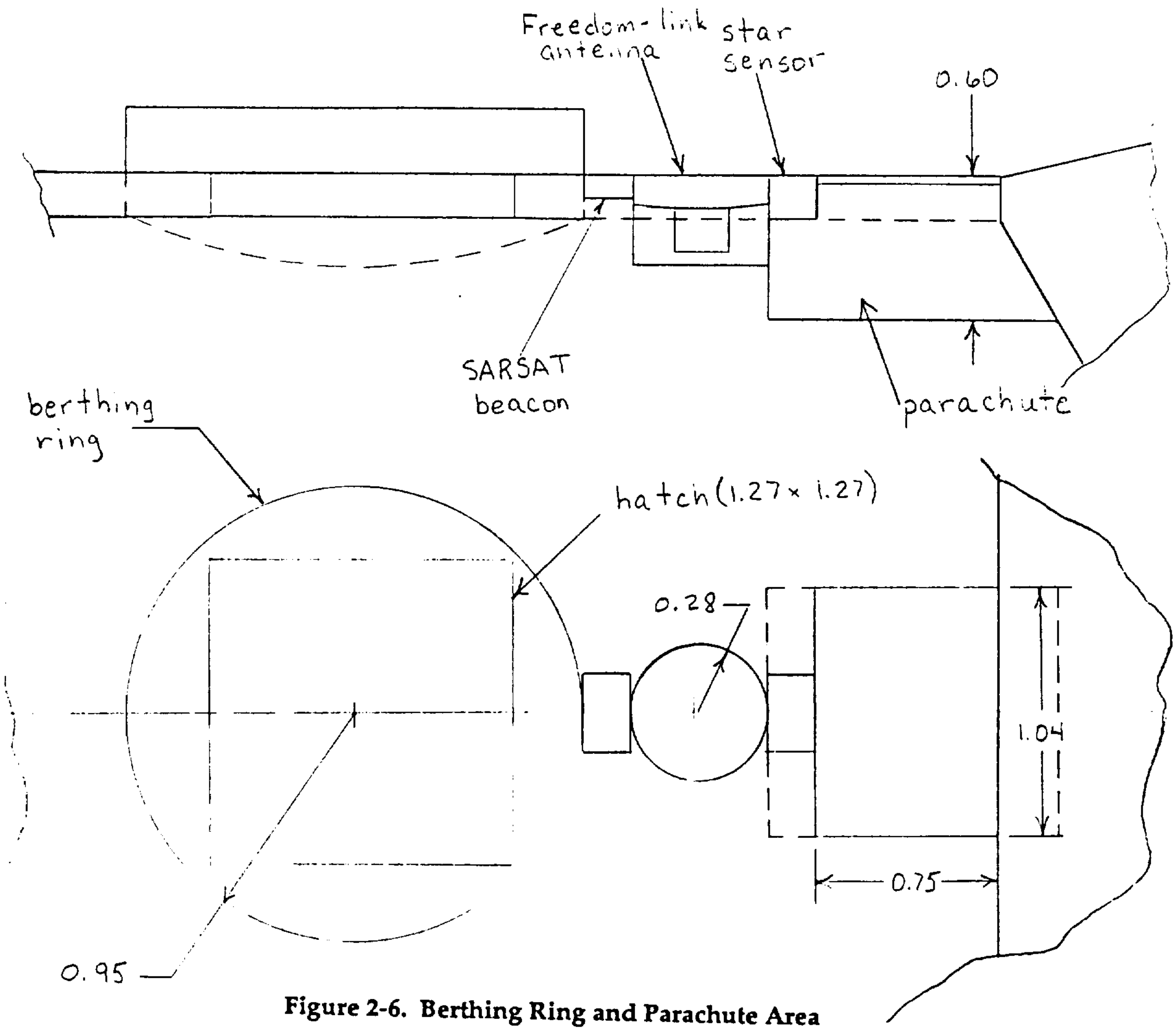


Figure 2-6. Berthing Ring and Parachute Area

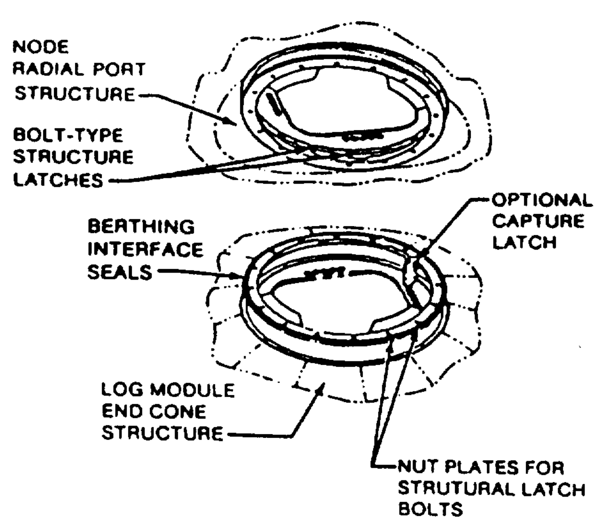


Figure 2-7. Berthing Ring (Ref. 2.2)

ORIGINAL PAGE IS
OF POOR QUALITY

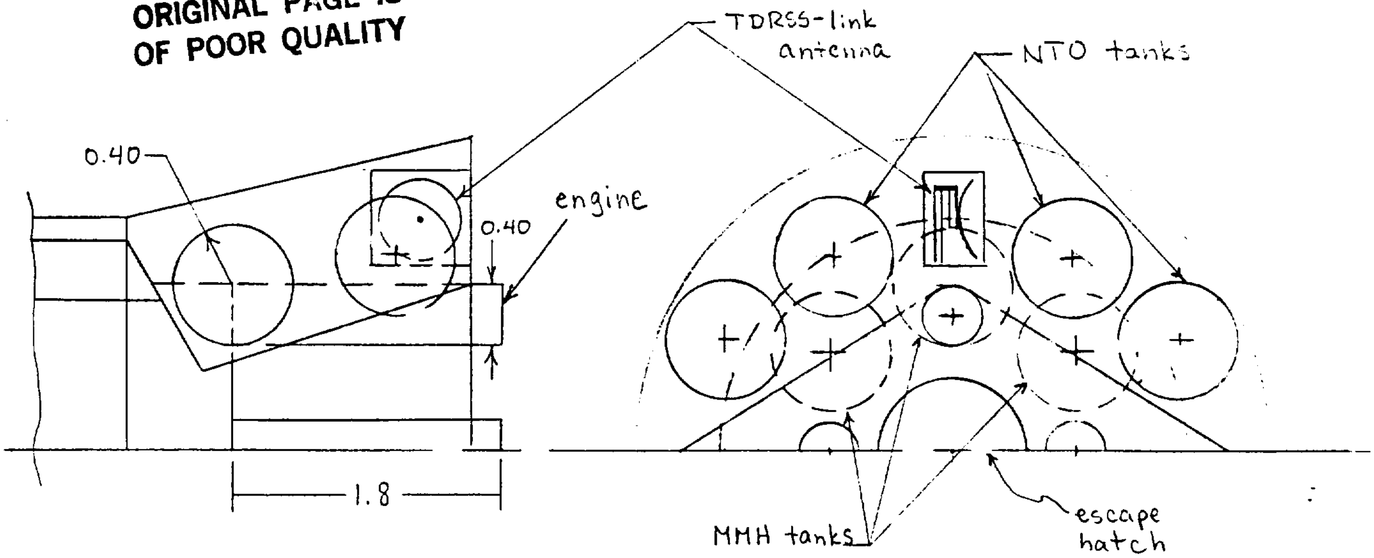


Figure 2-8. Detail of Propulsion System

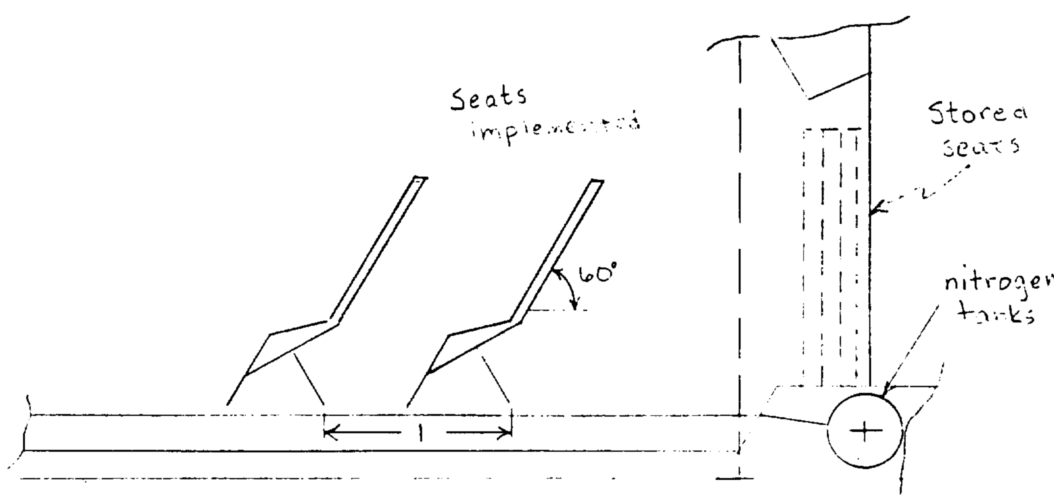


Figure 2-9. Possible Seat Placement

Appendix 2B

Table 2-1. Structural Materials Summary

Aluminum 2024 (clad)	276.729 kg/m^3
RCC	36.0789 kg/m^2
HRSI	8.0209 kg/m^2
LRSI	3.4228 kg/m^2
Nomex felt	1.561 kg/m^2

Table 2-2. Masses (kg)

Al-2024	1624.445
RCC	276.217
HRSI	282.106
LRSI	120.356
Nomex felt	93.3954
Miscelaneous	160

Totals	2550 kg

$$\text{Total inertia matrix} \quad \begin{bmatrix} 20993.1852 & -9.4842 & -1.0400 \\ -9.4842 & 3489.3821 & -1418.520 \\ -1.0400 & -1418.5207 & 21939.227 \end{bmatrix}$$

REFERENCES

2.1 AIAA Aerospace Design Engineers Guide, American Institute of Aeronautics and Astronautics, New York, 1987.

2.2 JSC 23264, "ACRC-CERV Option Reference Concept," Job Order 052-PA-301, Johnson Space Center, 15 November 1988.

2.3 Lembeck, Mike, "Thermal Analysis," AAE 241 Lecture Notes #17, University of Illinois AAE 241 Flight Vehicle Design Class, 1989.

2.4 Lembeck Mike, "Inert Program," AAE 241 Lecture Notes #11, University of Illinois AAE 241 Flight Vehicle Design Class, 1989.

2.5 Reding, Peter J., Lifting Entry Rescue Vehicle Configuration, Lockheed Missiles & Space Co., Inc., Sunnyvale, CA. August 1988.

2.6 SPACE SHUTTLE News Reference, NASA, Johnson Space Center.

POWER AND PROPULSION SUBSYSTEM

– RICHARD C. GIANVECCHIO

INTRODUCTION

Once the structural design of LiBERTy has been decided it is necessary to provide a means to propel logistics to and from Space Station *Freedom* and deliver any astronauts back to Earth in case of an emergency. The main objective of the power and propulsion subsystem is two-fold. In effect the subsystem itself can be broken up into two subsystems, a power subsystem and a propulsion subsystem. Each subsystem has its own set of requirements that were used to determine the design. For the propulsion subsystem the requirements are to provide the necessary propulsion to move LiBERTy from the Space Station back to Earth and make any necessary orbits to other platforms. This system was to be protected from potential failure and made safe during the re-entry maneuver. The power subsystem onboard LiBERTy also had a set of requirements upon it that determined the design. LiBERTy was to have a source of constant power to be distributed to the other subsystems and this source was to be safeguarded from failures. With the requirements that the design was based on known the design of LiBERTy's power and propulsion subsystem could be made.

PROPULSION SUBSYSTEM

The propulsion subsystem onboard is designed so that the requirement of orbit changes was one that can not be met if the primary task of LiBERTy is to supply logistics to Space Station and return astronauts in case of emergency. This is do to the fact that in order to make out of plane orbit changes the fuel needs would be so large that the vehicle would contain mostly fuel in order to accomplish a one-way maneuver of only a couple of tens of degrees. This can be seen by table 3-1 on the next page.

TABLE 3-1. FUEL REQUIREMENTS OF 90 DEGREE PLANE CHANGE

Total Mass of LiBERTy [kg]	15000
Orbit Altitude (Initial) [km]	290
Change in Orbit Inclination [Degrees]	90
Velocity at Orbit (Initial) [km/s]	7.73
ΔV for Inclination Change [km/s]	10.934
Mass of Fuel Needed for Inclination Change [kg]	14588
Volume of Fuel for Inclination Change [liter]	14140

Because of the amount of fuel required, it is necessary to remove this requirement and therefore the requirement for all orbital maneuvering has been removed. The propulsion subsystem for LiBERTy was designed to provide the necessary propulsion for re-entry based on a velocity change of 0.1524 km/s and a downward mass of 13,000 kg. The necessary calculations are found in Appendix 3-A.

A review of alternate propulsion subsystem techniques resulted in the final selection of a bi-propellant system, similar to that used on Shuttle, using Monomethylhydrazine (MMH) as fuel and Nitrogen Tetroxide (NTO) as the oxidizer at a mixture ratio of 1.65. Components were selected based on availability and proven reliability.

COMPONENT OPTIONS

Solid Propellant: This type of propulsion system has been ruled out. Although solid rocket fuel is easily storable for extended periods of time, it is not possible to throttle such an engine design and throttling capability is necessary for LiBERTy.

Gaseous Propellant: This type of propellant system is not used anymore. Due to restrictions on weight this is not a feasible method. The use of a gas as a working fluid requires the use of heavy tanks.

Liquid Propellant: This type of propellent can use easily storable liquid fuel and is capable of being throttled. It is usually split into mono-propellant and bi-propellant types. Current U.S. spacecraft use a bi-propellant system and that is why a bi-propellant system was chosen for LiBERTy. The most common type of system is a liquid oxygen and liquid hydrogen system, but due to the long storage times that are required when LiBERTy will be in orbit, and the fact that oxygen and hydrogen will leak out slowly over time, it has been determined that such a bi-propellant system was not feasible. Monomethylhydrazine and Nitrogen Tetroxide is the fuel combination used to propel Space Shuttle with the Orbital Maneuvering System (OMS). Since both of these can be stored indefinitely and provide a relatively high specific impulse with good thrust, this was the system chosen.

COMPONENT SELECTION

The selected configuration consists of three (3) 0.7112 m MMH tanks and solenoid valves from the shuttle Auxiliary Power Unit (APU) subsystem. Four (4) similar tanks are used to store NTO. Two (2) 0.3048 m pressurized nitrogen tanks of total filled mass of 65 kg are used for the system purge cycle and for filling the propellant lines with an inert gas during launch to the space station and storage there. Four (4) 3340 N engines with a specific impulse of 260 seconds have been selected. Seven (7) normally-closed, pyrotechnically-open, pyrotechnically-closed valves have been added to isolate the propellant tanks during storage on-station and to isolate the tanks during re-entry and recovery. The subsystem has a 20% fuel contingency reserve and power rating are determined from simultaneous operation of all valves and tanks. Total MMH mass is 136.67 kg per tank and total NTO mass is 123.75 kg per tank for a total mass of MMH equalling 410 kg and NTO equalling 495 kg.

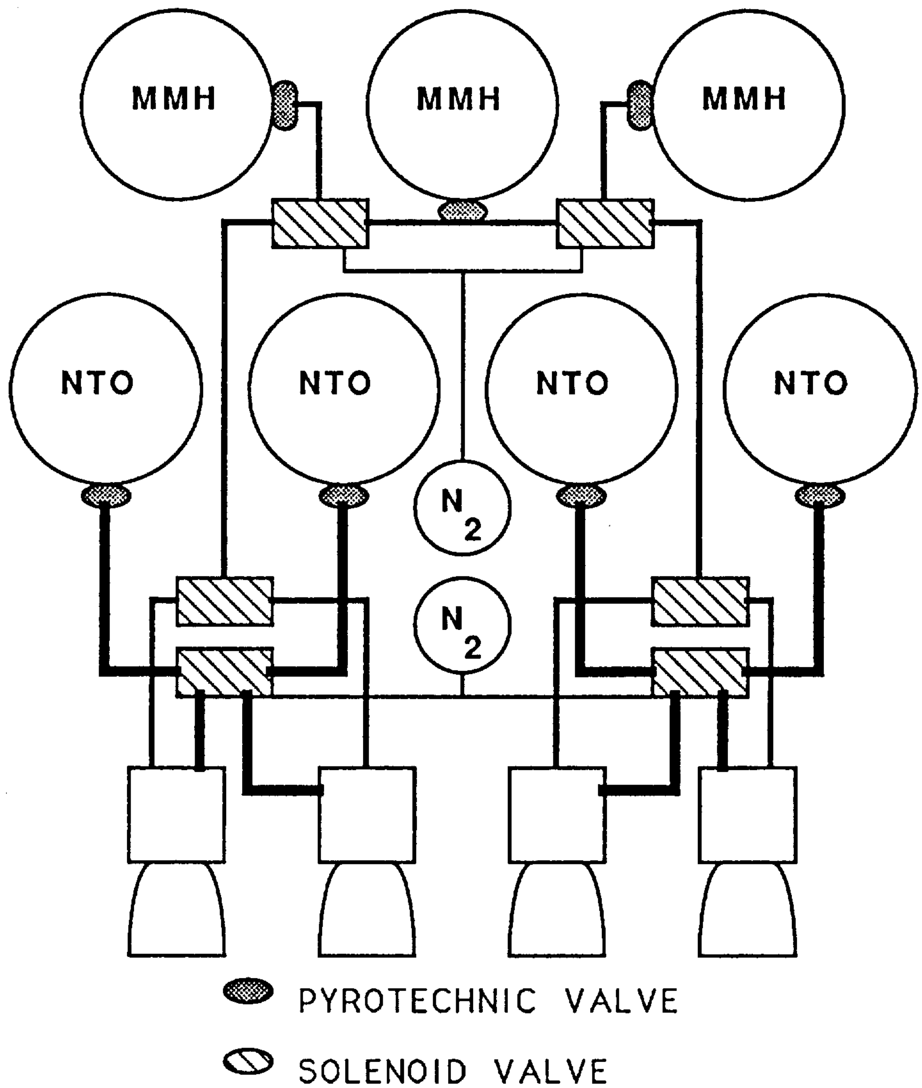


FIGURE 3-1. CONFIGURATION OF PROPULSION SYSTEM

CONFIGURATION

The proposed configuration of the forementioned propulsion system is seen in figure 3-1.

REDUNDANCY STRATEGY

The redundancy strategy of the propulsion subsystem is designed to tolerate the loss of a single string, i.e., a tank, valve set, line set, or engine and still be able to safely continue operations. Additional redundancy is provided to tolerate two failures.

POWER SUBSYSTEM

The power subsystem of LiBERTy was designed upon the requirements that it must provide the necessary power to keep all other subsystems of LiBERTy functioning. It must accomplish this safely and be able to handle faults to a limited capability. For LiBERTy the power has been chosen to be supplied by Lithium BromineComplex cells that are expected to be operational for manned space flight by 1994. This falls within the technology requirements on this design. This also allows the weight and volume of the power system to be minimized. It was decided that the use of solar energy collection devices would not be used because of their cost and the complexities involved in their deployment.

POWER REQUIREMENTS FOR OTHER SUBSYSTEMS

The power requirements of each subsystem are taken from peak power needed to run them, since power must be available at any time. The numbers seen on table 3-2. represent this.

TABLE 3-2. POWER PROFILE OF LIBERTY

SUBSYSTEM	30 HOUR ON-ORBIT	7 HOUR POST-ORBIT
PPS	560 W	NONE
ECLSS	300 W	15 W
AACS	240 W	NONE
CDC	225 W	5 W
TOTAL POWER	1325 W	20 W

Based on these numbers the total power needed is 39,890 WHr and from this number it has been determined that eight (8) battery packs of volume .0122 m^3 and mass 27.250 kg will be used, each supplying 270 AHr. The calculations that led to these numbers are found in Appendix 3-B. The idea of using battery packs was decided because it makes the system more flexible. By being able to add and remove battery packs, whether for replacement or to supply extra power, the system is made less complex and a failure of a single battery pack will not result in replacing the entire power system.

COMPONENT OPTIONS

Battery Type: A battery system for liberty could be either a system that relied on both solar arrays and batteries, or a system that relied on only solar arrays or batteries. Since LiBERTy will be orbiting the Earth by Space Station, the Sun would be unable to provide a constant source of energy, and a system based solely on solar arrays was ruled out. A combination of batteries and arrays was also ruled out due to complexity and cost. The ability to deploy such a system on a vehicle of LiBERTy's shape was determined as being too difficult to deal with, and since the requirements ask for a simple system, a combined power system was dismissed. Therefore a system that depended entirely on batteries was chosen, as it is much simpler to control and maintain, and was somewhat cheaper. The decision was now left as to what type of batteries would be selected. In 1994 it is projected that experimentation on a new battery type would be completed. The battery is a Lithium BromineComplex cell and it provides 3 volts per cell and 2.7 AHr per cell. This is currently much higher than any existing battery cell. For this reason the Lithium BromineComplex cell has been chosen.

COMPONENT SELECTION

Along with the batteries to supply the power, components are needed to make the system safe and provide a means of distributing the power to the other systems. Two power control units (PCU) have been installed and are able to function either by themselves or simultaneously. Monitored by either onboard personnel, ground link, or by Space Station, the PCU's take the generated power and distribute it to where it is needed. The PCU's have a volume of .0531 m^3 each and a mass of 27 kg each. The load distribution assembly and the wiring between systems account for an additional mass of 145 kg and volume of .909 m^3. The PCU's contain diodes to prevent cell reversal and charge buildup due to external sources. Fuses are used to protect the battery packs from a shorted battery PCU, and circuit breakers will protect the PCU from excessive charging or power loads. A schematic setup of the power distribution system is shown in figure 3-2.

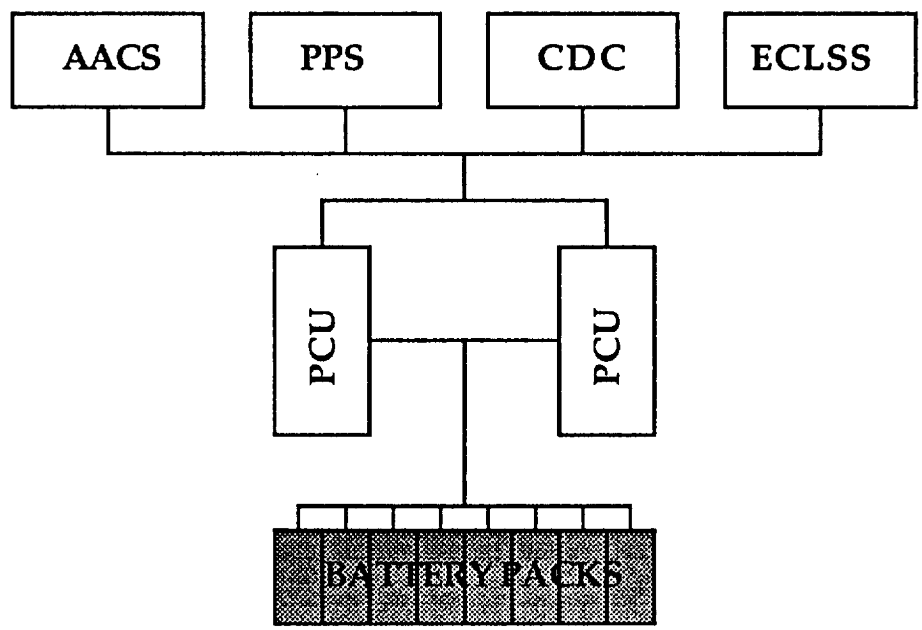


FIGURE 3-2. POWER DISTRIBUTION SYSTEM

REDUNDANCY STRATEGY

The power subsystem onboard LiBERTy has multiple diodes, fuses, and circuit breakers to safeguard itself. Multiple PCU's are used in case of failure of either one. Sufficient redundancy has been taken into account in battery sizing. The batteries are capable of providing 120% of the power requirements of LiBERTy and take into account shelflife degradation of the batteries.

PROBLEM AREAS

Throughout this subsystem analysis many assumptions such as component size and mass were estimated using percentage fractions of other existing similar systems (ref 3.3 , 3.4, and 3.5).

There exist great problems in determining from outside sources exactly how the plumbing of a propulsion system would be setup. The same difficulty is found in sizing and configuring the wiring and electronics for the power subsystem. The engines used in the propulsion system are smaller scale OMS engines limited to but a few firings and therefore are lighter and less costly than the OMS engines that they are derived from. They do not exist as yet, as far as could be determined, and thus would have to be developed for the mid 1990s.

CONCLUSION

In conclusion the power and propulsion system design set forth in this document is based on past systems, such as Space Shuttle OMS propulsion and APU tanks, along with the benefit of new technology to decrease weight and cost, such as the use of Lithium Bromide Complex power cells. It must be realized that this is a conceptual approach to the development of such a system and that what is printed here is a possible approach to the problem and is solved as best as could be within the given timeline. With the power and propulsion aspect of LiBERTy taken care of, the Attitude and Articulation Subsystem can be discussed.

APPENDIX 3-A

The propulsion system sizing requirements were based on the following calculations:

Downward mass of vehicle = 13,000 kg

Specific impulse of each thruster = 260 s

Re-entry $\Delta V = 0.1524$ km/s

From this information one can use Tsiolkovsky's equation to determine the mass of fuel necessary to accomplish this maneuver.

ΔV = (specific impulse)(gravitational constant) ln [Mtotal/(Mtotal - Mfuel)]

or

$0.1524 \text{ km/s} = (260 \text{ s})(.00981 \text{ km/s}^2) \ln [13{,}000 \text{ kg}/(13{,}000 \text{ kg} - \text{Mfuel})]$

This yields a Mfuel of 754 kg , and accounting for 20% contingency fuel Mfuel = 905 kg

The mixture ratio of MMH to NTO is 1.65 with molecular weights of 46.02 g/mole and 92.016 g/mole respectively. From this one can determine that 45.24 % Mfuel is MMH and this yields: M (MMH) = 410 kg and M (NTO) = 495 kg

The masses of other components are as such:

Engines = 20 kg each   Valves = 40 kg   Mountings = 70 kg

The total mass of the Propulsion Subsystem is 1496 kg

The burn time for the de-orbit maneuver is 150 seconds

Thrust/Mass = acceleration, Accel. x time = ΔV

$13360/13000 = 1.0277 \text{m/s}^2$ $(152.4 \text{ m/s})/ 1.0277 \text{ m/s}^2 = 150$ seconds

burn time is 300 seconds with only two thrusters

APPENDIX 3-B

The power subsystem was sized according to the following calculations

The total power of each subsystem for 30 hours on-orbit is 1325 W with an additional 20W of power for seven hours after landing. This totals to 39890 WHr.

At 3.0 Volts / cell and 10 cells / string, a total of 30 V a string is delivered.

39890 WHr/ 30V = 1330 AHr capacity required.

1330 AHr/ 27 AHr per string (2.7 AHr per cell) = 51 strings

1.2 x 51 strings (for redundancy) = 62 strings

1.3 x 62 strings (to compensate for time capacity degradation) = 80 strings

At 10 strings per battery pack a total of 8 battery packs are required to power LiBERTy

Each battery pack provides 270 AHr so if additional power is needed more battery packs can be placed in future missions.

Mass of each cell is 0.2268 kg and at 800 cells the mass = 181.405 kg

Adding 15% cell weight for structure and 5% cell weight for electronic circuitry yields:

Mass of Power System = 218 kg or 27.25 kg per battery pack

Density of cell and structure = 2242.2 kg/m^3

Therefore the volume of each cell = (27.25)/(2242.2) = 0.0122 m^3

Total Mass including PCU's and Load distribution assembly = 417 kg

REFERENCES

3.1 Sutton, G.P., Rocket Propulsion Elements, New York, N.Y., J. Wiley and Sons, Inc., 1986.

3.2 Yenne, Bill, Encyclopedia of U.S. Spacecraft, New York, N.Y., Exeter Books, 1985.

3.3 JCS 23264, "ACRC-CERV Option Reference Concept," Job Order 052-PA-301, Johnson Space Center, 15 November 1988.

3.4 Martin Marietta Corporation, System Definition Handbook- Space Shuttle External Tank (Lightweight Model) Configuration and Operation, Vol. 1, New Orleans, LA, 1980.

3.5 Trout, J. Barry, "Studies of Performance and Abuse of Lithium BromineComplex Cells for Manned Space Use", Johnson Space Center (JCS 20006), 1988.

ATTITUDE AND ARTICULATION CONTROL SUBSYSTEM

Dan Quitno

INTRODUCTION

Attitude and articulation control is important in any spacecraft's mission. It provides the ability to maneuver the spacecraft to acquire sensing information, dock, prepare for reentry and fulfill similar mission requirements. It compensates for disturbances of the spacecraft's orientation in space. And it is closely integrated with all other subsystems in designing a Logistics Resupply Module/Crew Emergency Return Vehicle.

REQUIREMENTS

In addition to the general system requirements outlined in the Request for Proposal for a LRM/CERV, such as (1) use artificial intelligence wherever possible to reduce costs and (2) provide a vehicle lifetime of six years (refer to Introduction Section for a complete listing of general requirements), the attitude and articulation control system (AACS) must also fulfill its own specific subsystem requirements. These are (1) provide attitude control; (2) provide pointing control; (3) maneuver for docking and rendezvous; and, (4) load and unload payload. In order to provide attitude control, AACS must be able to provide changes in the orientation of the spacecraft. For example, if LiBERTy is subjected to disturbances such as micrometeorites, magnetic effects, internal disturbances, or aerodynamic pressure, it must pitch, roll, or yaw to provide the proper attitude necessary to overcome these. Furthermore, if one of LiBERTy's sensors is not able to acquire one of its targets, it must change its orientation so it can acquire the target. LiBERTy must provide pointing control. In particular, it must provide scanning capabilities for the antenna. The maneuvers for docking are an extremely important facet of LiBERTy's mission because the maneuvers must be so precise. AACS must be able to continuously determine the range from the Space Station and effectively maneuver into the docking ports. Even small errors will result in failure to dock properly. Finally, the last requirement of this subsystem is loading and unloading of the payload.

This requirement will not be addressed by a "typical" attitude control system, however, for reasons to be discussed later; see page 4-5.

MISSION PROFILE/MODAL OPERATION

The mission profile of Logistics Resupply Module/Crew Emergency Return Vehicle includes three modes: (1) launch, (2) acquisition, which entails maneuvering the vehicle into the proper orbit and orientation required by the mission and testing the control equipment, and (3) specific mission operations such as docking with the Space Station and unloading payload. AACS is not integrated with the launch mode; any final orbit changes will be conducted by the main propulsion system. Refer to the Propulsion subsystem, Section 3, for more detailed information on this role. AACS is initiated once the acquisition phase begins. Maneuvers conducted by AACS during the acquisition phase include controlling LiBERTy's attitude and pointing direction. This is especially important because at this stage, the sensors must acquire their targets and the antennas must be in communication with the ground station. The hardware must also be tested during this phase of the mission in order to prevent any mishaps from occurring during a more crucial portion of the mission (as with docking). The final mode of the mission profile which is regulated by AACS is normal mission operations. For LiBERTy, this includes routine attitude corrections, docking with the Space Station, and preparing for reentry.

SUBSYSTEM INTERACTION

Since no single subsystem of a complex spacecraft design such as a LRM/CERV can operate independently of all the other subsystems, it is important to understand interaction between individual subsystems. The impact on AACS by the other subsystems is detailed below.

Mission Management, Planning, and Costing: This subsystem requires AACS to provide data on hardware used, particularly masses of each component. In return, MMPC must provide total mass of the vehicle in order for AACS to compute turn rates and thrust required for maneuvers, etc.

Structure: This subsystem requires AACS to detail attitude components used, including mass, dimensions or volume, and locations of the hardware. STRC must give AACS information on the shape and design of the vehicle.

Power and Propulsion: This subsystem requires AACS to detail power requirements for each component and both type and mass of fuel used in thrusters. In return, PPS will provide the power and fuel necessary to carry out the mission.

Command and Data Control: Interaction with this subsystem is especially important because all sensor information is funnelled through CDC, then the required adjustments are relayed back to AACS through the computers; the most crucial aspect of this is during docking using the Laser Docking System (LDS). In addition, AACS must also provide scanning capabilities for the CDC antenna; however, these antenna scanning operations are more thoroughly outlined in the CDC subsystem, Section 5.

Life Support: The interaction here is limited to requiring AACS to take precautions in the types of components used (e.g., no harmful magnetic effects) and in maneuvers conducted so as not to endanger the crew.

Reentry and Recovery: RRS requires AACS to orient LiBERTy at a -1.5° flight path angle to penetrate the atmosphere.

SELECTION OF COMPONENTS

Selecting components for an attitude and articulation control system was influenced by a number of factors. Torque required to yaw the vehicle 180° or to make simple attitude corrections to overcome micrometeorite or magnetic disturbances, accuracy for a spinning or non-spinning vehicles and for docking with another spacecraft, and scanning capabilities for antennas are just a few of the factors involved in choosing an attitude control system.

Attitude and Pointing Control: Components chosen for stabilization on LiBERTy depended on many specific factors. These factors most heavily considered included mass, accuracy, and redundancy requirements. Since the Request for Proposal required the vehicle to make emergency crew returns, spin and dual-spin stabilizations were immediately ruled out. A spinning spacecraft with crew members inside was inappropriate. Using a magnetic system was also deemed inappropriate because harmful effects on the crew members were anticipated. Control moment gyros were considered early in the design process, however, based on information on CMGs used on other spacecraft, the size of the CMGs necessary to conduct the maneuvers required for LiBERTy was determined to be too large and heavy. (See Reference 4-1.) Furthermore, a supplemental system would have been required, which was decided to be impractical. (Reference 4-1). Thus, a comparison study between the only seemingly sensible actuators, gas thrusters and momentum bias systems, was made. Refer to Table 4-1, page 4-8. After conducting these studies, gas thrusters were chosen because it will be more reliable and more effective to use only one system; reaction wheels require a supplemental gas thruster system to unload the momentum as well as a fault protection system.

In addition to studies made between thrusters and reaction wheels, comparisons were also conducted to determine the propellant used in the thrusters. Liquid bipropellants and cold gaseous propellants were considered. Liquid bipropellants such as monomethylhydrazine with nitrogen tetroxide have high specific impulses. Although useful when large torques are required to orient the spacecraft, they were ruled out because of expected plume impingement when docking with the Space Station. (See Reference 4-2.) Furthermore, there would be a problem of returning to earth with a hazardous propellant on board. Cold gases, on the other hand, do not ordinarily have as large specific impulses nor can they provide thrusts quite as large as liquid bipropellants. The greatest advantages acquired from a cold gas system are lack of plume impingement on the Space Station and the minimal propellant hazard involved in returning to earth with cold gas reserves still on board. Originally, a liquid bipropellant was chosen because of its high Isp and because of the advantage of storing the propellant reserves in the main propulsion fuel tanks. Later, however, it was decided there would be much plume impingement on the Space Station with a liquid bipropellant system but not with a cold gas system. (Reference 4-2.) Therefore a cold gas thruster system was selected. After conducting a trade study comparing specific impulse, density (at 0 ° C and 24.13 megapascals) and molecular mass, nitrogen was chosen as the cold gas propellant. See Figure 4-3, page 4-10 for results. (Reference 4-3.) Locations of the thrusters can be seen in the Structures subsystem, Section 2.

Sensors: Factors included in selecting sensors were accuracy, redundancy requirements, and updating requirements. Celestial sensors rely on external references while inertial sensors do not. However, inertial sensors frequently need to be updated by the absolute, external references. Gyroscopes, for instance, are accurate but often drift and therefore need corrections from an external source. Thus, it was decided not to use gyroscopes. Sun sensors will often have periodic dead time in which the sun is blocked from view. Star scanning sensors do not have that inadequacy. Most importantly, one of the requirements of the Request for Proposal was to dock with the Space Station. Extreme accuracy was then deemed essential for LiBERTy. A comparison study of sensors vs. accuracy was then conducted. See Figure 4-1, page 4-8, for results. After analyzing the comparison study, gravity gradient, magnetic field, and horizon sensors were immediately eliminated. Finally sun sensors were selected because of their accuracy and star scanners were chosen because of both their accuracy and their ability to complement the sun sensors. Use of two sun sensors was deemed necessary in order to improve the field of view and for redundancy. That was not required for the star scanner. Locations of the sun and star sensors are shown in Section 2.

Automated Docking: The most feasible method of docking was to utilize systems already compatible with the Space Station. A laser docking system with rangefinder and optoelectronic sensors were then chosen. (See References 4-4, 4-5.) All precise movements to dock with FREEDOM will be conducted by AACS.

Payload Loading/Unloading: Several options were considered to load and unload payload. The first option was to have an articulated appendage pick up the payloads and move them. This would require a special appendage with its own separate control system, which would only add mass and take up volume in LiBERTy. This system was therefore disregarded. Using a conveyor belt also seemed reasonable at first, but again the added mass plus generator to drive the system, estimated at 50 kilograms, were determined to be unnecessary. The control system finally selected for LiBERTy was consequently manual labor. This only requires the movers to secure themselves when transporting payloads; in a zero-g environment, equal and opposite reactions could easily propel the mover away from their intended direction. Securing movers in LiBERTy is not expected to cause problems.

SIZING OF COMPONENTS

Thrusters: After selecting the main components for attitude and articulation control, the proper sizing had to be determined. Sizing for thrusters was directly dependent on thrust required to yaw the vehicle 180°, which was assumed to be the maneuver requiring the maximum amount of thrust, as well as possible locations for the thrusters. The longer the moment arm for the thrust, the less thrust needed. However, the ideal length desired for the system had to be sacrificed because the thrusters could not be situated on the very ends of the vehicle; interference with main engines created one problem while the conical section of the vehicle created the other. See Figure 4-2 for a trade study between thrust and moment arm. In this study, exact values of mass moment inertia were not necessary, only general ones to give an effective estimate. Therefore, the mass moment of inertia about the translational axes of a generic cylinder of approximately the length and diameter of LiBERTy was calculated. Four different angular accelerations were held constant for varying moment arms to graph the force required. See Appendix 4A for equations. The best system under these conditions was thrusters with 90 Newtons force each and a moment arm of 3.5 meters. It is especially important for the thrusters to provide minimal thrusts in the one Newton range (for precise docking procedures) as well as ones as large as 90 Newtons. It was decided to use four packs with four thrusters per pack to give complete rotation around each axis. Two packs were placed on opposite ends of the vehicle. The packs at each end were then placed 180° apart, but not on the underside of LiBERTy because there they would be susceptible to heat effects during reentry. Again, see actual layout in the Structures subsystem, Section 2. The dimensions of the thrusters were then chosen by downsizing current and proven RCS engines from the space shuttle. Refer to Table 4-2 for these dimensions.

Propellant: The propellant selected for AACS thrusters was nitrogen gas. Fuel requirements for LiBERTy were calculated by estimating the number of 180° yaws and 180° rolls needed for one full mission. An extra 20% of propellant was added to compensate for minor attitude corrections made to overcome disturbances to the vehicle. It was assumed each vehicle could refuel on earth before the next launch. Maintaining propellant reserves for more than one mission was infeasible because it would increase the mass required several times. See Appendix 4B for equations. The mass of all fuel lines and valves were estimated based on a similar system in the SCRAM. (Reference 4-2.)

Sensors: Both the sun sensors and the star scanner were chosen with off the shelf philosophies. Masses, power required, and dimensions were thus estimated based on current components. (See Reference 4-1.) Refer to Table 4-2 for these specifications.

Automated Docking: Sizing the rangefinder and optoelectronic equipment was difficult. Assumptions were made based on material in References 4-4, 4-5, in order to determine their specifications. See the CDC subsystem for more information on the docking sequence involving AACS components.

PROBLEM AREAS

The problems which arose in designing the attitude and articulation control system were based primarily on the thrusters and docking. The mass of the fuel lines and valves was estimated by scaling down current systems. Although mass expulsion was selected because of its massive torque capabilities, the concern arose over docking. After a thorough literature search, it was finally assumed that the 90 Newton thrusters on LiBERTy could indeed provide the minimal thrusts required for precise docking procedures. In addition, the problem arose in determining thruster locations. Since such large torques were required to yaw LiBERTy 180°, the thrusters needed to be placed at the extreme ends of the vehicle. However, the main engines would have interfered at one end, while the conical shape would have actually served as a destabilizer at the other end. Another problem existed in sizing the LDS; again, assumptions were made based on reference materials to scale the rangefinder and optoelectronic equipment.

TABLE 4-1. Actuator Comparison

Control Actuator	Accuracy	Maneuverability	Comments
Mass expulsion	Good	Good	- Lifetime limited by fuel requirements - Excellent torque capability
Reaction Wheels (3)	Very accurate	Excellent	- Expensive - Need extensive fault protection - Requires momentum unloading
Pitch Momentum Bias	Limited	Poor	- Lower cost - Longer life - Subject to Nutation

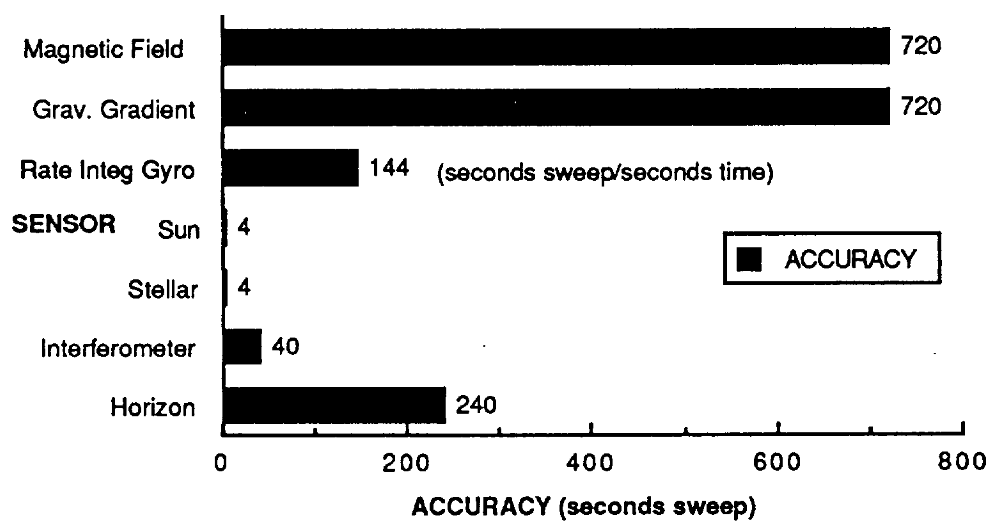


Figure 4-1. Sensor vs. Accuracy. See Ref. 4-6.

TABLE 4-2. AACS Components

Component	Mass (kg)	Power Required (Watts)	Dimensions (m) or Volume (m^3)
Computer	10	40	0.2
Star Sensor	2	2.5	0.3 x 0.3 x 0.4
Sun Sensor (2)	1 (each)	4 (total)	0.08 x .0.09 x 0.025
Rangefinder	4	10	0.15
Optoelectronic Sensors (2)	0.5 (each)	2	0.1 square
Gas Thrusters (16)	24 (total)	8 (per pack of four)	0.6 x 0.6 x 0.2 (per pack)
Fuel, Tanks (4), Lines, Valves	113 7	Not Applicable	Nitrogen : 0.11 (per tank) Pressurant: 0.035(per tank)

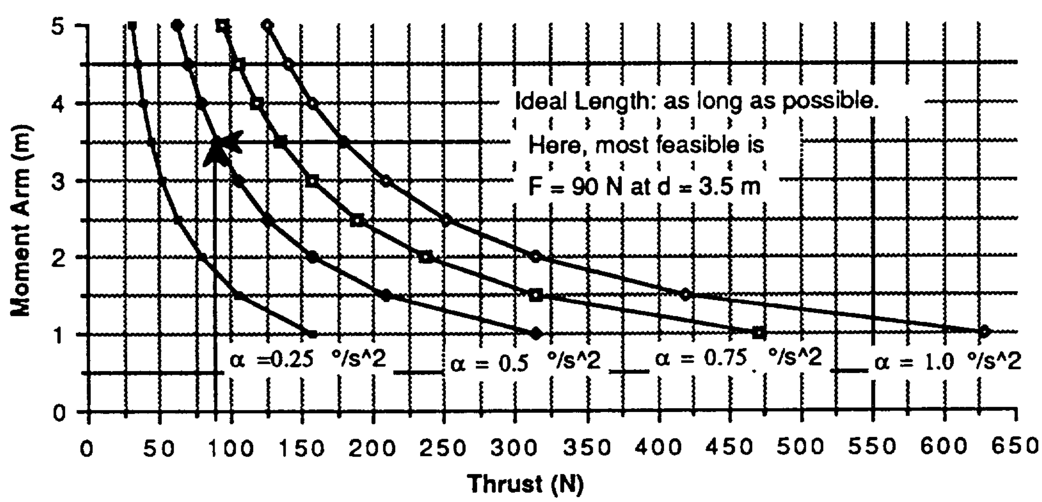


Figure 4-2. Moment Arm vs. Thrust

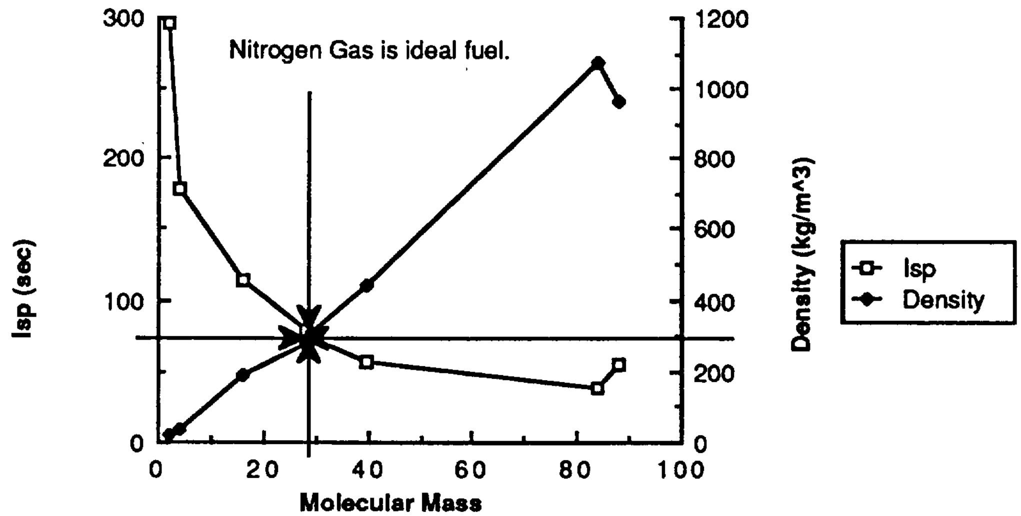


Figure 4.3. Isp/Density vs. Molecular Mass. Reference 4-3.

APPENDIX 4A

THRUSTER SIZING EQUATIONS:

$M = F^*d = J^*\alpha$

$F = \frac{J\alpha}{d}$

M	=	Moment, N-m
F	=	Thrust Force Per Thruster, N
d	=	Moment Arm, m
J	=	Moment of Inertia of Cylinder, kg-m^2
	=	$\frac{1}{12}m^*(3r^2 + L^2)$ about Pitch and Yaw Axes
	=	$\frac{1}{2}m^*r^2$ about Roll Axis
		m = mass of cylinder = 13,000 kg
		r = radius of cylinder = 2 m
		L = length of cylinder = 11 m
α	=	Angular Acceleration, rad/sec^2, varied
	=	$\frac{d\omega}{dt}$, ω = angular velocity, rad/ sec

Note that α was held constant at $0.25°/s^2$, $0.5°/s^2$, $0.75°/s^2$, and $1.0°/s^2$. Values of d varied from 1 m to 5 m. Thrust was then calculated.

APPENDIX 4B

DELTA-V REQUIREMENTS

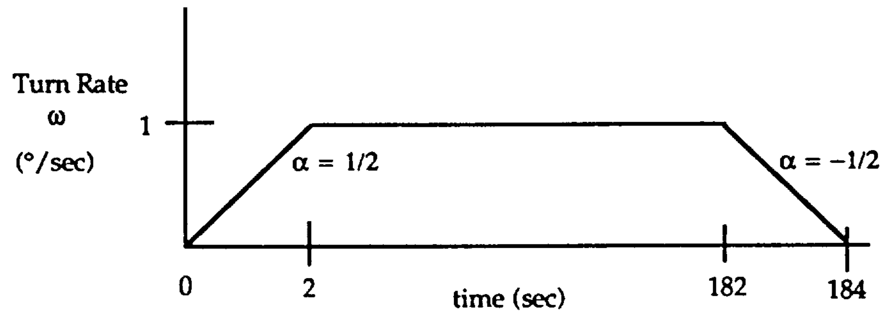


FIGURE 4B-1. Acceleration Profile: Turn Rate vs. Time

$$\Delta v = \int_0^2 \alpha dt + \int_2^{182} \alpha dt + \int_{182}^{184} -\alpha dt$$

$\Delta v = g^* Isp^* \ln(Mo/Mf)$

$M_{fuel} = M_o - M_f$

v = Velocity, m/sec
 = $r\omega$ for Rolls
 = $\frac{L}{2}(\omega)$ for Yaws and Pitches
g = Acceleration of Gravity
 = 9.81 kg^*m/sec^2
I = Specific Impulse of Nitrogen
 = 80 sec
M_o = Initial Mass, kg
M_f = Final Mass
M_{fuel} = Mass of Fuel Required for Δv

Sum M_{fuel} five 180° yaws/pitches and twenty 180° rolls during one mission for worst case estimate. Add additional 20% for corrections to disturbance torques.

TANK SIZING

Find radius of four spherical tanks.

$M_{fuel} = \rho\,(\frac{4}{3}\pi r^3)$

ρ = Density of Nitrogen
 = 278.24 kg/m^3
r = Spherical Tank Radius

REFERENCES

4-1. Wertz, James R., ed. *Spacecraft Attitude Determination and Control*, D. Reidel Publishing Co., Boston, 1978.

4-2. *Conceptual Design Study of a Crew Emergency Return Vehicle* (JSC-23103). Advanced Program Office, JSC, NASA, August,1988.

4-3. Sutton, George P. *Rocket Propulsion Elements*, John Wiley & Sons, New York, 1986.

4-4. Lembeck, Michael F. University of Illinois AAE 241 Aerospace Vehicle Design, Notes # 7, Notes #9, February, 1989.

4-5. Erwin, Harry O. "Laser Docking System," *Advances In The Aeronautics Sciences, Guidance and Control 1984*, American Astronautical Society, San Diego, 1984.

4-6. Chobotov, V.A. Northrop University, *Spacecraft Dynamics and Control*. For the Course AAE 391 E, University of Illinois, Spring 1989.

4-7. Kaplan, Marshall H. *Modern Spacecraft Dynamics & Control*, John Wiley & Sons, Inc., New York, 1976.

4-8. Sorenson, Albert A. and I.J. Williams. "Spacecraft Attitude Control," *Quest*, TRW Electronics & Defense, Summer 1982, pp. 23-48.

COMMAND AND DATA CONTROL SUBSYSTEM

DAVID J. QUINN

INTRODUCTION

The Command and Data Control Subsystem (CDC) will provide LiBERTy with the necessary communication link to the ground and space station. The design task of the CDC analyst is to provide an efficient means of communication and command control to be used during every phase of LiBERTy's mission. The status and condition of this spacecraft are determined by telemetry. Temperatures, voltages, switch status, pressures, sensor data, and many other measurements are transformed into voltages, encoded into pulses, and transmitted. The CDC subsystem is designed to satisfy requirements and guidelines outlined in the Request For Proposal (RFP).

CDC REQUIREMENTS

Collect telemetry from all subsystems.

Send telemetry to ground control and space station.

Receive commands from ground control and space station.

Send commands to subsystems.

Provide an automated rendezvous and docking procedure.

Allow a communication link for the crew.

CDC DESIGN CONSIDERATIONS

Technology. The CDC subsystem adheres to the request that all equipment used must be available before 1995, and where possible, off-the-shelf technology be employed. The lifetime of the subsystem components must exceed six years. These requirements are satisfied because the technology of low-orbit communication components has not changed significantly in recent years and does not require new scientific break-throughs. Therefore, proven, off-the-shelf CDC components can be used. The performance and wear-resistant capabilities of these components are already known. The technology involved for the automated docking procedure has recently been developed for other spacecraft and will be fully tested prior to use on LiBERTy. The use of existing technology stresses RFP guidelines such as reliability, simplicity and low cost.

Automatic vs Manual Control. The use of the latest advances in artificial intelligence was encouraged in the RFP. The decision not to automate the command and control process involved a comparison analysis (see Table 5-1). The options included control by ground and and space station personnel, control by expert systems and automation, or in some instances, control by the crew.

Automation. An automated system would provide quicker-than-human responses, optimum control, and increased efficiency in the long run. Due to a void in technology, the computers needed to handle the vast amount of data demanded by expert systems would be massive and complicated. Large, relatively archaic computer systems are used in space missions to resist the harmful effects of cosmic radiation, because the more efficient, smaller processors are highly susceptible to interference from cosmic rays. When these systems are used, they require significant redundancy and additional components to check signals for correctness. Typically, a voting procedure is used between three separate binary signals. If two or three match, whether correct or not, that command is processed. New technology is being developed to provide low-weight, efficient components to be used in space applications which can withstand cosmic noise, but such devices are not anticipated until after 1994. The use of current automatic systems would add mass and complexity to the design. To maintain a spacecraft system with a lengthy mission, such as the space station, automation is used to limit the need of an excess amount of ground support personnel; for LiBERTy's short mission, automation is not necessary for this purpose. Therefore, the rendezvous and docking procedure is the only example of significant autonomy provided on LiBERTy.

Ground and Space Station Control. Remote control from the ground and space station is the simplest method to develop. A manual means of control should exist for the possible over-ride of an autonomous maneuver, so a savings in weight, and reduced complexity exists if the system is designed for manual control only. LiBERTy's only maneuver requiring extreme precesion is the rendezvous and docking procedure. That operation should be automatic, but there is no need for the entire mission to be automated. A primary reason to control LiBERTy manually is that there is no significant cause to provide automatic control.

Crew Control. For the logistics return mission, external control of LiBERTy is necessary, but crew control could apply in the event of emergency crew return. To allow for commands from the crew, many additional components, such as input/output devices and controls, would have to be developed and placed in the vehicle in addition to the components necessary for remote control operations when the crew is not present. This would add significant cost and weight to the system. The primary factor ruling out crew control as an exclusive means to command the return is the anticipated poor health of the crew members returning. For the design of an emergency return vehicle, it is necessary to assume the passengers are seriously sick or injured. Their condition may not permit them, or others taking care of them, to perform the complex control operations. Even if the crew members are in perfect health, the control procedures would have to be drilled and studied. Since Space Station crews are not comprised of pilots, these manuvers may present a problem, and in the disaster scenario the crew members may be very shaken. However, there are a few initial commands the crew will be required to perform upon entering the return vehicle. These commands, which initiate the escape sequence, cannot be performed by another source; the space station is assumed to have serious malfunctions, and the ground station may be unaware of the disasterous situation or their signals may be inaccurate or too late. Therefore, the only commands given by the returning crew trigger the release from the space station. The ground station will assume control after this manuever and maintain a voice link with the crew.

Table 5-1 . Comparison of control options.

Control Options	Advantages	Disadvantages
automation	quick, efficient	complicated, heavy
ground/space station	simple, proven	time delay, human errors
return crew	effective timing	added equipment poor health

TDRSS. LiBERTy will communicate with the ground support station using the Tracking and Data Relay Satellite System (TDRSS). A principle advantage of the system is the elimination of many of the worldwide ground stations for tracking low-orbiting spacecraft. TDRSS consists of two geostationary relay satellites and a ground terminal located at White Sands, New Mexico. Due to the greater distance from Freedom's orbit and the relatively small size (3.8 m) of its S-band dish, the use of TDRSS requires a more powerful communication subsystem than the ground network requires, but TDRSS will provide communication line-of-sight time to approximately 90% of the space station's orbit as shown in Figure 5-1.

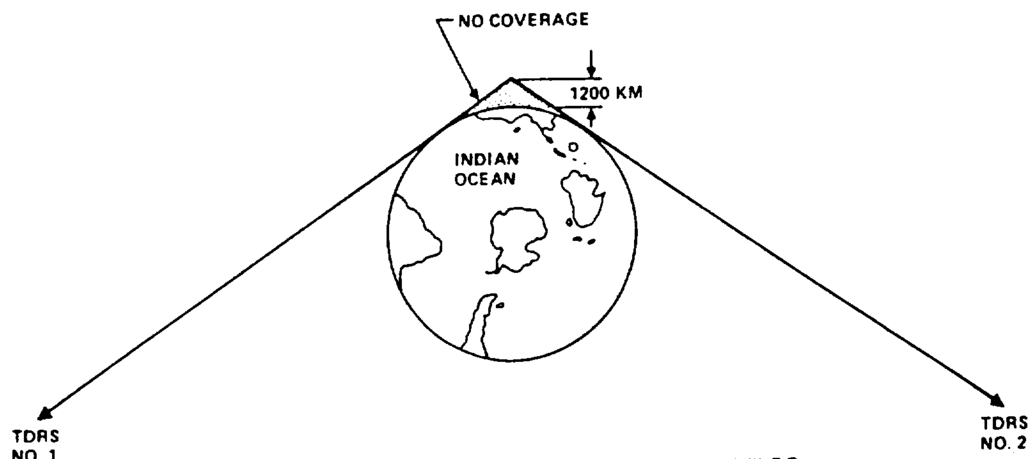


Figure 5-1. Line-of-sight provided by TDRSS.

This significant advantage over the ground-station network, which is on the order of 15 to 20% coverage, may be vitally important in the case of an emergency evacuation occuring around the globe. An S-band transmitter/receiver system will be used to relay data through the tracking satellite using the Multiple Access (MA) service provided by TDRSS. The high data-rates available using the Single Access (SA) service are not necessary for LiBERTy's mission.

TDRSS-link Antenna. To size the components of the CDC subsystem, comparisons were made with parameters from the Space Shuttle. The Shuttle uses a high-power (140W) S-band transmitter to send a data-rate of 128 kb/s. Utilizing Equation 5-1 in Appendix 5A, the estimated power received at TDRS from the shuttle's S-band transmitter is on the order of 1×10^{-11} W. From Shannon's Law, Equation 5-2 in Appendix 5A, it is evident that a reduced received power value results in a reduction in transmitted data-rate. The maximum data-rate permissible using the MA TDRSS service is 50 kb/s. Using estimates for unknown parameters and Equation 5-2, a value of $3x10^{-12}$ W of power received at TDRSS corresponds to a data-rate of 50 kb/s. Using Equation 5-1 to generate Figure 5-2, an approximate S-band antenna diameter and transmission power can be chosen keeping in mind that as the size of an antenna increases, so do its cost and structual problems.

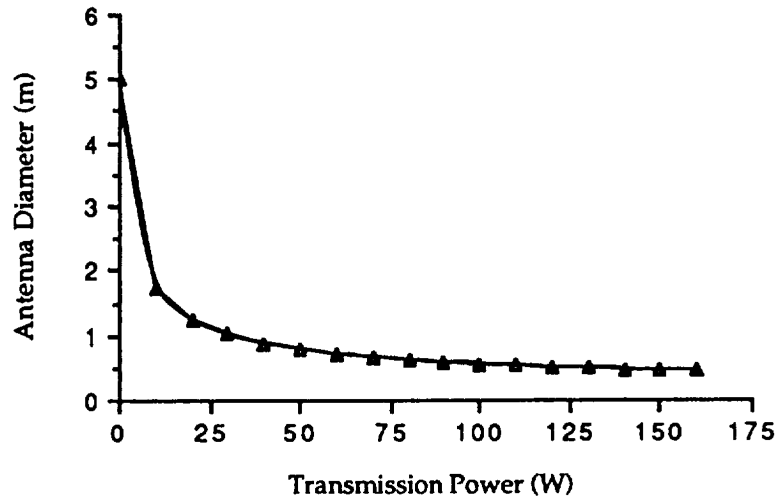


Figure 5-2. Size of TDRSS-link antenna vs. transmission power.

The plot shows that an antenna diameter of 0.6 m is capable of sending a transmission power of 80 W to TDRSS. This antenna will fit well housed in the rear-end of the spacecraft and insulated from extreme temperatures (see Figure 2-8). A rear hatch is a convenient location for the antenna, for it would obstruct the docking procedure if placed on the top of LiBERTy and disturb the protective shielding if housed near the front or bottom. Once in orbit, the hatch will automatically open and a telescoping, hinged arm will expose the antenna. The antenna will slowly rotate until an adequate signal can be obtained from the ground station. Active pointing using a set of actuators located on the arm, will allow the ground station to achieve optimum signal reception through TDRSS. This antenna will be active throughout the entire mission until it is retracted before deorbit. Figure 5-3 maps the communication and tracking links.

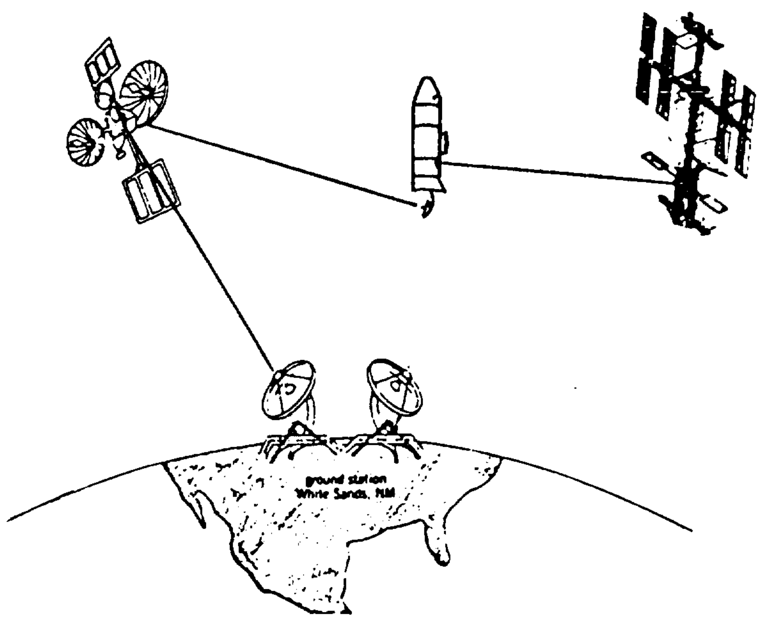


Figure 5-3. Communication and tracking network.

Freedom-link Antenna. The RFP requires that all vehicle components operate under positive space station control in the vicinity of the station. This requirement initiates the need for a separate command link. An S-band system was chosen for this application to remain compatible with the TDRSS S-band link components. Based upon Soviet rendezvous data, the space station may acheive sensor acquision or lock-on at a distance of 35 km. A low-gain S-band antenna with a diameter of 0.1 m and power output of 2 W will be adequate for this close range connection. The antenna is flush-mounted to the top of the spacecraft behind the berthing adapter. The flush antenna is overlaid with thermal protective material to provide the ability to survive the heat of entry. The location is chosen to favor the direction of the space station preceeding rendezvous, and to further reduce the heat seen upon reentry (see Figure 5-3). The space station assumes control of LiBERTy at a distance no greater than 35 km. It commands the vehicle until the automated rendezvous procedure.

Rendezvous and Docking. Automatic rendezvous and docking capability saves fuel and crew time and improves the safety of these maneuvers. The rendezvous procedure is initiated by stationkeeping, which is the maintenance of constant relative position, and attitude with respect

LiBERTy are capable of monitoring the position and velocity of the space station. The LiBERTy's range finder devices will be mounted directly to the berthing adapter. Commands for the operations will be generated by an expert computer system located in the nose of LiBERTy. The space station will have the capability to over-ride this system in case of a failure or malfunction. The docking maneuver requires that the station's range, bearing, attitude and rates be known. This begins at about 100 m out. The space station serves as a cooperative target and carries retro-reflectors placed in a known location and orientation. These reflectors are tracked by means of a modulated laser beam located on the top of LiBERTy. This Laser Docking System (LDS) determines the necessary data to achieve desired physical contact. The precision of measurements required for automatic dockings is possible only with laser type systems. All the vehicles required to dock with the space station should use similar systems to enhance simplicity and drive the cost of development down.

Return. While in orbit, the crew will be able to talk to the ground station and/or the space station through the S-band systems. Either station may control the deorbit operation. Upon return, CDC will utilize a Search and Rescue Satellite (SARSAT) Beacon to aid in the location of the vehicle. The S-band link cannot be maintained after deorbit, so a low-power UHF system will be used to provide communication with the SAR team. Communication with the SAR team only requires a transceiver/antenna combination which can be a simple, light-weight system. Two transceivers are supplied for redundancy; there is little weight and no power penalty in providing the additional unit. A hand-held survival radio is provided in case the crew has to abandon the vehicle.

CDC COMPONENTS

The CDC components include the S-band antennas, rendezvous and docking components, and UHF devices mentioned above, plus a series of special-purpose processors to interface the subsystems and crew with the transmission and reception devices. Table 5-2 containing the mass and power of CDC components is found in Appendix 5B, a schematic layout of the main communication network is shown in Figure 5-5.

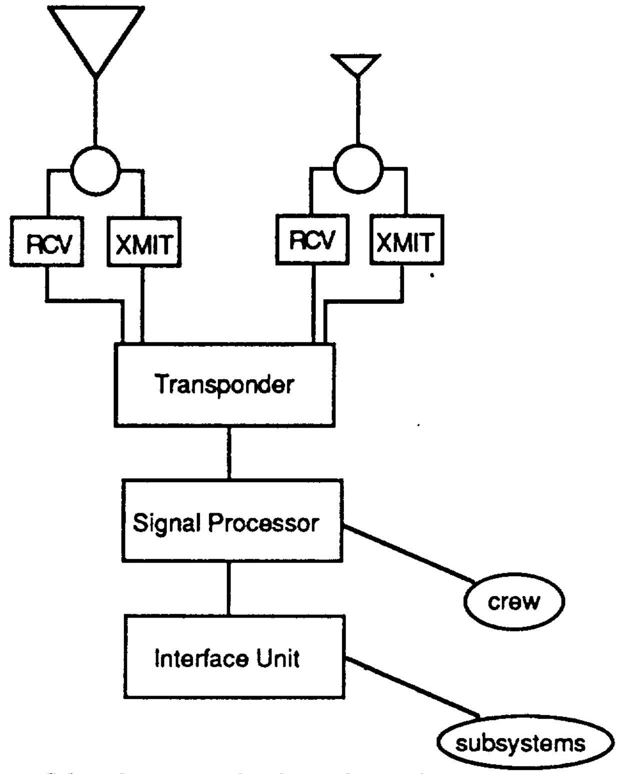


Figure 5-5. S-band communication schematic

Interface Unit. The inferface unit accepts data from other subsystems, converts it into transmittable form, and provides output telemetry to the signal processor to be sent through the radio system. Incoming commands are routed through the interface unit to the appropriate subsystem.

Signal Processor. The signal processor accepts telemetry data from the interface unit and analog voice data from the voice system. The voice is digitized and multiplexed with the operational telemetry and outputs are provided to the transponder, which serves as an S-band transmitter/receiver. The command and voice data received by CDC is accepted by the signal processor and demultiplexed. The voice data is converted to analog form and routed to the crew; the command data is routed to the interface unit for proper distribution.

Transponder. The function of the transponder is to provide a means of measuring range and range rate of LiBERTy by the control stations. This information is used to compute its orbit and trajectory. Range rate is measured from a comparison of the transmitted and received signals at the control station. LiBERTy will employ a NASA compatible transponder having a transmit to receive frequency ratio of 240:221 and phase-modulated. A Doppler shift measurement translates into range rate. The time delay between transmission and reception provides a means of computing range. The transponder also performs the functions of a command receiver and a telemetry transmitter.

PROBLEM AREAS & CONCLUSION

There were few problem areas or "show stoppers" encountered in the design of the CDC subsystem. The various CDC antennas could be better defined if the antenna gains of the space station and TDRS antennas was known. Certain information, needed to perform calculations, was unavailable, but educated estimates based upon current systems supplied necessary parameters to obtain conceptual design values. These numbers are located in Table 5-3 in Appendix 5A. This proposal for the Command and Data Control Subsystem adheres to the requirements and guidelines of the RFP for the conceptual design of a combination Logistics Resupply Module (LRM) and Crew Emengency Rescue Vehicle (CERV).

APPENDIX 5A

Equations (Reference 5–3)

Equation 5-1: $dt = (\frac{P_r}{P_t})^{.5} [(4cD)/(fzd_r\pi)]$

Equation 5-2: $B = W\log_2[P_r/P_n + 1]$

Table5-3. TDRSS-Link Design Parameters

Variable	Value	Description
D	40000 km	approximate maximum distance
d_r	3.8 m	TDRS S-band antenna diameter
P_r	$2x10^{-12}$ W	power received at TDRS
z	0.55	assumed efficiency
c	$3x10^8$ m/s	speed of light
f	2.3 GHz	approximate S-band frequency
P_t	80 W	power transmitted from LiBERTy
d_t	0.6 m	LiBERTy S-band antenna diameter

Table 5-2. Mass and Power of CDC Components

COMPONENT	MASS (kg)	POWER (W)
TDRSS-link antenna	3	0
Freedom-link antenna	0.6	0
Transponder	10	100
Signal processor	9	50
Interface unit	7	25
Laser Docking System (LDS)	7	17
computer	6	15
UHF antenna (2)	7	0
UHF transceiver (2)	8	5
Portable radio	0.5	10
SARSAT Beacon antenna	3.6	0
Transmitter	03	5
TOTALS	**62**	**227**

REFERENCES

5–1. *ACRC-CERV Option Reference Concept* (JSC-23264); Johnson Space Center, National Aeronautics & Space Administration; 15 November 1988.

5–2. *Conceptual Design Study of a Crew Emergency Return Vehicle* (JSC-23103); Advanced Programs Office, JSC, NASA; August 1988.

5–3. Lembeck, Michael F. (Department of Aeronautical & Astronautical Engineering, University of Illinois);
•AAE 241 lectures; Spring Semester 1989;
•conversations; Spring Semester 1989.

5–4. "Space Shuttle News Reference"; NASA; 1978.

5–5. Avvento, Gennaro J. *Reusable Unmanned Automated Resupply Freighters For Space Staton Operations;* AIAA Technical Paper 87-0687; March 24-26 1987.

5–6. *Encyclopedia of Physical Science and Technology;* Academic Press Inc. Volume 13; 1987.

5–7. *McGraw-Hill Encyclopedia of Science and Technology;* McGraw-Hill Book Company; Volume 17; 1987.

5–8. Solomon, Wayne (Department of Aeronautical & Astronautical Engineering, University of Illinois); discussion; April 1989.

5–9 Erwin, Harry O. "Laser Docking System," *Advances in the Aeronautical Sciences. Guidance and Control 1984;* American Astronautical Society; San Diego; 1984.

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SUBSYSTEM

ERNST P. JANENSCH

INTRODUCTION

In addition to its role as a logistics bus, LiBERTy also serves as an emergency return transport. If a catastrophe disables Space Station *Freedom*, or if a crew member becomes ill or is injured, LiBERTy provides a safe escape route to Earth. The Environmental Control and Life Support Subsystem ensures the successful completion of this critical task.

REQUIREMENTS

INITIAL REQUIREMENTS

The Request for Proposals (RFP) includes the following requirements relating to the ECLSS:

- Maintain pressurized atmosphere.
- Support crew throughout mission.
- Use pre-1995 technology, preferably off-the-shelf.
- Design for reusability and 6-year lifetime.
- Maximize reliability and performance.
- Minimize complexity and weight.

DERIVED REQUIREMENTS

The following requirements were then derived for the given reasons:

- Space Station compatibility.
  For quick escape, LiBERTy must be docked to *Freedom* at all times and must be compatible with its shirt-sleeve environment of 101 kPa.
- Crew size: 2–8.
  If a crew member becomes ill or is injured, at least one other crew member must accompany him/her. The maximum Space Station crew will be eight.
- Food & water provisions, waste disposal.
  Orbit and ocean waiting times dictate consideration of physiological needs.
- Medical equipment.
  In the event of a Space Station disaster, some crew members may be injured and there would be no time to collect Station medical equipment.
- Fire detection and suppression.
  Fire hazards must be accounted for in any spacecraft design.

DESIGN OPTIONS, TRADES, AND DECISIONS

CREW SIZE

The first design parameter to be determined was LiBERTy's maximum crew size. One of the derived requirements called for a range of two to eight crew members. In the injury/illness scenario, two crew members would return to Earth, leaving six in the Station. If a catastrophe were then to occur, six persons would need a method of escape. Therefore, either one vehicle capable of supporting six, or a combination of vehicles collectively capable of supporting six, would be needed.

Options

For efficiency, the number of vehicles needed by the Station at any one time should be minimized. It is also desirable to use as few docking ports as possible. Furthermore, the Structures Analyst indicated that LiBERTy's size would be rather large. In cooperation with the Mission Manager, it was determined that each LiBERTy vehicle should be capable of carrying at least six crew members. This means that only two LiBERTy vehicles need to be docked to *Freedom* at a time.

A LiBERTy crew size of six or seven would mean that, in the event of a disaster, two vehicles would be used to return the Station crew of eight to Earth. Search-and-rescue (SAR) forces would be required to locate and recover two vehicles, dozens or hundreds of kilometers apart. A LiBERTy crew size of eight, on the other hand, would enable the use of a single vehicle. SAR forces would then be able to concentrate on one vehicle alone.

Trade

Although a crew size of eight is preferable from a SAR viewpoint, comparisons needed to be made to determine the effects of ECLSS mass on LiBERTy 's over-all design.

The first comparison was ECLSS mass as a fraction of total vehicle mass for both a six-person and an eight-person vehicle. Initially, total vehicle mass was unknown. But the Mission Manager indicated that it would have to be kept below 15,152 kg (after allowing a 20% margin) in order to be launched by a Titan IV launch vehicle. ECLSS mass was not initially known either, but the envisioned heaviest system (Appendix, Part A) and the longest reasonable mission length (32 hours) were used to ensure the worst (i.e. heaviest) ECLSS-to-vehicle mass fraction. These maximum

ECLSS masses are 4.4% and 5.0%, respectively, of the maximum possible vehicle mass (see Figure 6–1). With regard to mass, therefore, an eight-person system places little more burden on vehicle design than does a six-person system.

The other factor considered was crew volume needs compared to vehicle volume. The Structures Analyst estimated an internal volume of at least 35 m^3. For a 32-hour mission, crew volume needs were calculated (Appendix, Part B) and found to be 58.6% and 77.0%, respectively, of total available vehicle volume. As Figure 6–1 shows, the needs of both a six-person and an eight-person crew are well within the vehicle's internal size.

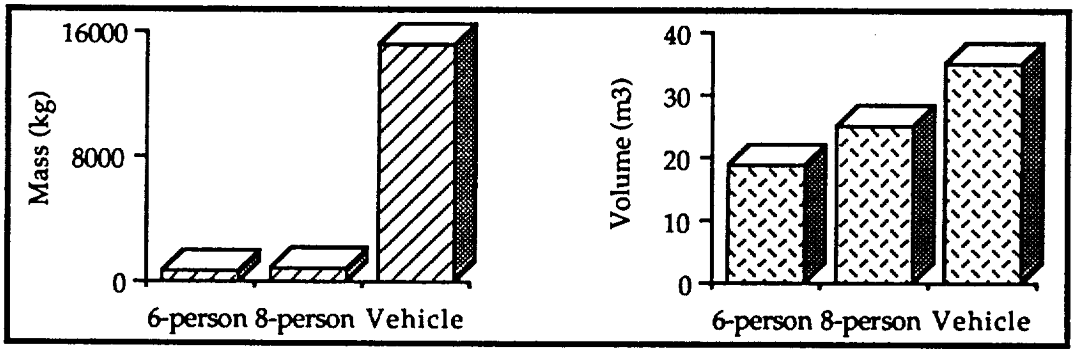


Figure 6–1: ECLSS vs. Vehicle Mass and Volume Comparisons

Decision

Therefore, since neither mass nor volume indicated that one size was more preferable than the other, an eight-person crew size was chosen to facilitate SAR.

ATMOSPHERE CONTROL

Options

There are three basic types of atmosphere control systems: regenerative, active, and passive.

Regenerative closed-loop atmosphere control acts together with its temperature control counterpart as part of an enclosed, self-sustaining ecosystem. Long-duration missions, like the Space Station, require such systems because it would be inefficient to continually resupply all consumables and jettison all waste products.

An active open-loop atmosphere control system, on the other hand, requires a steady supply of consumables. Waste products are not recycled, they are simply stored. Such a system has been used in most American spacecraft, including Apollo and the Shuttle. Fresh oxygen is supplied via a flow control orifice. Carbon dioxide (CO_2) is removed by lithium hydroxide (LiOH) canisters. A ventilation system ensures that oxygen is available throughout the vehicle for consumption and that CO_2 pockets do not form.

A passive atmosphere control system ignores cabin air and simply provides fresh air to the crew via helmets. Air flow can be manually controlled, and there are no moving parts.

Trade

A regenerative system was judged as much too complex for LiBERTy's needs. The only aspect of an active system which at all complicates matters is the need for a recirculation fan. To choose between an active and a passive system, comparative calculations were made (Appendix, Part C) of the mass of each system as a function of mission length (see Figure 6–2). Clearly, a passive system would weigh much more than an active system.

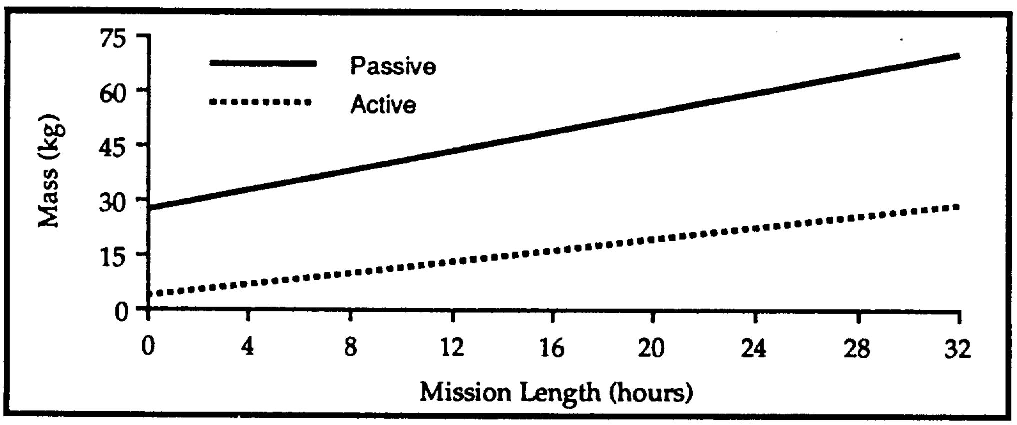


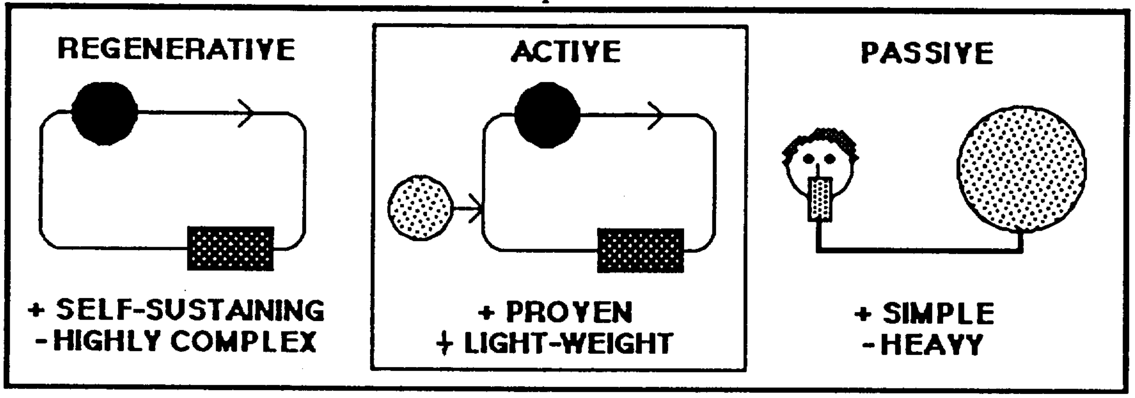
Figure 6–2: Atmosphere Control System Mass vs. Mission Length

Decision

Because an active atmosphere control system is the lightest and is only marginally more complex, it was selected for use in LiBERTy's ECLSS.

Table 6-1: Atmosphere Control Summary

Options



Trade

OPTIONS	REQUIREMENTS			
	REL +	COMP -	PERF +	WT -
REGENERATIVE	UNKNOWN	HIGH	HIGH	HIGH
ACTIVE	GOOD	LOW	GOOD	LOW
PASSIVE	GOOD	LOW	GOOD	HIGH

(REL=reliability, COMP=complexity, PERF=performance, WT=weight)

Decision

A REGENERATIVE SYSTEM WOULD BE TOO COMPLEX AND HEAVY. A PASSIVE SYSTEM WOULD ADD TOO MUCH WEIGHT TO THE VEHICLE. BECUASE OF ITS PROVEN RELIABILITY, ITS SIMPLICITY, AND ITS LOWER MASS BURDEN, AN ACTIVE ATMOSPHERE CONTROL SYSTEM WAS SELECTED FOR LiBERTy.

TEMPERATURE CONTROL

Options

The three basic types of temperature control systems are: regenerative, active, and passive.

A regenerative temperature control system works with its atmosphere control counterpart as a closed ecosystem. Little or no supplies are needed, and waste is recycled. A regenerative system is highly complex and is applicable to long-term missions.

An active open-loop system requires many complicated pieces of equipment (e.g. condenser, flash evaporator, heat exchanger) involving a liquid loop with piping, pumps, and air/water heat-exchanging interfaces. Active temperature control systems have been used on most US spacecraft, including Apollo and the Shuttle.

A passive system involves no complex hardware, consists of no moving parts, and requires no power. Avionics equipment is insulated from the crew cabin and the heat it generates is radiated to the space environment. Crew metabolic heat is absorbed by a phase-change material, such as wax. Heat energy is absorbed during the change of phase from solid to liquid, preventing the cabin temperature from rising. Although such a system has been studied, no spacecraft has yet employed it. This is because it is only appropriate either for short-duration missions or for use after long down-times, and recent spacecraft have not called for either.

<u>Trade</u>

A regenerative temperature control system was eliminated for the same reasons a regenerative atmosphere control system was. The choice between an active and a passive system was less obvious. Both the mass and the cost of each system would be comparable.

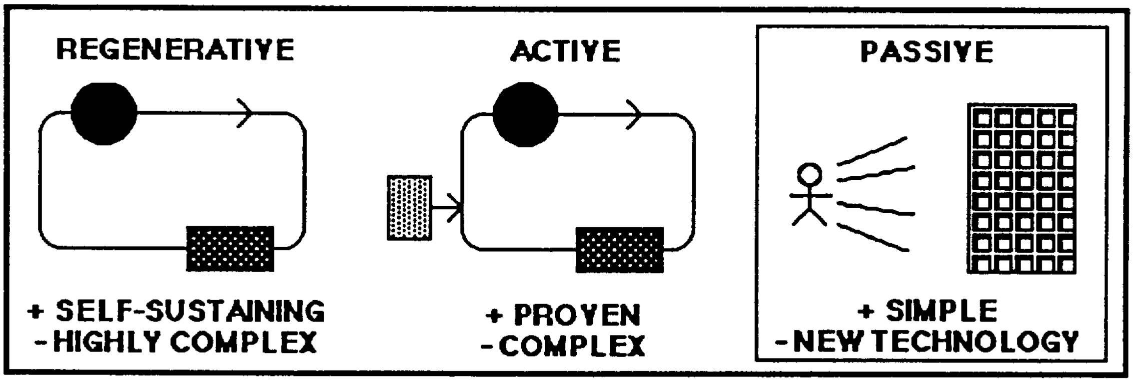
Some of the basic requirements for LiBERTy's design included maximization of reliability, simplicity, and performance. None of these factors is readily quantifiable. In choosing a temperature control system, therefore, qualitative reasoning was necessary. It should be remembered that a LiBERTy vehicle will be in orbit for months at a time. Maintenance procedures and verification tests will be difficult, and possibly not feasible. Furthermore, an active system would be highly complex and could very well break down. A passive system, on the other hand is incredibly simple and reliable.

<u>Decision</u>

Although using wax for heat absorption is a new technology, few problems are expected in its design. Because of its high reliability and its extremely simple nature, a passive temperature control system was selected for LiBERTy.

Table 6–2: Temperature Control Summary

Options



Trade

OPTIONS	REQUIREMENTS			
	REL +	COMP -	PERF +	WT -
REGENERATIVE	UNKNOWN	HIGH	HIGH	HIGH
ACTIVE	POOR	HIGH	GOOD	MODERATE
PASSIVE	GOOD	LOW	GOOD	MODERATE

Decision

A REGENERATIVE SYSTEM WOULD BE TOO COMPLEX AND HEAVY.
AN ACTIVE SYSTEM MIGHT NOT BE RELIABLE AFTER LONG DOWN-TIMES AND WOULD BE TOO COMPLEX.
A PASSIVE TEMPERATURE CONTROL SYSTEM WAS SELECTED BECUASE OF ITS HIGH RELIABILITY AND ITS EXTREME SIMPLICITY.

CONCEPTUAL DESIGN & RECOMMENDATIONS FOR FUTURE STUDY

ATMOSPHERE CONTROL

LiBERTy's atmosphere control system consists of an air flow duct, 2 recirculation fans, 4 flow control orifices, 4 isolation valves, 2 oxygen tanks, LiOH canisters, 3 pressure sensors, and 2 pressure valves. There are two independent ventilation systems, one at each end of the vehicle, for redundancy. At each inlet, LiOH will remove CO_2 from the air. A CO_2 partial-pressure sensor checks this procedure. At the outlet, fresh oxygen is bled in and a fan propels the air outward. The crew manually bleeds in oxygen with two flow control orifices and two isolation valves which are capable of serving up to four crew members. This eliminates the need for complex automated control. The oxygen tanks are Shuttle Manned Maneuvering Unit (MMU) nitrogen tanks. Each holds enough oxygen for about twenty hours, more than enough even if one ventilation system becomes inoperable. All components are Apollo, Shuttle, and Spacelab off-the-shelf hardware.

It is unlikely that major changes would need to be made in the atmosphere control system design. All components have been flight-proven, and the system is inherently simple. Tests of the integrated system should be made to ensure smooth operation and elimination of any bugs. Throughout the remainder of the design process, past experiences and successes can be relied upon, since no new technology is needed. Once a complete system is integrated, manned simulations should be conducted to verify proper performance.

TEMPERATURE CONTROL

The Structures Analyst designed LiBERTy's exterior such that little if any heat would transfer between the internal crew area and the external space environment. All avionics are placed apart from the crew in a thermally insulated compartment. Heat produced by the LiOH-CO_2 reactions was ignored for the purposes of this conceptual design. Therefore, only crew metabolic heat was considered.

Temperature control is achieved through the use of wax blankets for each crew member and wax panels covering the interior of the vehicle. The blankets are composed of n-heptadecane, which melts at 21.1°C. Each crew member can adjust his/her blanket to cover the warmest portions of the body. Because the temperature of the Space Station is nominally higher than 21.1°C, the blankets must be stored near a cold-air lay-in duct supplied by the Station. Wall-panels of n-octadecane, which melts at 27.7°C, ensure that the over-all cabin temperature does not exceed comfortable levels. After splashdown, a fan brings in outside air for temperature control and oxygen.

NASA has performed numerous analyses and concludes that a wax system could work (Reference 6–4). But because a passive temperature control system of this nature has not yet been utilized, more studies must be made to determine exactly how effective the use of wax would be. Development of this technology will then be necessary, as well as full manned tests under simulated conditions. The feasibility of locating avionics in a thermally contained area should also be further investigated.

Although a passive temperature control system agrees with the requirement for simplicity and reliability, weight can become a problem. As Figure 6–3 shows, the amount of wax needed increases with mission length.

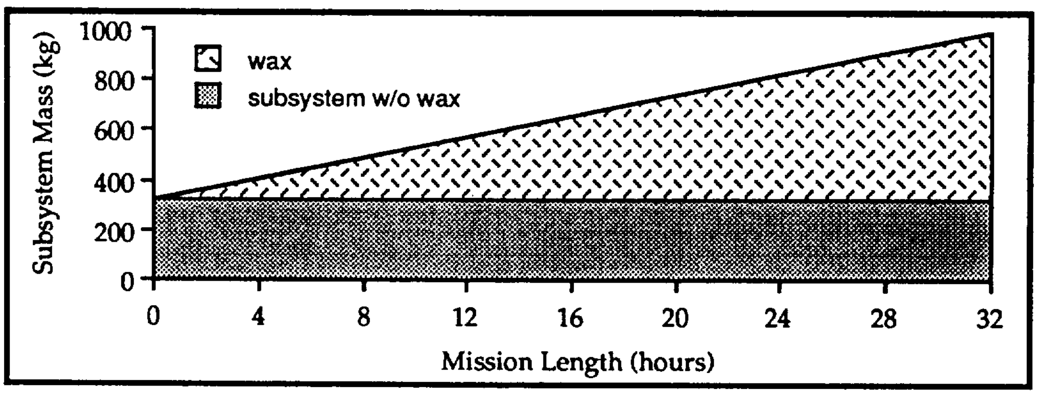


Figure 6–3: Subsystem Mass vs. Mission Length

However, it is desirable to allow as much in-orbit loitering time as possible to ensure selection of a good landing site. In cooperation with the Mission Manager and the Reentry & Recovery Analyst, a maximum mission length of 10 hours was decided upon. The minimum time between undocking from *Freedom* and splashdown is about two and a half hours. A safety margin of 4 allows a maximum mission length of 10 hours, leaving seven and a half hours for loiter time. The amount of wax required for 10 hours is about 210 kg (see Appendix, Part D). LiBERTy's design is very flexible, and it may be determined in the future that maximum mission length should be shortened or extended.

FIRE CONTROL

Fire is a danger aboard spacecraft because trash and equipment act as fuel, avionics- and crew-caused sparks provide ignition sources, and the crew's oxygen enables combustion. As Apollo 1 proved, spacecraft fires are particularly dangerous because toxic products fill up a small volume from which there is no escape.

Fire detection in a small space can easily be accomplished by crew senses. Fire, heat, or smoke would be readily observed, felt, or smelled. The Mercury, Gemini, and Apollo programs utilized this method of fire detection, and because LiBERTy's mission profile is quite similar, it will depend on crew detection also.

Several fire suppression methods were investigated. Halon 1301 is highly toxic, and because LiBERTy's avionics are located separately from the crew cabin, special electrical-fire suppression methods are unnecessary anyway. A recent NASA symposium (Reference 6–13) recommended water-based sprays for spacecraft, and this seems to be the cleanest and simplest method. Although Apollo astronauts depended on food rehydration water bottles for fire extinguishers, more advanced hardware is presumably available.

MEDICAL EQUIPMENT

If disaster should strike *Freedom*, there will be no time to use its medical provisions for crew injuries. LiBERTy therefore carries a modest amount of medical supplies and equipment. The Shuttle Orbiter Medical System Type B Kit (SOMS–B) contains injectables, diagnostic items, medications, bandages, a defibrillator, an intravenous system, and other basic supplies. LiBERTy contains this kit plus a respirator and a heart monitor. In the future, it may be determined that more equipment is needed.

FOOD & WATER PROVISIONS AND WASTE DISPOSAL

Because orbit loitering and ocean recovery delays may last several hours, provisions for food and water must be made. LiBERTy carries Shuttle-developed foods requiring no heating, as well as two Apollo-technology water tanks. Waste management bags are also included. Depending on SAR methods, recovery delays may necessitate more provisions.

CREW ACCOMMODATIONS

Eight seats will be stored flush with the wall during normal logistics missions. During an emergency return, the seats will be locked into position once the vehicle is clear of the Station. Although no major problems are foreseen, retractable seats will have to be designed and tested.

When a more final design is completed, the cabin layout must be designed. Human factors engineering and ergonomics techniques (Reference 6–8) should be used to ensure that all systems are easily operable under emergency conditions.

CONCLUSION

In summation, LiBERTy's Environmental Control & Life Support Subsystem utilizes simple and reliable methods and requires no technological leaps. Most components are off-the-shelf hardware, and the only new technology required is the use of wax for temperature control. Table 6–3 gives a component breakdown. Figure 6–4 shows the ECLSS portion of LiBERTy's mass and cost.

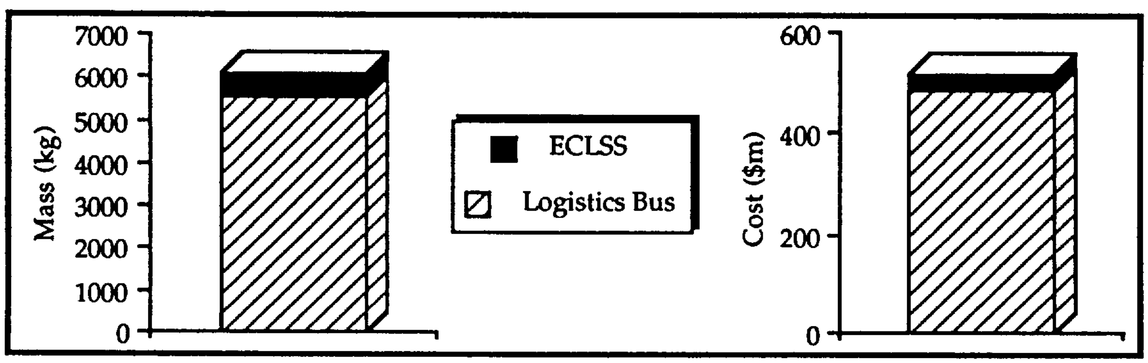


Figure 6–4: ECLSS vs. Logistics Bus

Interestingly, the burden placed on the vehicle by the ECLSS is quite small. By combing the two seemingly disparate functions of logistics bus and emergency return transport into a hybrid vehicle, great savings in development and launch costs are made.

Table 6-3: Component Breakdown

component	mass (kg)	power (W)	source
atmosphere control			
ducting	10.00 (est.)	0.00	TBD
recirculation fans (2)	3.81	34.00	A: GA 826070
flow control orifices (4)	.25	0.00	SL: MC 2094-0001-1
isolation valves (4)	3.08	0.00	STS: MC 3516-0001-1
lithium hydroxide	4.60	0.00	STS: HS SV755510
oxygen	12.70	0.00	—
oxygen tanks (2)	25.86	0.00	STS: MMU
pressure sensors	1.01	1.04	STS: HS SV755532 & SVT55537 & MC 2767-0001-1
pressure valves	2.88	0.00	A: GA 810450STS & MC 2765-0001-1
temperature control			
PS fan	1.91	17.00	A: GA 826070
PS shut-off valve	2.27	4.4 (140 max)	A: GA 816032
wax	209.40	0.00	NT
fire control			
extinguisher	3.00 (est.)	0.00	TBD
medical equipment			
kit	8.00	0.00	STS: SOMS-B
heart monitor	20.41	TBD	TBD
respirator	9.07	TBD	TBD
food, water, hygiene			
food	5.00 (est.)	0.00	STS
water	16.33	0.00	—
water tanks (2)	6.35	0.00	A: GA 812370
waste bags (8)	1.21	0.00	TBD
crew accommodations			
seats (8)	181.44	0.00	NT
TOTAL	**528.58**	**35.04 during mission 21.40 (157 max) PS**	

A=Apollo, GA=Garret/AiResearch, HS=Hamilton Standard, MC=Moog/Carleton, NT=new technology, PS=post-splashdown, SL=Spacelab, STS=Shuttle, TBD=to be determined

Data from References 6-1, 6-4, 6-11, & 6-14.

APPENDIX

Part A: Worst-Case ECLSS Mass (Reference 6–1)

A fully active ECLSS system was assumed to weigh the most.

item	mass (kg) 6 persons	mass (kg) 8 persons
seating	136	181
medical supplies	52	52
loaded system	480	528
total	668	761

Part B: Crew Volume Needs: (Reference 6–10)

$V=[(1.133 \cdot 10^{-4})t^2 + (4.026 \cdot 10^{-2})t + (2.302)]nt$

V=volume needed by crew (m^3), t=mission length (days), n=number of crew members

Part C: Atmosphere Control Trade (References 6–4 & 6–10)

Components exclusive to each system were compiled.

Passive

For a crew of eight, .32 kg of oxygen are consumed every hour. Because whole breaths are required, nitrogen is also "consumed". Nitrogen accounts for 76.7% of air (by mass), so 1.04 kg per hour will be needed. This adds to 1.35 kg of air per hour.

item	mass (kg)
helmets (8)	26.85
air	1.35/hour
total	26.85 + 1.35/hour

Active

For a crew of eight, .32 kg of oxygen are consumed every hour. Because carbon dioxide must be eliminated, .46 kg of lithium hydroxide per hour is needed.

item	mass (kg)
fans (2)	3.81
O_2	.32/hour
LiOH	.46/hour
total	3.81 + .78/hour

Part D: Wax Needed (Reference 6–4)

wax heat of fusion: h_f=241.89 kg/kJ

human metabolic heat generation rate: q_h=5064 kJ/hr (for 8 crew members)

$q_h=q_w=h_w m_w/t$

$m_w=q_h t/h_w$=20.94 kg/hr

REFERENCES

6–1 *ACRC-CERV Option Reference Concept* (JSC-23264); Johnson Space Center, National Aeronautics & Space Administration; 15 November 1988.

6–2. *Apollo Program Summary Report* (JSC-09423); JSC, NASA; April 1975.

6–3. Booker, P. J.; *Project Apollo: The Way to the Moon*; Chatto & Windus; 1969.

6–4. Buckley, Theresa M. (Crew & Thermal Systems Division, JSC, NASA);
•*Conceptual Design of a Simple, Short-Duration Environmental Control & Life Support System for the Station Crew Return Alternative Module* (JSC-23183); 20 September 1988;
•phone conversation; April 1989.

6–5. *Conceptual Design Study of a Crew Emergency Return Vehicle* (JSC-23103); Advanced Programs Office, JSC, NASA; August 1988.

6–6. "ECLSS for Hermes Examined"; *Aerospace Engineering*; p. 8, January 1988.

6–7. Gerlach, Ronald H. (CERV Office, JSC, NASA); phone conversation; March 1989.

6–8. Goodman, Jerry Ronald; *Crew Station Aspects of Manned Spacecraft Design*; Department of Industrial Engineering, University of Illinois; 1972.

6–9. Koepke, Andrew (Department of Aeronautical & Astronautical Engineering, University of Illinois); conversations; Spring Semester 1989.

6–10. Lembeck, Michael F. (Department of Aeronautical & Astronautical Engineering, University of Illinois);
•AAE 241 lectures; Spring Semester 1989;
•conversations; Spring Semester 1989.

6–11. Rambaut, Paul C.; "Foods & Nutrition in Space" (SAE TP-860926); *SAE Transactions/Aerospace*; p. 7.144, vol. 95, 1986.

6–12. Skoog, Å. Ingemar; "ECLSS for the European Spaceplane HERMES" (SAE TP-871431); Society of Automotive Engineers; 1987.

6–13. "Spacecraft Fire Safety" (NASA CP-2476); Scientific & Technical Information Branch, NASA; 1987.

6–14. "Space Shuttle News Reference"; NASA; 1978.

6–15. Stone, Howard (Langley Research Center, NASA); phone conversation; March 1989.

6–16. Svendsen, Harold O. (Lockheed); phone conversation; March 1989.

6–17. Van Vliet, Robert M.; *Passive Temperature Control in the Space Environment*; Macmillan; 1965.

REENTRY AND RECOVERY SUBSYSTEM

Jeffrey C. Berg

INTRODUCTION

For the majority of its flights, **LiBERTy** will be serving in its logistics role. On a regular 90 day cycle it will be returning waste and other materials which are no longer needed for space station operations. However, there may be a time when all or part of the operating crew will need to return to earth. For this case, reentry will need to be as safe as possible in order to keep the crew alive until they can get help back on earth. For the most part injuries will be known, the seriousness determined so that a final decision can be made whether it is necessary to return the crewmember to earth or not. During this decision process the ground controllers can begin making the necessary arrangements for a possible landing. In the worst case, because of an immediate danger to the crew, **LiBERTy** will depart immediately without any preliminary planning. By designing the module to handle a live cargo, the module will also be able to safely return any other precious material payload. Since the overall size of the **LiBERTy** module is small enough to fit into the bay of the shuttle, it could be returned this way as well. This last scenario would only likely occur if a space shuttle was already in orbit.

SUBSYSTEM REQUIREMENTS

My subsystem requirements originate from this emergency crew escape need. Reentry will have to be as gentle as possible so that an injured crew member can live until he can receive the necessary medical attention. The g forces on the vehicle will have to be kept as low as possible during atmospheric entry and during touchdown. The thermal loads experienced as the vehicle starts to interact with the earth's atmosphere will have to be radiated away and not absorbed by the vehicle. Finally the other main requirement for my subsystem is to dissipate as much of the orbital energy as possible in the atmosphere during free fall.

<u>REENTRY SCENARIO</u>

Typically, the reentry will be a normal planned occurrence. NASA will have the reentry trajectory calculated and a landing site chosen ahead of time. Primary landing sites will be near Hawaii in the Pacific Ocean, and near the Kennedy Space Center (KSC) in the Atlantic Ocean. "KSC is an obvious landing site choice due to existing search and rescue forces trained for spacecraft rescue, and to the existing SRB recovery capability. Hawaii, with extensive air/sea rescue capability, has been proposed as a practical alternate."(see Reference 7-1) If the weather is bad, high winds or storms, at the particular landing sites a secondary site can be chosen. (Refer to table 7-3.) For most of the landings it will be desirable to land during daylight in order to facilitate recovery operations. It will be assumed that landing will be constrained to a 9 hour time band at any site. Therefore a backup deorbit should also be planned in case the original attempt is missed.

LiBERTy will follow the same sequence to return whether a crew is present or not. I will be describing the mission from the crew escape standpoint.

Crew Departure

(Refer to Figure 7-1.)

The crew enters the module and activates all systems, for example power and life support. Explosive bolts fire releasing the module from the space station dock. If any angular rotation is imparted due to the release, the RCS is designed to quickly damp out this motion and stabilize. A quick impulsive burst of the RCS engines moves the module .8 km away. The module needs to be far enough away from the space station so that the deorbit Δv does not impinge the space station. The ground controllers will then initiate the necessary deorbit burn at the correct attitude.It takes approximately 90 minutes to completely orbit the earth. Therefore, the module can loiter for 90 minutes in order to get in the correct place to impart the Δv. There is enough propellant and life support allowed for in case the initial deorbit attempt is missed.

Entry Interface

The Δv slows the module from the orbit of the space station, decaying toward earth. Attitude adjustments are then made to give the module a entry flight path angle of -1.5°. At this entry flight path angle, the g forces experienced on the module over three minutes will be less than 4 g's. To radiate the heat associated with reentry interaction with the atmosphere, **LiBERTy** is coated with the tiles similar to those that protect the Space Shuttle. The heat experienced at reentry is near 1650°C. **LiBERTy** begins interacting with the atmosphere of the earth at an altitude of 121.9 km. The overriding feature of the atmosphere, as far as its effect on the spacecraft is concerned, is the density. The effects of density as the module falls to earth as well as the decelerating role of the parachutes will be enough to dissipate the orbital energy in the atmosphere. The **LiBERTy** module will enter the atmosphere travelling 7620 m/sec.

Parachute Deployment and Touchdown

At an altitude of 7.75 km, two drogue chutes are released and opened to begin providing stability to the falling module. At an altitude of 3.0 km, pilot chutes deploy three main parachutes to slow the vehicle to 30 m/sec on impact with the water. There is redundancy introduced here in case one of the drogue chutes or main parachutes fail to operate. The chutes are triggered for deployment by using barometric pressure with navigational altitude as a backup. A UHF beacon is also activated. The signals will be received by a Search and Rescue Satellite to help with the recovery operation. On touchdown the crew will open vents in the cabin and turn on circulating fans until they are recovered. The naval recovery teams will move in to help the crew and pick up the module for processing and return to service.

Impact

Water impact forces will be a maximum of 10 g's for a duration of .2 seconds. The crew couches will be constructed with shock attenuators. The couches will be placed so that 90% of the impact forces will be felt through the x-axis of the body. The other 10% can be divided between the y and z axes.

Accuracy in hitting the desired target point plays a vital part in determining the amount of time required for rescue. Contributions to the landing footprint include errors or uncertainties in the orbit impulse, navigation, vehicle mass, aerodynamics, and atmospheric density. The landing footprint area is most effected after drogue chute deployment at an altitude of seven kilometers.

Table 7-1 Entry Load Effects on Downrange and Crossrange

Error Source	Miss Distance Sensitivity at 7 km
Deorbit burn magnitude	148.9 km/mps
Deorbit burn attitude	
In-plane -	50 km/deg
Out-of-plane -	10.7 km/deg
Vehicle deorbit weight	3.3 km/kg
Ignition delay	6.5 km/s
Initial orbital altitude	44 km/km
Density bias	1.9 km/% bias
L/D	298 km downrange/0.1 L/D
	33.4 km crossrange/0.1 L/D

With this error information and the possible landing sites in mind, in the worst case the module would be adrift for 7 hours. For the majority of the returns a recovery will be achieved within 2 hours.

PROBLEM AREAS

This design will have to go through further design and modelling before it is ready to be deployed. Over the course of development several drop tests will have to made as well wind tunnel and hypersonic modelling of the basic LiBERTy design. The overall driver for this design is that it employs all present day technology and proven tested materials. As more is discovered, it might be wise to add these innovations to the project. Additionally, more formal studies will have to done on the affects that g forces have on an injured astronaut. What is the upper limit that an injured human can withstand during the return to earth? Healthy pilots can withstand 9g's or possibly more however a weak or injured heart should not be put to this kind of test. Another area of concern is making the landing area as small as possible so that rescue and recovery crews can get to the module as quickly as possible. A current study on parafoils gives an indication of there application to the LiBERTy module to provide some steering during descent. The parafoil design study is to be completed and ready for deployment by 1995 but has already seen numerous design setbacks effecting this schedule. Therefore, if the design can be proven to work prior to the completion of **LiBERTy**, it can be easily added to the system. Finally, during the course of design the naval recovery teams will have be trained on how to handle recovery of **LiBERTy**. The same naval recovery forces that were present during the Apollo Space Program are no longer active.

Table 7-2. Reentry Mission Timeline

Event	Time (Mission Elapsed Time)
Crew Entry	0:00
Air Supply StartUp	0:02
Systems Activation	0:04
Departure from Space Station	0:05
Landing Site Selected	0:10
Pre-burn attitude established	0:15
Burn Initiation calculated by the computer	0:16
Loiter to reach burn Position (90 minute maximum)	0:16
Deorbit Burn Initiated	1:46
Burn Terminated	1:57
Entry attitude established by RCS	2:20
Entry interface	2:25
Parachute deployment/UHF Beacon activated	2:33
Touchdown	2:38
Manual vents open/circulation Fans started	2:40

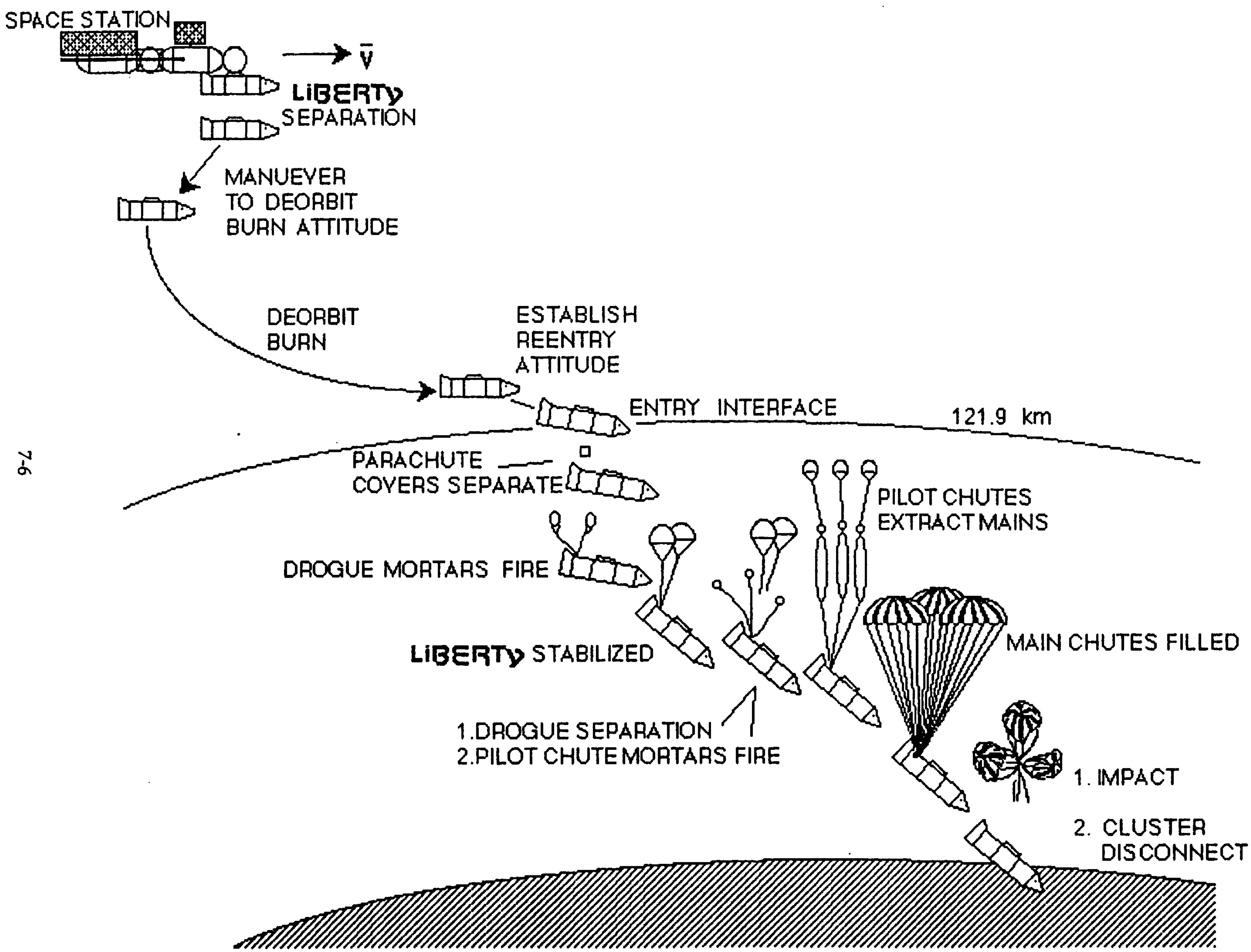


FIGURE 7-1 LiBERTy REENTRY PROFILE

Table 7-3. Selected Landing Sites

Landing Site	Location	
	Longitude (deg)	Latitude (deg)
1. KSC	279	28.5
2. Dakar	342	15
3. Diego Garcia	71	-7
4. Okinawa	126	27
5. Guam	144	14
6. Fraser Island	152	-25
7- Hawaii	201	22

Table 7-4. Landing Parachute Sizing

SYSTEM	NUMBER	SIZE (cm^3)	WEIGHT (kg)
Drogue Parachute Assembly	2	55716.0	36.3
Pilot Parachute Assembly	3	19664.5	17.7
Main Parachute Assembly	3	5497773.2	190.5
TOTAL			244.5

Table 7-5. Water vs. Land Touchdown

	advantages	*disadvantages*
WATER	simpler design	possible water damage
	less expensive	
	uses present day technology	
LAND	land close to medical assistance and processing	complicated structural design
		landing gear
		25-30% greater cost

Table 7-6. Parachutes vs. Retro-rockets

	advantages	*disadvantages*
PARACHUTES	uses present day technology	lacks much control during descent
	less expensive	
RETRO-ROCKETS	good control	never been tried by US
	low impact speed	one more element to fail
		adds explosion danger during reentry from propellants

APPENDIX 7A

$$a_{max} = \frac{V_{entry}^2 \sin \gamma_{entry}}{2\ e\ g\ H_s}$$

a_{max}	= max acceleration in g's
V_{entry}	= entry velocity (in m/sec)
γ_{entry}	= entry flight path angle
e	= e^1 = 2.718281828459
g	= gravitational acceleration
H_s	= Atmospheric Scale Height (~ 6920 m)

entry is at 121.9 km altitude

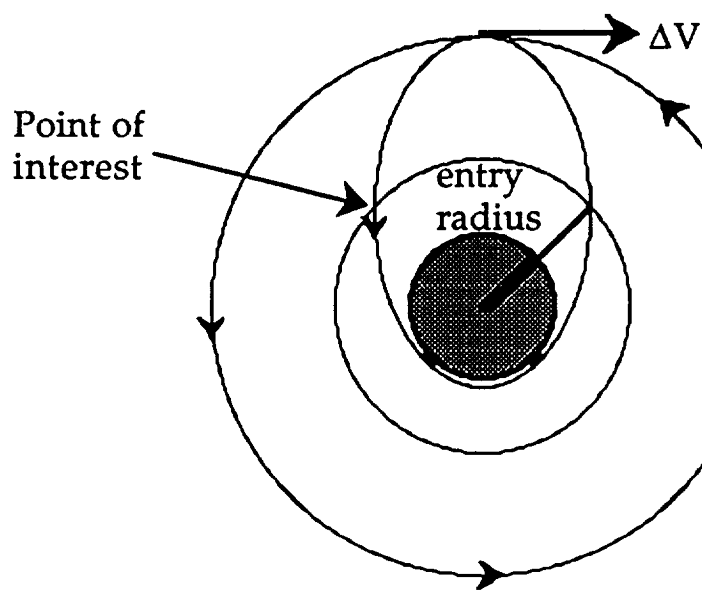


Figure 7-2. Orbital Transfer

V_{entry} is determined using the Vis-Viva Equation

$$V_{entry}^2 = \left(\frac{2}{r} - \frac{1}{a}\right)$$

$$\cos \gamma_{entry} = \left[\frac{a^2\,(1-e^2)}{r\,(2a-r)}\right]$$

Table 7-4. Entry Interface Velocity, Maximum g's, and Δv requirement

V_{entry} (km/sec)	γ_{entry}	g's	Δv (km/sec)
7.580367	-1.0°	-2.717298	0.1512502 max
7.580367	-1.25°	-3.396525	0.0882554 min
7.580367	-1.5°	-4.075688	
7.580367	-2.0°	-5.433768	
7.580367	-3.0°	-6.791434	
7.580367	-2.5°	-8.148583	
7.580367	-5.0°	-13.56994	

Thermal Control

Thermal Loads

Peak Stagnation Heat Rate

$$q_{max} = \frac{3.5 \times 10^7}{\sqrt{KR_n}} \left(\frac{V_e}{V_c}\right)^3 \frac{W}{m^2}$$

Total Heat Loads

$$Q_o = \frac{10^8}{V_c}\left(\frac{V_e}{V_c}\right)^2 \left\{\frac{m}{C_D A \rho_o}\left(\frac{H\pi}{R_n \sin|\gamma_e|}\right)\right\}$$

where

$$K = \frac{C_D A \rho_o H}{m \sin|\gamma_e|}$$

V_c = circular speed (7900 m/sec)

R_n is nose radius (meters)

C_D is the drag coefficient

$= 1 - \cos^4\theta$

A is the Reference Area

$= \pi R_n^2$

REFERENCES

7-1 ACRC-CERV *Option Reference Concept* (JSC-23264); Johnson Space Center, National Aeronautics & Space Administration, 15 November 1988.

7-2 *ARS Phase 2 Third Quarter Review: Advanced Recovery Systems Study for the Next Generation Space Transportation System,* Contract No. NAS8-36631, Melbourne, Florida: Pioneer Systems, Inc. Aerospace Recovery Division, July 12, 1988.

7-3 *Conceptual Design Study of a Crew Emergency Return Vehicle* (JSC-23103); Advanced Programs Office, JSC, NASA, August 1988.

7-4 Lembeck, Mike. "Reentry", AAE 241 Lecture Notes #238.13, University of Illinois AAE 241 Flight Vehicle Design Class, 1989.

7-5 Meyer, Scott. "Atmospheric Entry", AAE 241 Lecture Notes #238.16, University of Illinois AAE 241 Flight Vehicle Design Class, 1989.

7-6 Loh, W. H. T. Dynamics and Thermodynamics of Planetary Entry, Englewood Cliffs, New Jersey: Prentice-Hall, Inc. 1963

7-7 Olstad, Walter B. ed., Entry Heating and Thermal Protection, New York: American Institute of Aeronautics and Astronautics, Volume 69-Progress in Astronautics and Aeronautics, 1980.

7-8 Vinh, Nguyen X., Adolf Buseman and Robert D. Culp, Hypersonic and Planetary Flight Mechanics, Ann Arbor, Michigan: The University of Michigan Press, 1980.

Group 7

David M Lhuhus [redacted]
Steven L Woods [redacted]
Eddie Foley
John J Beirne [redacted]
Shaun P. Murphy
Glenn Fermogle [redacted]

C-2

Command and Data Subsystem for LEMBEC
by Glenn Fermoyle

The command and data control sybsystem will be the nerve center of the logistic resupply module. It must perform a variety of operations critical to the success of every mission.

One of the basic requirements of the command and data subsystem (CDS) is to collect telemetry from all of the other subsystems. It must also send this telemetry to the ground. After receiving data from the subsystems, the CDS must interpolate this telemetry and send commands to the subsystems to tell them what to do. Included in this procedure is the power switching. Power onboard a spacecraft is severely limited, therefore, the CDS must decide where the power is needed.

The CDS must have an interface with the crew members, if any, while the ship is in flight. How much interaction the crew will have will be discussed in a later section.

One of the basic designs of this resupply module is that it be autonomous. This includes rendezvous and docking. This will be done by the astronauts in the space station once the module approaches the station within one hundred feet. All other vehicle components will be operated under positive space station control at all times.

The basic components of the command and data subsystem are the computer, radio, antennae and crew interface.

The computer used on the module will be either a 1553 or an OBDH standard which is a low speed serial databus. The databus will be connected to everything that needs to give it data and receive commands. The guidance, navigation and control systems will have their own databus connected to the main databus. The guidance, navigation and control databus will have the ability to control the entire flight autonomously through the interface between the GNC and main databus. Other ways the main databus can be told how to control the module are by the crew interface or manual override controls.

The main way to give the databus instructions is by the S-band or the tracking and data relay satellite system (TDRSS). When directly overhead of a ground station, the module will receive commands directly from the ground. When out of range, the commands will be sent through TDRSS by the Ku-band frequency. Both the S-band and Ku-band can be used to communicate with TDRSS.

The Ku-band system will also be used for the autonomous rendezvous and docking procedure by using it as a pulse Doppler radar to get the module to the space station.

A UHF transceiver will be used for the transmission and reception of voice with air traffic control facilities and ground command stations during decent. Also, the UHF can be used for communication with astronauts participating in extra vehicular activities around the space station.

The information on the S-band, Ku-band and UHF systems are located in the communication appendix at the end of the paper.

The information about antenna numbers and type are also in the communication appendix. The size of the parabolic antenna used for the Ku-band transmission and reception is 0.6 meters in diameter as noted in the JBIS article for module of this size and function. Every antenna except the parabolic antenna will be flush mounted on the surface of the vehicle and covered with the same thermo-insulating material as the rest of the spacecraft. The mass of each subsystem (including antennae) is listed in the communication appendix as is the power of each subsystem.

The crew interface with the systems will be as minimal as possible because the module is specified to be completely autonomous. Due to the relatively short flight duration (24 hour maximum) and method of reenty (capsule splash-down), minimal crew interface will be applicable.

COMMUNICATIONS

APPENDIX

Parameters for the S-band, Ku-band and UHF systems

UHF Communication System Characteristics Summary

Information Channel	Response	Modulation	Frequency, MHz
Voice	300-3000Hz	Amplitude	243.0 259.7 296.8

S-band to TDRSS forward link characteristics

Information channel	rate	carrier frequency
voice 1	32 kbits/s	2106406300 to 2041947900 Hz
voice 2	32 kbits/s	
command	6.4 kbits/s	

S-band to TDRSS return link characteristics

Information channel	rate	carrier frequency
voice 1	32 kbits/s	2287.5 to 2217.5 MHz
voice 2	32 kbits/s	
telemetry	128 kbits/s	

Ku-band link interface characteristics

channel	rate	carrier frequency
voice 1	32 kbits/s	
voice 2	32 kbits/s	
command	6.4 kbits/s	13.775 GHz

from IEEE 1978

System Mass & Power Breakdown

SUBSYSTEM	MASS(kg)	OPER.PWR.(watts)
Data Handling	48	60
S-Band comms	18	20
Audio comms	24	30
Ku-Band comms	128	90
GNC	70	90
	288	290

Antenna Characteristics

Antenna	Quantity	Frequency	Type
UHF	1	UHF	Annular slot
S-Band quad	4	S	Crossed dipole fed cavity fixed array
S-Band hemi's	2	S	Crossed dipole fed cavity
Ku-Band	1	Ku	Parabolic

from JBIS

COMMUNICATION BIBLIOGRAPHY

Carrier, Louis M.
Pope, Warren S.

1978 "An Overview of the Space Shuttle Orbiter Communication and Tracking System" IEEE Transactions on Communications, vol 26, no. 11, November 1978.

Cravey, Donald N.
Harton, Paul L.
Johnson, John H.
Kuo, Yeng S.
Loh, Yin-Chung
Teasdale, William E.
Tu, Kwei

1987 "Space Shuttle Communications and Tracking System", Proceedings of the IEEE, vol. 75, no. 3, March 1987.

Hempsell, C.M.
Hannigan, Russell J.

1989 "Multi-role Capsule System Description", Journal of the British Interplanetary Society, vol. 42, no. 2 February 15, 1989.

MISSION MANAGEMENT, PLANNING AND COSTING SUBSYSTEM

for
LEMBEC

System Analysis: Steven L. Woods

University of Illinois
AAE 241
May 2, 1989

Mission Management, Planning, and Costing Subsystem

Expectations

It is expected of the Mission Management, Planning, and Costing Subsystem of the LEMBEC to identify payloads, integrate payloads into transport module, find the mass budget, select a launch vehicle, find the orbit insertion altitude and velocity, find the mission delta-V, give a mission timeline, derive the mission costs, and the effects of planning on other subsystems.

Payload Identity

A ninety day total logistic requirement excluding the ECLSS was derived by the Spring '89 AAE 241 class. The requirements are broken down into the up/down mass and up/down volume. These totals are then further broken down into pressurized, unpressurized, fluids, and propellants values for the crew/station and customer. The total is summed to 16220.92 kg up and 13904.30 kg down. The total volume up is 69.06 m^3 and 62.59 m^3 down. See Table 1 for the complete values. The internal volume of the LEMBEC must accomodate the totals listed. The individual gas and liquid bottles are not accounted for. The logistics turns out to be the driving factor in sizing one LEMBEC for 90 days, as opposed to an eight-man crew where their volume takes up approximately 18 m^3 and their mass total (based on a 90 kg man) sums to 720 kg. See the ECLS Subsystem for a closer look at the eight-man totals.

90 Day Total Logistics Requirements

	Mass Up (kg)	Mass Down (kg)	Volume Up (m**3)	Volume Down (m**3)
Pressurized				
Crew/Sta.	4148.56	3497.99	14.78	11.50
Customer	4954.14	4757.39	13.92	13.75
Unpressurized				
Crew/Sta.	513.01	513.01	4.53	4.53

Table 1

Customer	4152.18	4152.18	32.64	32.64
Fluids (gas/liquids)				
Crew/Sta.	360.61	0.00	0.45	0.00
Customer	365.14	173.73	0.50	0.17
Propellants				
Crew/Sta.	45.36	0.00	0.57	0.00
Customer	1681.92	0.00	1.68	0.00
Totals	16220.92	13094.30	69.07	62.59

Crew/Station				
Pressurized	4148.56	3497.99	14.78	11.50
Unpressurized	513.01	513.01	4.53	4.53
Total Pressurized Sum	4661.57	4011.00	19.31	16.03
Customer				
Pressurized	495.14	4757.39	13.92	13.75
Unpressurized	4152.18	4152.18	32.64	32.64
Total Pressurized Sum	9106.32	8909.57	46.56	46.39

Notes:

1) Largest item guaranteed to fit through hatch (127 cm x 127 cm).

2) Internal volume must accomodat totals listed here, individual fas and liquid bottles need not be accounted for (i.e. no need to size tanks for logistics items, or worry about umbilicals to these items).

3) ECLSS for CERV operations not accounted for here.

A decision to pressurize the entire LEMBEC has been made. Justification of that decision are as follows: (1) When unpressurized logistics are loaded onto LEMBEC on earth, they are already pressurized, and (2) Complications occur when materials need to be moved from one section to the other. Since the decision to have the entire module pressurized, respective pressurized and unpressurized values can be summed to one total pressurized value for the crew/station and one total pressurized value for the customer.

The up and down logistic mass and volume figures for the crew/station come from such things as: consumables, personnel support, housekeeping, waste management, compacted trash, replacement spares, and etc. The figures for the customer support come from such things as: MTL, SLM(US), plant/animal, human research, ESA research, customer servicing, and etc.

In the case of an emergency escape, the crew become payload. The number of escaping crew members can range from 2-8, depending on the situation at hand. A couch changeout will need to be made if this is the case. The arrangement and seating priority will be discussed in the Structures Subsystem.

Payload Integration

Payload integration has three aspects to it: (l) up logistics configuration, (2) down logistics configuration, and (3) crew escape configuration. The transportation accomodation for the standard LEMBEC will consist carriers composed of racks (106.68cm x 189.12cm x 91.44cm)/non racks, refrigerator/freezer and life sciences accomodations. A metamorphasis will need to take place at the Space Station in case of emergency escape. Carriers will be exchanged for couches. For a layout of the couches and carriers, see Structures Subsystem. The largest items are guaranteed to fit through the 127 cm x 127 cm hatch.

Mass Budget

A critical aspect of the LEMBEC is the system's mass. A subsystem breakdown is given in Table 2. This table corresponds to a Titan IV expendable launch vehicle. The mass estimates are used for two purposes: (1) Selecting a lauch vehicle, and (2)

Parametric costing techiques. The procedure for estimating the mass is adopted from JBIS (Hannigan 69,79-81).

The system budget in Table 2 breaks the subsystems down into masses used in each. The first column of the table gives the raw estimated mass by each subsystem, and the second column gives subsystem masses after unit and subsystem level margins have been added giving the subsystem specification mass. The total margin held by the subsystems is generally greater than 10%, and in a couple cases 20%. Usually 5% is held at the subsystem levels while the rest is distributed to equipment.

System Mass Budget

SUBSYSTEM	Estimated Mass (kg)	Specified Mass (kg)	% Margin
MMPC			
Logistics	16221	16221	
Structures			
Structure	350	389	11
Miscl.	154	162	5
Power and Propulsion			
Fuel	1722	1894	10
Propulsion	389	475	22
Power	257	290	13
Attitude and Articulation Control			
A. A. Propulsion	144	151	5
Sensors	65	72	10
Command and Data Control			
GNC	70	80	12
KU Comm/Radar	128	145	13
Data Management	48	55	13
S- Band Comm	18	20	10
Audio Comm	24	30	20
Life Support and Crew System			
ELCSS	133	149	12
Reentry and Recovery System			
Thermal Protection	1310	1493	14
Recovery	236	270	13
TOTAL MASS	21269	21896	
Margin	957 (4.3%)	330 (1.5%)	
Specified Mass in Orbit	22226	22226	

Total available margin, the raw estimate to systems specified maximum, is 4.3% of the 22226 kg available. 1.5% has been distributed to the other subsystems and equipment. Concluding from this table, nearly all of the launch vehicle capabilities are exhausted.

Launch Vehicle

The $110 million Titan IV was selected as the primary launch system for the LEMBEC. The reasons for this choice are that it is the largest and most powerful of the launch vehicle under consideration. See Table 3 for comparisons to other Expendable Launch Vehicles (ELV's). An initial drawback of the Titan IV, it has yet to experience its first launch. The initial launch capability will be 1994.

Expendandable Launch Vehicle Comparisons

	Titan IV SRM	**Titan IV SRMU**	Titan III	Atlas IIA
Orbit	185.2 km x 185.2 km	**185.2 km x 185.2 km**	185.2 km x 185.2 km	185.2 km x 185.2 km
Upperstage	NUS	**NUS**	NUS	NUS
Launch Site	VAFB, CCAFS	**VAFB, CCAFS**	CCAFS	CCAFS
PLF	17.07 m	**17.07 m**	----	3.29 m x 10.36 m
Capability	17690.1	**22226.03 kg**	14152.08 kg	7121.40 kg
Record	unproven	**unproven**	proven	proven
Cost	$110 M	**$110 M**	$110 M	$59 M

Information provided by John J. Neilson

Bold indicates selected ELV

The Titan IV is a commercial launch vehicle which is based on the heritage of the Titan family going back to the Titan I and II ICBM's. See Table 4 for Titan IV Space Launch Vehicle Configuration. The Titan IV consists of three stages. Stage 0 consists of the large Solid Rocket Motors (SRM) ignited on the ground. State 1 and 2 (collectively called the core vehicle) use storeable propellants. Compared to the Titan 34D, it has a distinctive hammerhead shape fairing, stretched tankage on both Stage 1 and Stage 2, 1-1/2 additional segments on each solid rocket motor and is designed to operate with an Inertial Upper Stage (IUS) or with No Upper Stage (NUS) at all. The NUS version was selected for the LEMBEC on the basis that it gives a sufficient altitude.

Titan IV Space Launch Vehicle Configuration

No. of Stages	Name or Designator	Propellants	Thrust	Height	Diameter
0	SRM	Solid	3,324,000 lb	112 ft	10 ft
1		Aerozine 50/ N_2O_4	549,700 lb	86.5 ft	10 ft
2		Aerozine 50/ N_2O_4	105,900 lb	29.9 ft	10 ft

information provided by Nielson

Note: This is the configuration with the SRM.
The SRMU information was unavailable, but should be similar.

The LEMBEC will need the upgraded SRM's (SRMU) for the Titan IV which are currently being developed. Only limited information on the performance of the SRMU versions are available at this time because the development program is not complete. Currently the payload capability of the Titan IV NUS with SRMU's is about 22,226 kg for a low earth orbit (185.2 km x 185.2 km) from CCAFS.

The Titan IV will inject the payload into a 28.6°, 185.2 km x 185.2 km parking orbit from which the LEMBEC can propel itself to a desired orbit between 209-430 km. See Power and Propulsion Subsystem for this additional delta-V needed. Payload in this case is defined as the total weight injected into the parking orbit including the spacecraft.

The fairing size is a limiting factor to the overall size of the LEMBEC. Caution must be taken to make sure that the module fits in the fairing. The Titan IV offers three different Payload Fairing (PLF), sizes. The PLF has a diameter of 5.08 meters and can be constructed in lengths of 17.07, 20.17, 23.16, 26.21 meters. A more desirable fairing would have a length of 8.08 meters. Figure 1 shows a LEMBEC inside a Titan IV PLF. As an aside, the Titan IV was designed as the Complementary Expendable Launch Vehicle (CELV) which was intended to provide back-up launch capability to the shuttle for certain DoD payloads. This accounts for the four fairing sizes.

Titan IV PLF loaded with LEMBEC

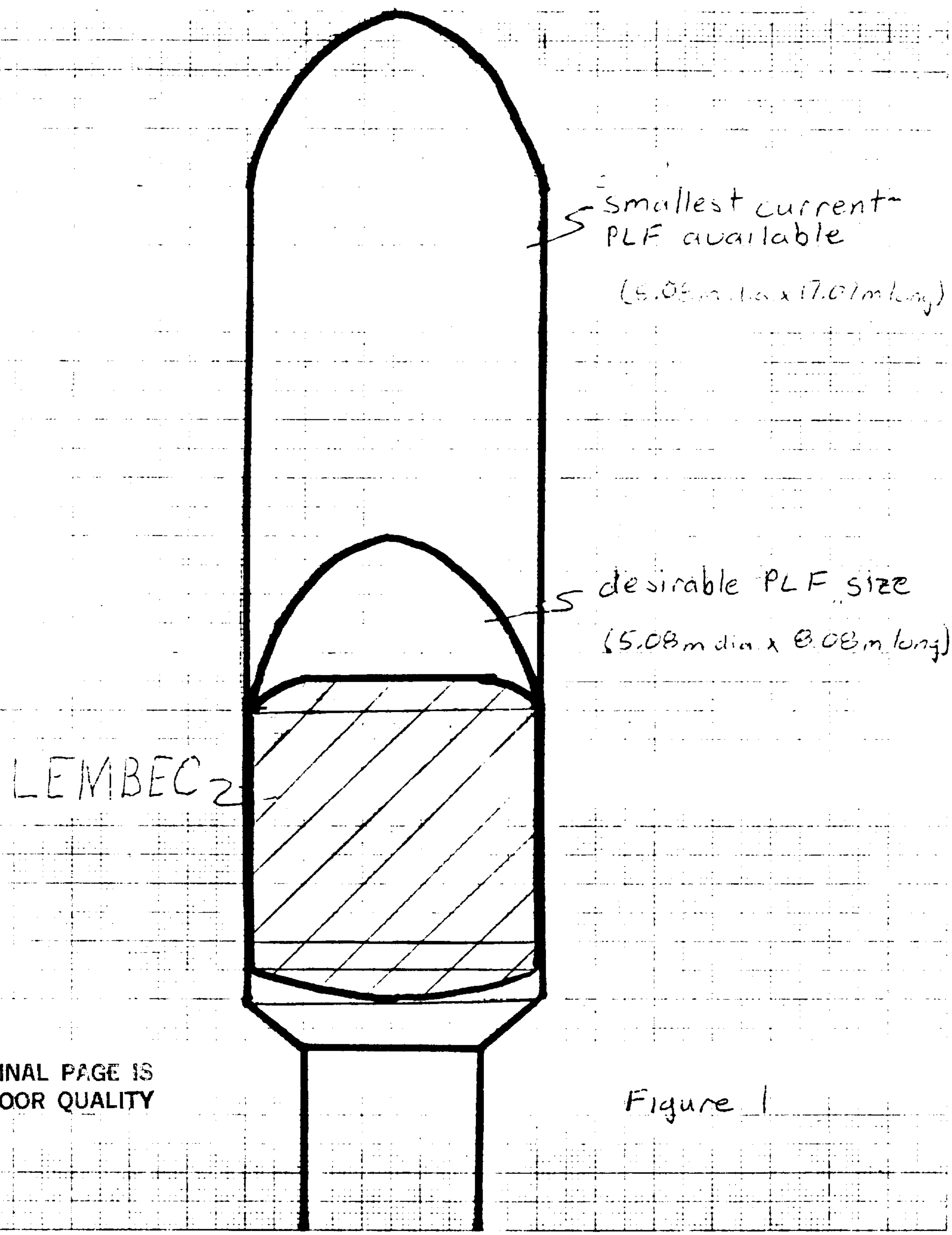


Figure 1

There are two possible launch sites for the Titan IV. At Cape Canaveral Pads 40 and 41 will have NUS capability and at Space Launch Complex 4 East (SLC-4E) at Vandenberg. An additional complex (SLC-7) is planned for the initial launch capability in 1994.

The Titan IV has no launch history of its own, but draws on the same heritzge as the Titan III. Martin Marietta quotes a success rate of "in excess of 96%" based on a record of 130 successes out of 135 launches. Titan IV launches at both CCAFS and VAFB will be conducted by Martin Marietta launch crews under Air Force direction. (Nielson 8-9)

Delta-V Requirements

The LEMBEC goes through many velocity changes during a mission. In this section, the delta-V's will be summed to show the required delta-V for the mission. Table 5 shows a breakdown of individual delta-V's. For the derivation of delta-V's, see the respective subsystems.

Occasion	ΔV (m/s)
Launch	7767.652
Out to orbit	141.037
Braking at Space Station	0.089
6 platform trips	5.832
Leaving Space Station	
w/ logistics	0.139
w/ 8-man crew	0.300
Miscl. slews	0.003
Reentry slew	0.145
Reentry	108.610
Atmospheric drag	0.009

Mission ΔV with logistics only: 8023.516 m/s

Mission ΔV with logistics up and 8-man down: 8023.677 m/s

Initially the LEMBEC experiences a velocity change at launch. It's velocity will have an approximate delta-V of 7769 m/s, neglecting earth's rotational velocity, which will get the module to a 185.2 km circular orbit. See Appendix A for launch calculation. Then an additional delta-V of 141.031 m/s is needed to achieve the 430 km Space Station orbit (Space Station orbit can range from 290 km to 430 km). A delta-V to brake the module at the Space Station is 0.089 m/s. For each platform trip a total delta-V of 0.972 m/s is required. For this mission, six platform tours will be figured in.

When the LEMBEC is ready to return home it may have logistics or crew as payload. The delta-V needed to leave the Space Station when loaded with logistics is 0.139 m/s. With a eight man crew the delta-V required will be 0.300 m/s. Once off the Space Station a delta-V slew is required at 0.145 m/s to set up for reentry. Miscellaneous slewing is required during the arrival and departure of LEMBEC to total a delta-V of 0.003 m/s. For reentry, the required amount will be a delta-V of 108.610 m/s. A small delta-V of 0.009 m/s is also accounted for by the atmospheric drag.

All the delta-V's summed for a logistics only mission with six platform trips comes to a total of 8023.516 m/s. While a logistics launch, six platform trips, and an emergency return total the delta-V at 8023.677 m/s.

Development Program Timeline

The timeline is adopted from the idea developed in JBIS (Hempsell 92-94). The development of the LEMBEC follows a four phase program similar to the development phase of the earlier Gemini and Apollo missions. This development corresponds more closely to the development of a commercial satellite than to the Space Station or the Space Shuttle. The LEMBEC is relying mostly on currently available technologies, so a lengthy research period is not necessary.

Development Program

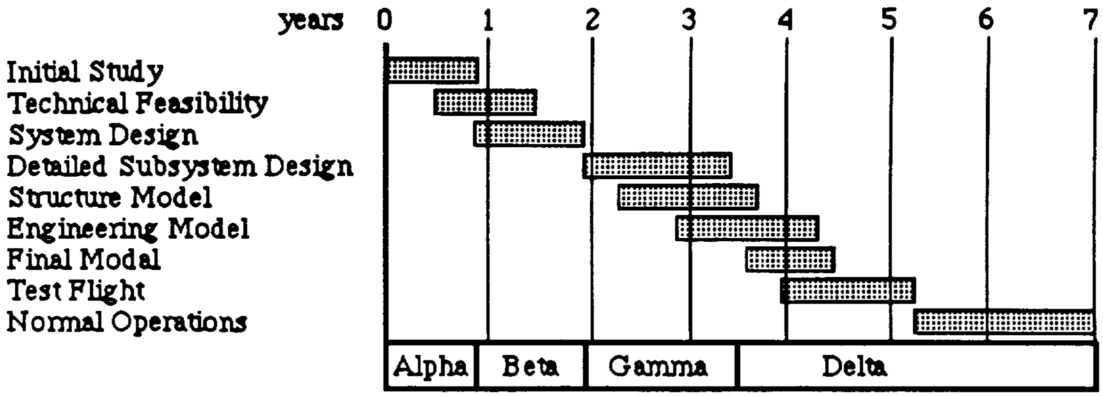


Figure 2

Figure 2 is a bar graph showing the four phases known as Alpha, Beta, Gamma, and Delta. The Alpha Phase consists of an initial study of the problem and the techical approaches needed to solve the problem. The Beta Phase is a year long, and consists of a system level design leading to the individual subsystems. This phase for Space Station is considerably longer due to the systems complexity. The Gamma Phase begins with the start of detailed subsystem designs. A structural model is to be built soon into this phase, so detailed structual work is needed from the Beta Phase. The engineering model is to be built to test subsystem interactions. Phase Delta begins with the building of a final model. This model is not intended to be flignt capable, but it should perform as the LEMBEC is expected to. The initial test flight will consist of carrying non-critical logistics to the Space Station, so the module's lose would not adversely affect the station. A first flight directly to the Space Station is used instead of a simple orbit because the LEMBEC relies on current technologies which are already proven. During return to Earth, the LEMBEC's life support system will be operated, and monitoring equipment will be used to ensure its proper operation.

Number of Vehicles

The LEMBEC program will use a total of four eight-man vehicles in operation at once being launched individually every 90 days. For the reasoning behind having one eight-man vehicle every 90 days, see the costing section of this subsystem. At least two modules will be in space at all time, and this is done because if one or several members of the Space Station have to return to Earth, a module will still remain as an escape for the remaining crew members. If one of the two modules is touring another platform while the other module returns to Earth, the Space Station has a Health Maintenance Facility (HMF) which can provide critical care for up to 28 days to assist crew members who might be hurt in the interum before the module at the platform returns. If one module returns to Earth with an injured crew member then, of course, only one module will be in space until the next logistic resupply occurs. Based on the experience of Antartic bases and submarines it is estimated a crew member will need to be evacuated from the Space Station on an average once every four years (Hempsell 53). So, if another critical injury occurs, then the Station may have to be abandoned temporarily and the entire crew returned to Earth. Idealistically, someday there would be three LEMBEC's in space at once. Two would act as the emergency vehicles, while the other was touring a platform. This too has its drawbacks, since one module could be in space up to 9 months (possibly idle for the last six).

Logistic Resupply Timeline

The timeline in Figure 3 traces the normal operating cycle of the LEMBEC program. The operation timeline begins with the recovery of the module. At the same time there would be one module each at the 90, 180, and 270 day marks, so each module is separated by 90 days on the timeline.

Normal Operating Timeline of LEMBEC

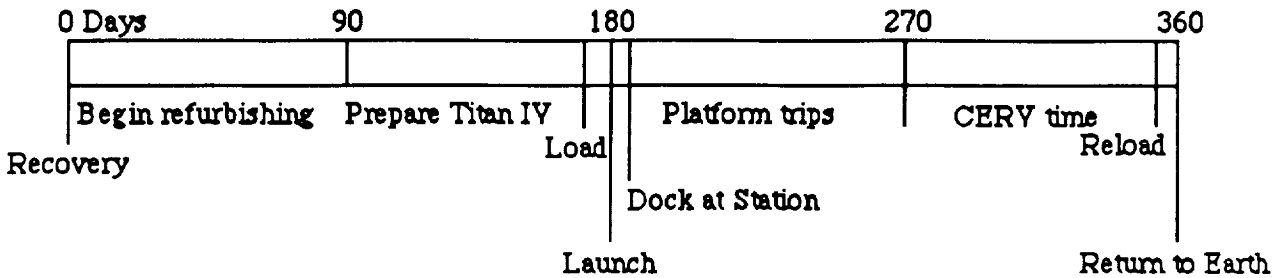


Figure 3

Since it is desired to have four vehicles with two in orbit at all times, 180 days is then provided for unloading the module, refurbishing and testing all systems on the LEMBEC, preparing the Titan IV for launch (which takes 153 shifts), and then loading and launching the module. Providing a full half-year to turn around one module will ensure plenty of time to correct problems or plan alternatives to keep the Space Station supplied.

After launching and docking at the Space Station, the module will be unloaded by hand by the astronauts. The zero gravity environment will make this a relatively easy task, and no more than two days is anticipated for an unloading. The module is to stay in orbit for a half-year. For the first 3 months, when it is not visiting a platform, it will be the stand-by module for a rapid evacuation of the crew. At the 270 day mark, one other LEMBEC will arrive and one other one will return to Earth. For the 270 to 360 day marks, this module will be partially loaded with items not easily stored on the Space Station while the newly arrived module will serve as the emergency vehicle. Within a week before the arrival of the next logistic resupplier, this module will be loaded with all of the material to be returned to Earth. Upon arrival of the new logistics, the module will return to Earth, be recovered, and start the standard unloading and refurbishing process again.

Crew Evacuation Planning

The scenarios for a crew evacuation are virtually infinite. To aid in planning, however, two types of evacuation will be considered, a total crew evacuation and a partial crew evacuation.

A total evacuation can be an emergency requiring immediate departure or the more likely case of an evacuation that is unhurried but necessary. The threats the Space Station faces have already been discussed, so this section concentrates on planning for departure rather than the reasons for departing. A total crew evacuation can take place instantaneously via the stand-by module which always has the seats in place and is ready to depart. The crew can seal themselves off from the station, engage the LEMBEC's autonomous life support system, and then detach from the station. In the case that the LEMBEC is unattainable by the crew, due to a closed off Space Station module, the automated LEMBEC would be flown to an accessible Space Station module. The ability of the LEMBEC to support 8 people for 24 hours will allow the module to orbit Earth while ground control assesses and attempts to correct the problem which necessitated departure. If the Space Station is deemed habitable, then the crew can return to the station, but it if the problem cannot be corrected, then the crew will return to Earth. If the module returns to the station, the ECLS will be resupplied from Space Station stocks.

A partial crew evacuation will occur when one or several crew members become ill or injured and need medical attention on the ground. Those injured will leave, possibly with an uninjured crew member as an aide, while the remainder of the crew will stay on the station and rely on the one remaining module as their escape route. If time permits, the module used in the partial evacuation will be loaded with materials to be returned to Earth. This will allow the remaining module to remain an additional 90 days to preserve the logistic resupply timeline. If waste materials are not able to be returned with injured crew members, then either the ground refurbishing for the next launch will have to be sped up, or a Space Shuttle mission will have to be rescheduled for resupplying and removing materials from the Space Station.

Costing

Cost estimates for development and production of LEMBEC were made using parametric cost analysis, with mass as the principle component, at both system and subsystem level. This cost analysis is based on Rockwell International Cost Estimating Relationships (CER's). The breakdown of the subsytem's coefficients, parameter, and scaling exponent are shown in Appendix A. The Percent Design, Design Complexity, Production Complexity, and Escalation Index, also shown in Appendix A, are completely estimated solely for cost estimations. Masses for the parameters are used from the Mass Budget section of this subsystem.

Rockwell International uses the following cost estimating methodology for hardware, shown in Figure 4.

- Cost estimating relationships basic form
- $\text{Cost} = A\,\text{WGT}_{ss}^{\ B}$

A = coefficient
B = scaling exponent
WGT = subsystem weight

- $\text{Cost}_{\text{DDTE}} = A\,\text{WGT}^{B} \begin{Bmatrix}\text{percent}\\ \text{new}\\ \text{design}\end{Bmatrix} \begin{Bmatrix}\text{design}\\ \text{complexity}\end{Bmatrix} \begin{Bmatrix}\text{escalation}\\ \text{index}\end{Bmatrix}$

- $\text{Cost} = A\,\text{WGT}^{B} \begin{Bmatrix}\text{production}\\ \text{complexity}\end{Bmatrix} \begin{Bmatrix}\text{escalation}\\ \text{index}\end{Bmatrix} \begin{Bmatrix}\text{quantity}\end{Bmatrix}$

Figure 4

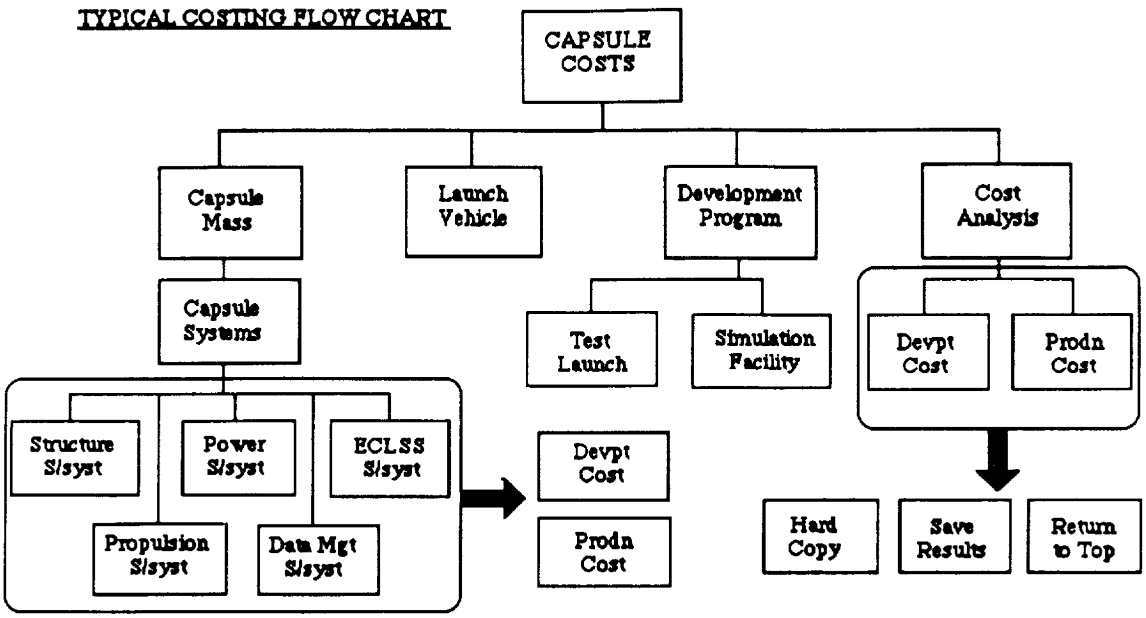


Figure 5

To find a total cost for one LEMBEC, the addition of several costs must be made. Those costs include Development and Production Costs, Launch Vehicle Costs, Development Progress Costs, and the cost of Cost Analysis itself. Figure 5 shows a typical costing flow. Do to the unavailability of all those costs, only an estimation to the cost of LEMBEC can be made at this time. The cost that is being proposed for one LEMBEC will be in the area of $1.9 Billion, or $7.9 Billion to produce four.

For a view from a costing standpoint for the number of vehicles selected, an example of two smaller vehicles launched every 45 days versus one larger vehicle every 90 days will be used. It would unrealistic to think that a smaller vehicle sent twice as often would have a cost equivalent to one sent every 90 days. The recurring cost per flight would have to be at least half that of one vehicle every 90 datys to break even. Additional vehicles would also be needed, since it takes over 45 days to refurbish one. From a logical costing standpoint, there would be unnecessary costs incurred if one vehicle con do the job in a 90 day interval.

Conclusion

The decision to have four eight-man vehicles launched every 90 days each can quickly be summed up by covering the major points in this subsystem. The decision came from the fact that: (1) eight men can fit in the sizing required by the logistics, (2) one launch vehicle can do the job, (3) the expectation for emergency use is not great, and (4) cost wise it is more sensible.

Works Cited

Hannigan, Russell J. and C.M. Hempsell. "Multi-Role Capsule System Description." JBIS 42 (1989): 69,79-81.

Hempsell, C.M. "Multi-Role Capsule -- An Introduction." JBIS 42 (1989): 53.

Hempsell, C.M. and R.C. Parkinson. "Multi-Role Capsule: Programme and Costs." JBIS 42 (1989): 92-94.

Neilson, John J. "Use of Expendable Launch Vehicles for OSSA Missions." Final Report SAIC Subcontracted No. 16-890046-96, 1988.

Environmental Control and Life Support

by David Schaefer

Introduction

The environmental control and life support subsystem (ECLS), as the name implies, is concerned with keeping the crew alive and healthy. The system is used whenever humans are in the module when it is detached from the space station. The ECLS of the LEMBEC is conceptually very similar to the one used on the Space Shuttle with the one major difference being the size of the system. These two life support systems are similar for three reason. First of all, the mission lengths are similar; the Shuttle's length is less than a week and the LEMBEC's is at most 24 hours. Secondly, the Space Shuttle's technology is currently available. Lastly, the Space Shuttle's technology is simple and proven. By modeling the LEMBEC's ECLS on the Space Shuttle's, a workable, safe, proven, and relatively inexpensive life support system was obtained.

The ECLS contains a number of requirements which must first be thoroughly presented to define precisely the problem to be solved. Of course, the overriding requirement is to keep the crew alive. To do this, the necessary consumables, such as oxygen and food, must be identified and the appropriate amounts determined. In addition, the storage vessels for the consumables must be determined. The wastes produced by the humans on board must also be identified and appropriate means for removing them provided. Another requirement involves keeping the temperature in the cabin at a comfortable level while also controlling the cabin's humidity. Furthermore, the required volume for the crew to live comfortably is required along with a layout of the crew in the module. The ECLS system is also required to provide some type of health care for sick or injured crew members and lastly, to detect and suppress any fires on board the module.

Environmental Set Points

A further definition of the problem involves a determination of environmental set points. These are ranges of values which define the composition of the environment. Subsequent sections of this ECLS section are, to a great extent, concerned with keeping the environment at these points.

Environmental Conditions

Parameter	Unit	Operational	Emergency
Total Pressure	$Nx10^3/m^2$	99.9-102.7	99.9-102.7
O_2 Partial Pressure	$Nx10^3/m^2$	9.5-23.1	15.8-23.7
CO_2 Partial Pressure	N/m^2	400 max	1600 max
Temperature	C	18.5-24.1	15.8-32.4
Ventilation	m/sec	.08-.20	.025-1.02

Humidity: 25-75%

ref:(Life Support for JEM)

A brief description of these various points will reveal the importance of the operational and emergency ranges.

The total pressure is kept in a range as close as possible to the terrestrial atmosphere at sea level. Humans can survive at lower pressures for a period of time, as they do in space suits at .3 atm, but a range close to 1 atm provides the most comfort.

The oxygen partial pressure is kept in a range to make the oxygen content of the air about 21% as it is on Earth. Excess oxygen can be toxic to the body causing muscle twitching and lung irritation. A shortage of oxygen can cause hypoxia.

The carbon dioxide content should be about .5% of the cabin's air. It can increase to 2%, but head aches and nausea may occur. The absence of CO_2 has no adverse effects on the body.

The temperature range is maintained to provide a shirt sleeve working environment. Too low of a temperature can give shivering and impared sensory functions while too high of a temperature can induce heat stroke. The humidity is tied in to temperature to give the proper vaporing of sweat.

The air flow velocity is important to ensure the proper heat transfer from the body's surface to the air. It is also important for keeping the air mixed and to avoid stagnant pockets in the cabin.

<u>Volume and Layout of the Crew</u>

To determine the volume required for an eight man crew, as we have, equations exist which were obtained from empirical data and computer generated polynomial fitting algorithms. These volume requirements only serve as "rule-of-thumb" measurements since they don't take into consideration such factors as crew training and motivation or, for our case, the extraordinary situation of an emergency escape.

The minimum volume required is given by:

$$V_{min} = -(0.0040)x^2 + (1.4219)x + 81.3071$$

where x = days and V_{min} is in ft^3/man-day. For our requirement of an 8 man crew being in the module for a maximum of one day, $V_{min} = 661.8\ ft^3 = 18.74\ m^3$.

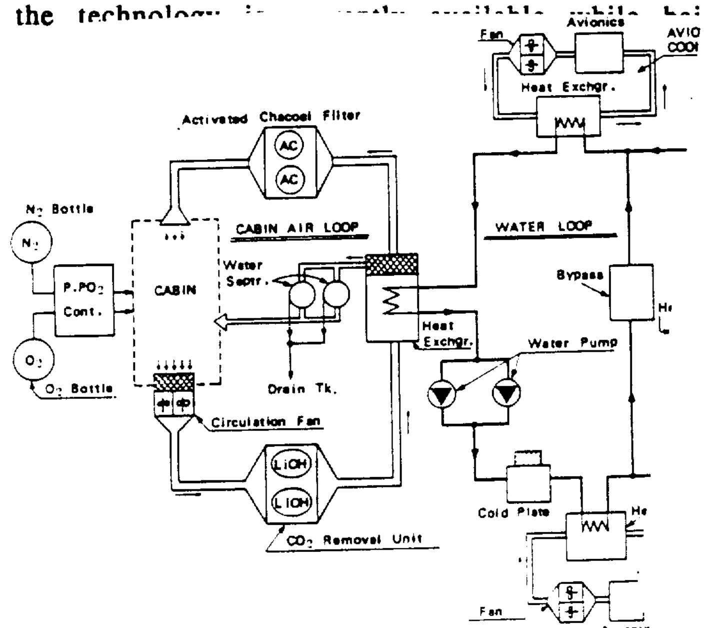
This required minimum volume is considerably less than the up volume calculated by the mission planner (see Mission Management Subsystem). So, the volume of the LEMBEC was driven by the logistic resupply needs rather than the V_{min} needs of an 8 man crew. The designed volume of $V = 69.06\ m^3$ is more than spacious enough for the crew.

To see the layout of the crew, refer to the Structure Subsystem. Whenever humans are to use the LEMBEC, detachable seats which are stored in the Space Station will be attached. These seats are designed for quick and easy attaching, and their storage in the Space Station helps conserve precious volume and mass on the LEMBEC. Two fold down beds are also provided as permanent features of the LEMBEC to be used in emergencies by injured evacuees.

Life Support System

The feature that distinguishes this life support system from the Space Station's closed loop system is that the LEMBEC uses an open loop. A closed loop system is concerned with recycling the consumables for reuse. For long duration missions such as the Space Station's, the increased initial complexity and cost as opposed to an opened loop system is justified because of the long term savings in resupply needs. The LEMBEC's life support system, however, is only intended to be used for a maximum of one day for 8 people. Therefore, an open loop is desired to give a simple, reliable, and less costly system.

The following diagram shows the various flow loops and the major components for the LEMBEC's ECLS. This schematic represents the Space Shuttle's ECLS, but LEMBEC's is so similar as to be identical. Again, it is stressed that the similarities exist because the technology is currently available while bei[illegible]le and proven.



The way this system works is that the consumables, oxygen and nitrogen, are provided by supply tanks to the cabin. Fans circulate the air to the CO_2 removal unit and then to the heat exchanger which controls both the temperature and the humidity. The air is then passed through an activated charcoal filter before reentering the cabin. The waste heat from the cabin and the avionics is transferred at heat exchangers to a water loop powered by pumps. The heat in the water loop is then transferred to a freon loop, again powered by pumps, for radiation to space (see Structure Subsystem). The waste heat isn't transferred directly from the air loop to the freon loop for two reasons. First of all, it is safer to have a water loop in case of a leak to prevent freon from entering the cabin. Secondly, the differences in the heat transfer properties makes it more efficient to go from air to water to freon instead of from air to freon directly.

Consumables

Following is a table of the consumables, the rate of use, the amount needed for 8 men for one day, and the amount we will carry for safety reasons. The N_2 is necessary to maintain an Earth normal partial pressure for oxygen while the LiOH is used in the CO_2 removal unit. The other 3 are self explanatory.

Consumables

Consumable	Rate of Use	Amount Needed(kg)	Amount Carried(kg)
N_2	3.6 kg/day-veh	3.6	5.4
LiOH	1.36 kg/man-day	10.91	16.36
O_2	.836 kg/man-day	6.8	10.2
Water	3.09 kg/man-day	24.73	28
Food	1.34 kg/man-day	9.09	9.09

ref:(Proceedings of Sixteenth ICES Conference pp. 311,312,325)

The water will be contained in 28 one liter plastic bottles; this gives a few extra liters than what is required. The food will be stored in plastic bags and will be dehydrated because it allows compact storage and has a long shelf life. The LiOH will be carried in three 5.45 kg CO_2 removal units. The oxygen and nitrogen will be carried in pressure tanks. As with the LiOH, 3 separate tanks will be used for O_2 and N_2. Again, as with the LiOH units, any two tanks provide enough O_2 or N_2 for the mission length; the third tank is for safety. The tank calculations are provided in the appendix. The LEMBEC is using spherical tanks with dimensions for O_2 of r= .1465m , mass= 6.52 kg and for N_2 of r= .1239m , mass= 3.687 kg. For the placement of these components see the Structure Subsystem.

Wastes

Following is a table of the wastes produced, the rate of production, and the total amount produced by an 8 man crew in one day.

Wastes

Waste	Production Rate	Total
Metabolic Heat	1270 W	1.097×10^5 kJ
Electronic Heat	3809 W	3.292×10^5 kJ
H_2O Respirated		
H_2O Perspirated	1.83 kg/man-day	14.62 kg
CO_2	1 kg/man-day	8 kg
Urine	1.5 kg/man-day	12 kg
Feces	.14 Kg/man-day	1.09 kg
Unused Food and Packaging		1 kg max

ref:(Proceedings of the Sixteenth ICES Conference pp. 312,318,325,775,821)

The heat is removed at the heat exchangers and radiated to space. At the same time, the excess water vapor is condensed and collected by centrifuge. The urine is collected by a tubing system similar to the Space Shuttle's and stored in bags. The feces is collected in plastic bags with adhesive linings. Privacy screens are provided for crew members when performing excretory functions. The unused food and packaging will be collected in a small, fold-out trash bin.

The Cabin Air Loop

The cabin air loop is concerned with circulating the cabin's air to remove CO_2, excess heat, and excess water vapor. Before returning to the cabin, the air passes through an activated carbon filter to purify it.

The carbon dioxide is removed via LiOH canisters. 3 canisters exist on separate lines for redundancy and safety. Any 2 canisters alone would suffice to remove the anticipated CO_2. LiOH removes CO_2 by the following process:

$$2\ LiOH + CO_2 \text{ --- } LiCO_3 + H2O_{(g)} + Q$$

The $LiCO_3$ is stored in the canisters while the heat exchanger removes the heat and the water vapor. Other means exist for removing CO_2, such as molecular sieves and amine granules, but they are best for closed loop ECLS systems while LiOH, because of its simplicity and smaller size, is best for an open loop ECLS.

The temperature and humidity in the cabin is controlled by removing excess heat and water vapor at a heat exchanger (see Appendix - Life Support). The avionics are also air cooled at a heat exchanger. As was already described, a water and freon loop are used to transfer the heat to space. Designing a heat exchanger and a centrifuge for collecting the water poses a major technical problem area for me because of the lack of gravity. My design assumes gravity, so the result isn't precise, but it is sufficient for the conceptual design as the RFP requires. A real heat exchanger and centrifuge system does exist for the Space Shuttle, however, so I know that my proposal is feasible.

Sensors and Control

LEMBEC's ECLS is continuously monitored and controlled by an array of sensors controlled by computer software. Using the sensor data, the computer controls the various flow rates in the loops and such things as the cabin's heater to maintain the environmental set points in the operational range. Control and monitoring involves real-time data processing, system fault tolerance and redundancy management, caution and warning, and health monitoring. To control this system, LEMBEC will rely to a large extent on artificial intelligence to make the system more reliable and to free the crew from monitoring. The Space Station itself contains over 1000 sensors and 500 actuators. LEMBEC, of course, will contain a considerably lesser amount, but this conceptual design has no means of accurately estimating the required number.

Redundancy

To make the LEMBEC's ECLS as fail safe as possible, multiple, redundant systems are employed. The critical consumables O_2, N_2, LiOH are each contained in three separate units any two of which can fulfill the mission requirements. Each pressure tank also contains multiple valves and regulators to help ensure proper operation. Multiple pumps are also used in each flow loop to ensure against one failing, and lastly, multiple sensors are used to ensure adequate monitoring.

Except for the control and monitoring system, the ECLS is a relatively low-tech system. The components are simply tanks, valves, pumps and fans, and ducting mostly. A multiple number of each mechanical part should ensure safe operation.

Medical Equipment

Most of the medical supplies needed for injured or sick crew members will be taken from the Space Station's Health Maintenance Facility (HMF). The HMF is designed to provide critical care for injured crew members for up to 28 days, so the equipment provided is substantial (see Proceedings of 16th ICES Conference: pp. 113-118 for complete list). Space on the LEMBEC will be allocated to hold any of the special electronic equipment that might be taken from the HMF. The LEMBEC will always carry a first-aid kit to treat injuries if there is no HMF equipment on board.

Fire Control

Designing a fire sensing and control system poses my greatest technical problem area, especially when dealing with fires inside the equipment without easy access to humans. However, since the ECLS is modeled after the Space Shuttle's, I know that this problem has been adequately solved with technologies currently available.

The LEMBEC's proposed fire control system involves using smoke/gas and temperature sensors to indicate the presence of fires. Fires in the equipment will be extinguished by closing the section containing the fire and spraying in a fire retarder. The section will then be vented to remove the byproducts. Internal fires will be fought automatically by the control system. Fires in the cabin will be fought with extinguishers containing foam or CO_2. Whenever fires occur, the astronauts will don oxygen masks to avoid inhaling the fire's fumes.

Appendix-Life Support

Spherical Pressure Tanks

Constants

σ_y= 300000 psi = 2.068×10^9 N/m^2 (for steel)

R_{O2}= 48.28 ft-lbf/lbm-R R_{N2}= 55.15 ft-lbf/lbm-R

SF= 2 Dens= .289 lbm/in^3

p= 3000 psi = 2.068×10^7 N/m^2 T= 80 °F

Thin wall theory since t/r <= .1
ref:(Introduction to Aerospace Structural Analysis p. 484)

$r = r_i + t/2$

$\sigma_L = pr/2t$

Vol= $(4\pi r^3)/3$

Analysis for oxygen (nitrogen analysis is the same except for $mass_{N2}$ and R_{N2})

3 tanks with 3.4 kg O_2 each

Vol_{O2} = nRT/p = .451 ft^3 = .0128 m^3

$r_i = (3V/4\pi)^{1/2}$ = 5.71 in = 14.50 cm

$t = pr_i/(2(\sigma_y/SF)-p/2)$ = .0574 in = 1.46 mm

$mass_{tank}$ = $dens(vol_{steel})$ = $dens(4/3)\pi((r_i+t)^3-r_i^3)$
= 6.865 lb
= 3.12 kg

Totals- 3 O_2 tanks: r = .1465 m
mass = 6.52 kg
3 N_2 tanks: r = .1239 m
mass = 3.69 kg

Trade on Tank Shape (sphere/cylinder) vs. Mass

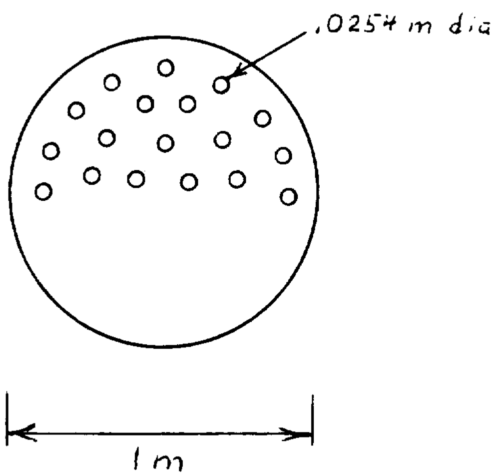
A simple trade study revealed that the mass of a spherical pressure vessel is less than that of a cylindrical vessel with hemispherical ends. This is so because the stress in the cylindrical walls is 2 times that in the spherical walls, so more steel is needed. Cylindrical tanks are easier to place more compactly, but our spherical tanks will be clustered to conserve volume and the space in between tanks will be used for additional life support components.

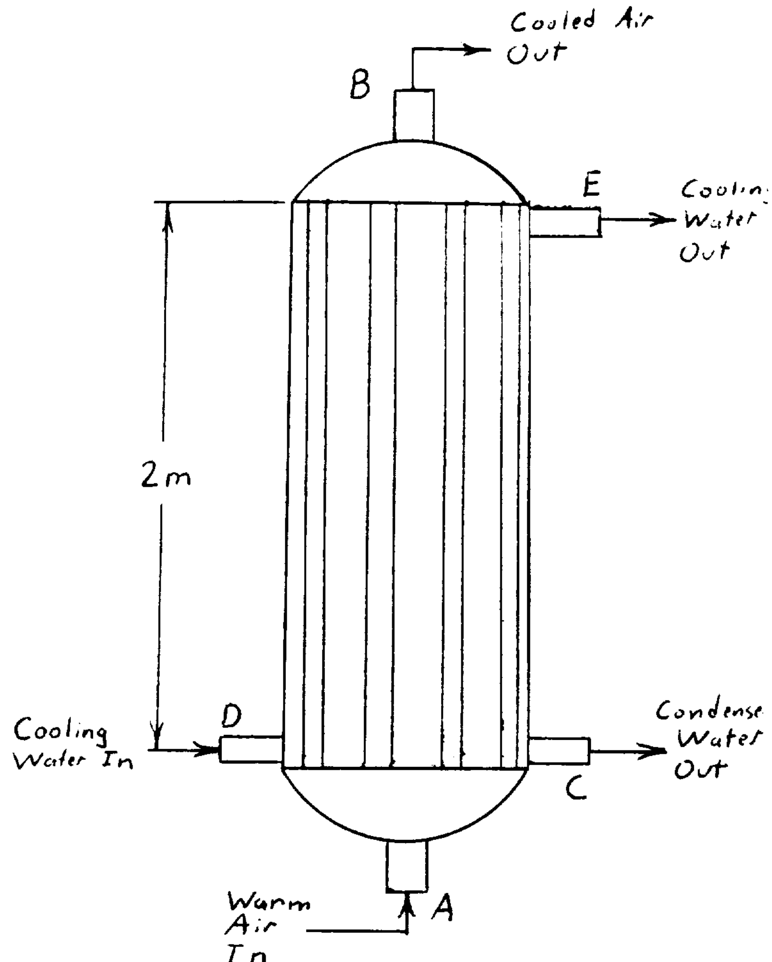
A graph usually accompanies a trade study, but this trade doesn't lend itself to a graph. A simple comparison of numbers will show the mass savings. For 3 spherical O_2 tanks and 3 spherical N_2 tanks, the total mass is 30.63 kg. Using a value of $r = 7.62$ cm consistent with thin wall theory for a cylindrical tank and using 3 O_2 and 3 N_2 tanks, the total mass is 47.24 kg. A mass savings of 16.61 kg is realized by using spherical tanks. Different r values for cylindrical tanks can be used, but the mass stays in the range of 47.24 kg.

Appendix - Summary of Heat Exchanger/Condenser

(5 pages of detailed calculations available upon request. Length restrictions by management prevented their inclusion.)

Total of 862 tubes





Summary of Stream Properties

Stream #	Temperature	Pressure	Components	Flow Rates (Volumetr
A	24.1°C	76300 $\frac{N}{m^2}$	Air, H_2O	.14 $\frac{m}{s}$ (veloci
B	18.5°C	74861 $\frac{N}{m^2}$	Air, H_2O	
C	18.5°C	74861 $\frac{N}{m^2}$	H_2O	45.28 $\frac{ml}{hr}$
D	10.0°C	76300 $\frac{N}{m^2}$	H_2O	111.80 $\frac{m^3}{hr}$
E	20.0°C	76300 $\frac{N}{m^2}$	H_2O	111.80 $\frac{m^3}{hr}$

Bibliography - ECLS

Allen, David H., Haisler, Walter E., Introduction to Aerospace Structural Analysis, John Wiley and Sons Inc., New York, 1985

Finch, John D., Consultant (Heat transfer analyst), Urbana, Illinois

Gordon, T.J., Germis, R.L., "Critical Engineering Problems of Space Stations", Advanced Space Stations and Planetary Systems, McDonnell-Douglas Astronautics Company, Huntington Beach, California, 1968

Kanda, Fujimoto, Hattori, Shimizu, Matsumiya, "Life Support System Study of Japanese Experimental Module of Space Station"

Proceedings of the Sixteenth ICES Conference: P-177, Aerospace Environmental Systems, San Diego, California, July 14-16, 1986

Tumulty, William T., "Space Station Technology", AIAA-NASA Space Systems Technology Conference, June 5-7, 1984, NASA Headquarters, Washington, D.C.

Vallerani, E., "Manned Pressurized Elements for Space Station Applications", Earth-orientated Applications of Space Technology, Vol. 6, No. 1, pp. 25-34, 1986

Ward, Timothy, Habitability Considerations for a Space Operation Center, Thesis, Urbana, Illinois, 1983

The Power and Propulsion Systems of L.E.M.B.E.C.

by Edward J. Goletz

Propulsion System

Requirements

The first requirement of the propulsion system is to provide the required velocity changes that will be needed for the various missions that the LEMBEC will be called on to perform. These will include the ΔV needed to reach the space station from the orbit the ELV leaves the capsule in, the ΔV needed to de-orbit from the the space station, and the ΔV needed to reach the unmanned platforms that are serviced from the space station.

Other requirements of the propulsion system are that the system must be able to remain unused or to have little use for long periods of time, on the order of six to nine months. Also, the propulsion system must be as safe as possible in order to avoid damage to crew and equipment and the cost of the system must be kept to a minimum. In addition, off the shelf technology should be used when possible and no technology can be used if it is not expected to be available by 1994. Simplicity and reliability are to be stresses.

From the appendix of this section, we can see that the ΔV needed to reach the space station is 0.141 km/sec and the ΔV needed to de-orbit is 0.1093 km/sec for the space station in a 430 km altitude. However, the ΔV needed to reach the polar orbiting platform and return is a ridiculously huge amount: 17.6 km/sec. If we were to use an engine with a specific impulse of 500 seconds, the ΔV needed for a one way trip to the platform would be 8.8 km/sec, which would require a fuel mass of 17,000 kg for a 20,000 kg vehicle. The mass of fuel needed for a round trip for a 20,000 kg vehicle would be 19,500 kg!

From these calculations it is quite clear that it is not reasonable to expect the LEMBEC to rendezvous with the polar orbiting platform. It would probably be much cheaper to simply support the platform from the ground rather than to design some huge propulsion module for the LEMBEC to enable it to support the platform from the station.

This leaves us with the ΔV needed for reaching the space station and the ΔV needed to de-orbit, which sum to approximately 2.5 km/sec. We shall allow an extra 0.5 km/sec for any contingencies, and this leaves us with a ΔV capacity needed of 0.30 km/sec.

Method of attack

The first question to answer for the propulsion system is what fuel to use. Electric propulsion is out because of the extremely low thrust associated with it. Also, as the engines must be throttlable, this leaves us with liquid propellants. Of the liquid propellants, only the ones storable at room temperature have been considered, as the propellants will have to remain in their tanks for months at a time. Narrowing down the selection to those that have higher specific impulses and to those with a large database of information available, we have selected the nitrogen tetroxide/monomethylhydrazine combination. Hydrazine also looked desirable, but we could find little information on engines using this fuel. There was one reference that showed some specific impulses for various fuels with beryllium, Be, added to them. These fuels had a much higher specific impulse, but no other information could be found about them. If this is undeveloped technology, it might be worth looking into the development costs.

For the engines, we have chosen a reusable design in order to reduce the overall cost of the system over its expected lifetime. There are many types of engines currently using NTO/MMH, and reviewing these has been helpful in estimating the thrust and size of the engines.

System description

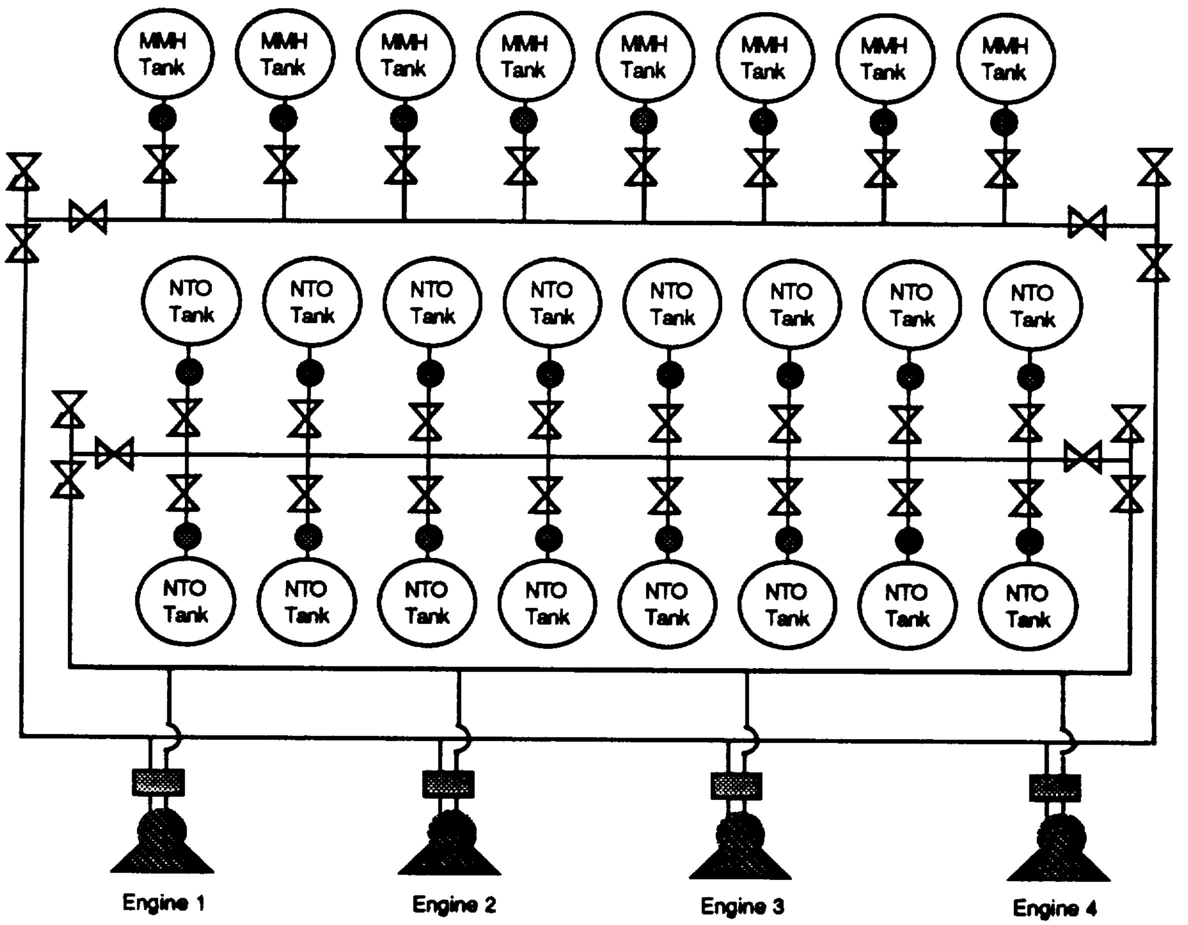
In order to reduce costs, the engines for LEMBEC will be reusable. Also, because the LEMBEC might have to remain in orbit and inactive for several months, the fuel used must be storable for long periods of time. The fuel/oxidizer chosen for LEMBEC is Monomethylhydrazine (MMH) and Nitrogen Tetroxide (N_2O_4) because these are both storable at room temperatures and they give a high specific impulse. Monomethylhydrazine was chosen over Hydrazine because the database on this fuel is much greater, as there are many engines already using this fuel/oxidizer combination.

The theoretical specific impulse of of this fuel/oxidizer combination is 343.8 seconds, assuming a chamber pressure of 1000 psia, vacuum expansion, and an expansion ratio of 40. If we assume a Isp of 340 seconds, the mass of fuel/oxidizer needed for a 0.30 km/sec ΔV for a 20,000 kg spacecraft it 1720 kg, which we have made the fuel capacity for the LEMBEC.

We had data on the the vapor pressure of MMH up to 428 K (155° C) and on nitrogen tetroxide up to 328 K (55° C). Therefore, the tanks have been sized such that they can withstand the vapor pressure (with a safety factor of 3) of pressures up to these values. As the tanks should always be at room temperatures, these limits should not present any problems. The boiling point of each chemical is 361 K and 294 K respectively. There are eight fuel tanks and sixteen oxidizer tanks in the vehicle, each holding an appropriate fraction of the fuel/oxidizer load. The fuel and oxidizer have been spit up into the 32 tanks for the logistics of placing them and for the redundancy involved. Each tank has its own pump and valve. With each tank having its own pump, the failure of one pump will have little effect on the overall pressure of the fuel or oxidizer bus.

There will be four engines space symmetrically around the top hatch. Each engine is a pump-fed, regeneratively cooled engine. Based on the mass and performance of the XLR-132 engine made by Rocketdyne and scaling up, it is estimated that each engine will mass approximately 75 to 80 kg and have a thrust of approximately 25,000 N. With a combined thrust of 100,000 N and a total mass of 20,000 kg, the LEMBEC should have an acceleration of 0.5 G. Also, based upon a total ΔV capacity of 0.30 km/sec, the engines should have a total burn time (before fuel runs out) of 60 seconds. Each engine also has a shut off valve connecting to the fuel and oxidizer bus so that any individual engine can be shut off while the rest of the engines continue to work. If one engine should fail, that engine and the engine opposite it could then be shut off while the remaining two engines would still provide thrust without generating any moments.

Below is a diagram of the propulsion system.



Power System

Requirements

The requirements of the power system are to provide power to all of the craft's systems, to supply a constant voltage and frequency (if AC voltage), to protect the craft systems from surges and spikes, and to be as simple and inexpensive as possible. The power system should also be as reliable and robust as possible and should observe the same technology restrictions as listed for the propulsion system above.

Before the power system is designed, we need to have an estimate of the power requirements of the crafts systems. Below is such an estimate.

System Power Requirements

Communications	300 watts
ECLSS	300
Thermal control	200
Other	100
total	900
margin	100
TOTAL	1,000 watts

A study done within British Aerospace for a multi-role capsule and published within *JBIS* was helpful in arriving at this estimate. It is likely that at times communication, ECLSS, and thermal control will all need maximum power at the same time, so the power system should be able to provide 1,000 watts of power continuously.

Method of Attack

The worst case scenario (in terms of power) that the LEMBEC will experience, according to mission planning, will be that eight space station personnel will leave the space station inside the LEMBEC, wait near the space station for 24 hours, and then de-orbit. If this happens, then the power system will need to supply maximum power continuously for 24 hours, alternating between 45 minutes in sun and 45 minutes in eclipse.

We do not want batteries alone to supply this power, as the weight of the batteries would be prohibitive. Also, we can not rely solely on photovoltaic cells, as half of the time the LEMBEC would be in the eclipse of the earth. Fuel cells are an attractive option, but the cryogenic storage of the fuel for months at a time precludes their use. Radioactive thermal sources do not provide enough power, and nuclear reactors are too heavy and politically questionable.

This leaves us with a photovoltaic-battery combination. Both photovoltaic cells and batteries have been used extensively in space and both can be stored for long periods of time.

For the batteries, nickel cadmium batteries will be used (even though nickel hydrogen batteries are used on the space station) for safety, cost, and technical risk considerations. For the solar arrays, there will be one array on either side of the capsule, attached near the docking adapter. Each array will be retractable for storage, radiation protection, and protection during maneuvers. Each array will have actuators that will allow them to be rotated towards the sun. There are some types of solar cells that offer a greater efficiency than the standard space cells used in this design; however, these high efficiency cells cannot deviate from direct sunlight by more than a few degrees, or they cease to generate power. This liability makes them unsuitable for our purposes.

The power bus could be either a regulated or unregulated type. An unregulated power bus would, roughly, have the solar arrays connected directly in parallel with the battery charger. This type of power bus is simpler, but as it is unregulated, it cannot guarantee a stable voltage. We have chosen a regulated power bus. Altho this system is more complex, the added risk is very low, while the benefits of a stable power supply are very great.

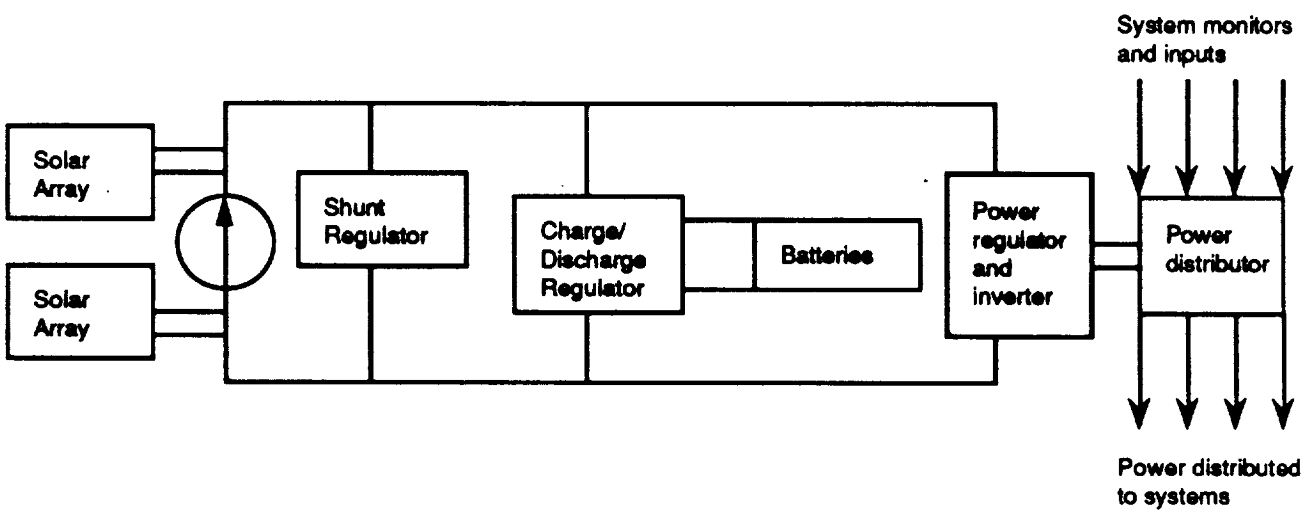
Another question needing answering is at which voltage and frequency should the power system operate at. We have chosen 208 volts, 20 khz, as this matches the power systems of the space station and unmanned platforms, and so adds additional redundancy to the systems (both the space station's system and ours.)

System Description

The power system will consist of 75 30 w-hr nickel cadmium storage cells that can provide 1,000 watts for up to 1 hour with a depth of discharge of 45%. While exposed to sunlight, 3,000 watts of power will be provided by two solar arrays. One third of that power is allocated to the craft's systems, and the remaining 2,000 watts are used for recharging the batteries. At this rate, the batteries can be recharged from a depth of discharge of 45% in only one half hour. The craft will actually be in sunlight for a maximum of 45 minutes and in eclipse for a like amount. This one half hour recharge time and one hour power capacity of the batteries is a type of built in safety factor for the power system.

The batteries mass of total of 84 kg and the each of the two solar arrays measure 12 m^2 and mass 20.5 kg. The power provided by the arrays and storage cells are in parallel with a shunt regulator and a power regulator/inverter. The power regulator/inverter converts the DC power of the arrays/batteries into 208v, 20 khz power for the power distributor. All systems using power receive their power from the power regulator. This subsystem monitors the use of power and it receives and evaluates requests for power from the various systems and it grants power accordingly.

Below is a diagram of the power system.



Appendix of calculations

ΔV needed to reach space station from the orbit the ELV leaves the LEMBEC in

The ELV under consideration will be able to put the LEMBEC in an orbit 185 km above the earth at 28.5 degrees inclination. The space station might be in an orbit anywhere from 290 km to 430 km at the same inclination. Therefore, the propulsion system must be able to take the LEMBEC from a 185 km altitude to a 430 km altitude. We will assume a Hohmann Transfer will be used for this maneuver.

$$\Delta V_p = \sqrt{\frac{\mu}{R}}\left(\sqrt{\frac{\frac{2R_a}{R_p}}{1+\frac{R_a}{R_p}}} - 1\right)$$

$$= \sqrt{\frac{3.96 \times 10^5 \text{ km3/sec2}}{185\text{km} + 6378\text{km}}}\left(\sqrt{\frac{\frac{2(6378\text{km} + 430\text{km})}{6378\text{km} + 185\text{km}}}{1+\frac{6378\text{km} + 430\text{km}}{6378\text{km} + 185\text{km}}}} - 1\right)$$

$$= 0.0708 \text{ km/sec}$$

$$\Delta V_a = \sqrt{\frac{\mu}{R}}\left(1 - \sqrt{\frac{2}{1+\frac{R_a}{R_p}}}\right)$$

$$= \sqrt{\frac{3.96 \times 10^5 \text{ km3/sec2}}{430\text{km} + 6378\text{km}}}\left(1 - \sqrt{\frac{2}{1+\frac{6378\text{km} + 430\text{km}}{6378\text{km} + 185\text{km}}}}\right)$$

$$= 0.0702 \text{ km/sec}$$

$$\Delta V_{tot} = 0.0708 \text{ km/sec} + 0.0702 \text{ km/sec} = \boxed{0.141 \text{ km/sec}}$$

ΔV needed to de-orbit

The calculations showing what the ΔV needed to de-orbit are shown in the section on reentry and recovery. That section tell us that in order to de-orbit from a 290 km orbit, we will need a ΔV of 0.0860 km/sec and to de-orbit from a 430 km orbit we need a ΔV of 0.1093 km/sec.

ΔV needed to reach the unmanned platforms

Of the unmanned platforms, there will be one or more platforms in the same orbit as the space station and there will be one polar orbiting platform. The ΔV needed to reach the platforms near the space station will be negligible. The ΔV needed to reach the polar orbiting platform, however, is very great and will be the driver for the whole propulsion system (if such a maneuver is required; see below). Below are the ΔV calculations.

The space station will be in a circular orbit anywhere from 290 km to 430 km altitude and at an inclination of 28.5 degrees. The polar orbiting platform, however, will be in an orbit of orbit to 824 km at 98.7 degrees. We will assume the space station to be in a 290 km orbit, as this is the worst case scenario.

One type of orbit transfer that could be used would be a Hohmann transfer from the 290 km orbit to the 824 km orbit and then to do a one burn plane change to bring the LEMBEC into an orbit at the same altitude and inclination as the polar orbiting platform.

$$\Delta V_p = \sqrt{\frac{\mu}{R}}\left(\sqrt{\frac{\frac{2R_a}{R_p}}{1+\frac{R_a}{R_p}}} - 1\right)$$

$$= \sqrt{\frac{3.96 \times 10^5 \text{ km3/sec2}}{290\text{km} + 6378\text{km}}}\left(\sqrt{\frac{\frac{2(6378\text{km} + 824\text{km})}{6378\text{km} + 290\text{km}}}{1+\frac{6378\text{km} + 824\text{km}}{6378\text{km} + 290\text{km}}}} - 1\right)$$

$$= 0.147 \text{ km/sec}$$

$$\Delta V_a = \sqrt{\frac{\mu}{R}}\left(1 - \sqrt{\frac{2}{1+\frac{R_a}{R_p}}}\right)$$

$$= \sqrt{\frac{3.96 \times 10^5 \text{ km3/sec2}}{824\text{km} + 6378\text{km}}}\left(1 - \sqrt{\frac{2}{1+\frac{6378\text{km} + 824\text{km}}{6378\text{km} + 290\text{km}}}}\right)$$

$$= 0.144 \text{ km/sec}$$

$$\Delta V_{transfer} = 0.147 \text{ km/sec} + 0.144 \text{ km/sec} = 0.291 \text{ km/sec}$$

The ΔV for the plane change, however, is much greater than this.

$$\Delta V_{plane\ change} = 2V_c \sin\left(\frac{\phi}{2}\right)$$

$$= 2\sqrt{\frac{3.96 \times 10^5 \text{ km3/sec2}}{824\text{km} + 6378\text{km}}} \sin\left(\frac{98.7 - 28.5}{2}\right)$$

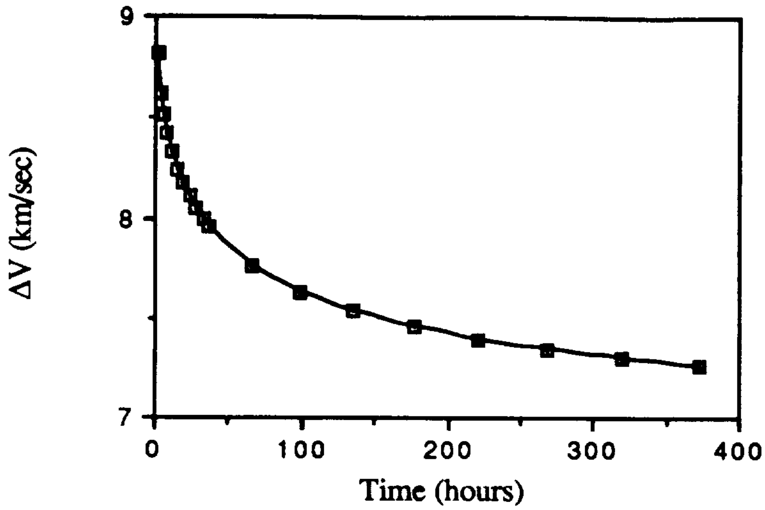
$$= 8.53 \text{ km/sec}$$

$$\Delta V_{total} = 0.291 \text{ km/sec} + 8.53 \text{ km/sec}$$

$$= \boxed{8.82 \text{ km/sec}}$$

However, the one burn plane change is not the most economical plane change maneuver. The three impulse transfer is always more efficient than the one impulse maneuver, and maximum efficiency is achieved with the theoretical bi-parabolic transfer. The ΔV and transfer times for various three burn plane changes for this orbit change has been calculated on a spreadsheet for many different manoeuvres. Below are the results in graphical form.

Time and ΔV needed to reach polar orbiting platform



This shows the ΔV and transfer time for a one way trip to the polar orbiting platform. It can be seen from the data that even if the orbit change maneuver is extended to an entire week, the resulting savings in ΔV is only around 1.5 km/sec. In order to achieve this savings in ΔV, the apogee altitude must be several hundred thousand kilometers! Such a transfer orbit is undesirable because of the long transfer time, the passing through the Van Allen radiation belts, and because at such a great distance from the earth, the orbit might be perturbed by outside influences. As the LEMBEC is also an emergency crew rescue system, we would not want the LEMBEC away from the space station for more than a few hours at most. We will therefore use the ΔV for the one impulse plane change when calculating the mass ratios needed to reach the polar orbiting platform, as the other three impulse plane changes are either only marginally more efficient or undesirable for the reasons mentioned above. We can thus conclude that the ΔV needed to reach the polar orbiting platform will be 8.8 km/sec and a like amount to retürn, for a total ΔV of 17.6 km/sec.

$\Delta V_{\text{polar platform, round trip}} = \boxed{17.6 \text{ km/sec}}$

<u>Mass ratio needed to reach platform</u>

$$\Delta V = (\text{Isp})(g)\text{In}\left(\frac{m_1}{m_2}\right)$$

$$8800 \text{ m/sec} = (500 \text{ sec})(9.8 \text{ m/sec2})\text{In}\left(\frac{m_1}{m_2}\right)$$

$$\frac{m_1}{m_2} = 6.025$$

or if m_1 = 20,000 kg, mass of fuel needed is 17,000 kg!

<u>Mass of fuel needed to reach platform and return.</u>

$$\Delta V = (Isp)(g)\ln(\frac{m_1}{m_2})$$

$$17600 \text{ m/sec} = (500)(9.8)\ln(\frac{m_1}{m_2})$$

$$\frac{m_1}{m_2} = 36.30$$

or if m_1 = 20,000kg, mass of fuel needed is 19,500 kg!

<u>Fuel needed for 0.30 km/sec ΔV capacity</u>

$$\Delta V = (Isp)(g)\ln(\frac{m_1}{m_2})$$

$$300 \text{ m/sec} = (340 \text{ sec})(9.8 \text{ m/sec})\ln(\frac{m_1}{m_2})$$

$$\frac{m_1}{m_2} = 1.094$$

or if m_1 = 20,000 kg, fuel/oxidizer needed is $\boxed{1722 \text{ kg}}$

ratio of oxidizer to fuel is 2.37

$(\frac{1}{3.37})(1722 \text{ kg}) = 511 \text{ kg}$ fuel

$(\frac{2.37}{3.37})(1722 \text{ kg}) = 1210 \text{ kg}$ oxidizer

fuel/oxidizer density is 1.20 g/cm^3

$(1{,}722{,}000 \text{ g})(cm^3/1.2 \text{ g}) = 1{,}435{,}000 \text{ cm}^3$ fuel/oxidizer = 1.435 m^3 fuel/oxidizer

ratio of oxidizer to fuel is 2.37

$(\frac{1}{3.37})(1{,}435{,}000 \text{ cm}^3) = 452{,}800 \text{ cm}^3$ fuel

$(\frac{2.37}{3.37})(1{,}435{,}000 \text{ cm}^3) = 1{,}009{,}000 \text{ cm}^3$ oxidizer

<u>Size and mass of fuel and oxidizer tanks</u>

example calculation for MMH

Assume spherical fuel tanks made of 5Cr-Mo-V steel.

vapor pressure = (0.689 Mpa)
=(689,000 pa)(1 psi/6895 pa) = (100.0 psi)

$$\text{volume(for each of 8 tanks)} = \frac{425820\ \text{cm}^3}{8} = 53230\ \text{cm}^3$$

$$r_{\text{inside sphere}} = \sqrt[3]{(53230\ \text{cm}^3)\left(\frac{0.061024\ \text{in}^3}{1\ \text{cm}^3}\right)\left(\frac{3}{4\pi}\right)} = 9.19\ \text{in}^3$$

yield strength = 200,000 psi

assuming axial stress is greater than tangential stress

$$\text{thickness of steel} = \sqrt{\frac{P\ r_i}{\text{yld str} + P/2}} = \sqrt{\frac{(100.0\ \text{psi})\ (9.19\ \text{in})}{(200{,}000\ \text{psi}) + \left(\frac{200000\ \text{psi}}{2}\right)}} = 0.00458\ \text{in}$$

$$\text{Vol steel} = \frac{4\pi}{3}((r_i + t)^3 - r_i^3) = 4.84\ \text{in}^3$$

density of steel = 0.28 lbm/in^3

$$\text{mass of steel} = \text{(density)(volume)} = (0.28\ \text{lbm/in}^3)(4.84\ \text{in}^3) = 1.36\ \text{lbm}\left(\frac{1\ \text{kg}}{2.2046\ \text{lbm}}\right)$$

$$= \boxed{0.62\ \text{kg per tank}}$$

A similar analysis for 16 oxidizer tanks shows that the radius of each tank will be 24.6 cm and that the mass of each tank will be 0.44 kg.

Burn time and acceleration of vehicle

We will use the Newtonian equations of motion assuming a constant mass of 20,000 kg.

Velocity = (acceleration)(time)

V = (A)(T)

Force = (mass)(acceleration)

F = (M)(A)

or $A = \frac{F}{M}$

so $V = (\frac{F}{M})\,T$

or $T = (\frac{V\,M}{F})$

which gives us

$$A = \frac{100000\ N}{20000\ kg} = 0.5\ m/sec^2$$

and

$$T = \frac{(300\ m/sec)\ (20000\ kg)}{100000\ N} = 60\ seconds$$

Batteries and Solar Arrays

Assume nickel-cadmium batteries will be used, with a depth of discharge of 45% and a specific energy of 27 w-hr/kg. Also assume maximum time in eclipse will be 1 hour and maximum time in sunlight will be 0.5 hours. Of course, the arrays will actually be in eclipse and in sunlight for 0.75 hours each, but this assumption gives us a built in safety factor and accounts for times when the arrays might be turned away from the sun, such as during certain maneuvers.

$$\text{Batteries' stored energy} = \frac{(P_L)(T_E)}{DoD} = \frac{(1000\ w)(1\ hr)}{0.45} = 2{,}225\ w\text{-}hr$$

where 'P_L' is the power load, 'T_E' is the time in eclipse, and 'DoD' is depth of discharge of the batteries.

With a 30 w-hr maximum per cell, number of cells will be

$$\text{number of cells} = \frac{2225\ w\text{-}hr}{30\ w\text{-}hr/cell} = 75\ (30\ w\text{-}hr)\ cells$$

$$\text{mass of each cell} = \frac{30\ w\text{-}hr/cell}{27\ w\text{-}hr/kg} = 1.12\ kg\ per\ cell$$

for a total mass of (1.12 kg/cell)(75 cells) = 84 kg

$$\text{Solar Array power} = P_T = P_L + \frac{C}{N}V$$

$$\text{where } C = \frac{(P_L)(T_E)}{(DoD)(bus\ voltage)} = \frac{(1000\ w)(1\ hr)}{(D0.45)(\ 208\ v\ )} = 10.684\ w\text{-}hr/v$$

and N = Ts/DoD

$$P_T = \frac{(10.684 \text{ w-hr/v})(208 \text{ v})}{(0.5 \text{ hr})/(0.45)} = 3{,}000.0 \text{ watts}$$

The array size can be determined from the equation below.

$$P = (S)(Cr)(e)(A)(1- a(T \ - 25))$$

where 'P' is the power of the arrays, 'S' is the solar constant, 'Cr' is the packing factor, 'e' is the cell efficiency, 'A' is the area of the arrays, 'a' is the temperature degradation factor, and 'T' is the operating temperature in centigrade.

We will assume a packing factor of 0.90, a cell efficiency of 0.12, a temperature degradation factor of 0.005, and an operating temperature of 50° C. Also, the solar constant at 1 AU is 1350 w/m^2.

Solving the above equation for A yields

$$A = 23.52 \text{ m}^2 \approx 24 \text{ m}^2$$

If we assume an areal density of 1.70 kg/m^2, we have a mass of

$$M = (24 \text{ m}^2)(1.70 \text{ kg/m}^2) = 41.0 \text{ kg}$$

<u>Actual mass of fuel used for our system</u>

The actual mass of the LEMBEC on the way up to the space station will be 22,000 kg while the mass on the way down will be 18,900 kg (not counting spent fuel from the way up.) Because of this mass differential, we need to do a more accurate calculation of the mass of fuel consumed in the various maneuvers.

The ΔV needed to reach the space station when it is at 430 km is 0.141 km/sec.

$$\Delta V = (Isp)(g)\ln(\frac{m_1}{m_2})$$

$$141 \text{ m/sec} = (340 \text{ sec})(9.81 \text{ m/sec})\ln(\frac{m_1}{m_2})$$

$$\frac{m_1}{m_2} = 1.043$$

or if m_1 = 22,000 kg, fuel/oxidizer needed is 911 kg

On the way down, the mass will be 18,900 kg minus the mass of the fuel used on the way up, or

$$18{,}900 \text{ kg} - 911 \text{ kg} \approx 18{,}000 \text{ kg}$$

$$\Delta V = (Isp)(g)\ln(\frac{m_1}{m_2})$$

$$109.3 \text{ m/sec} = (340 \text{ sec})(9.81 \text{ m/sec})\ln(\frac{m_1}{m_2})$$

$$\frac{m_1}{m_2} = 1.033$$

or if m_1 = 18,000 kg, fuel/oxidizer needed is 579 kg

Total fuel needed is 911 kg + 579 kg = 1490 kg

Total fuel capacity is 1720 kg. The fuel left over after these two maneuvers are performed is 230 kg, which is 13.4 % of the total fuel capacity.

References

Conway, Bruce A. and Prussing, John E., *Orbital Mechanics*, not commercially published - obtained at a Kinko's Copies, Champaign, Illinois, 1988

Greenwood, Donald T., *Principles of Dynamics*, Prentice-Hall, Inc., Englewood Cliffs, New Jersy, 1988

Haisler, Allen, *Introduction to Aerospace Structural Analysis*, John Wiley & Sons, New York, 1985

Hempsell, C.M. and Hannigan, Russell J., "Multi-Role Capsule System Description", *BIS*, vol. 42, pp 67-81, 1989

Lembeck, Mike, AAE 241 Lecture Notes Series, not commercially published - obtained at a Kinko's Copies, Champaign, Illinois, 1989

Rockwell International Rocketdyne Division brochures
A Past to Build a Future On
Rocket Engine Liquid Propellants

Sutton, George P., *Rocket Propulsion Elements*, John Wiley & Sons, New York, 1986

Structural Analysis

by Shawn Murphy

The structural analysis of the L.E.M.B.E.C. is comprised of a number of different aspects which will be considered. Requirements for the structural analysis from the Request for Proposal (RFP), and those distilled from the RFP are to (1) determine the size of the module needed for the payloads, (2) determine the shape of the module, (3) design the pressure vessel, (4) determine the layout of the components, (5) balance the load, (6) determine the materials to be used, (7) state how the material is to be fabricated, (8) provide adequate micrometeorite impact and radiation shielding, (9) provide adequate thermal control, (10) determine an adequate safety factor, and (11) include a docking adapter. To begin with the size of the module will be considered.

The size of the module is determined by the volume of the logistics needed to be carried on board , the number of people that will be transported, as well as the volume of the components of the other subsystems. As the mission planning analyst has pointed out, the L.E.M.B.E.C system is designed so that a single module can carry out all the requirements as stated in the RFP. Therefore the volumes that drive the design of the L.E.M.B.E.C are approximately:

Logistics:		Volumes(m^3)
	Pressurized Logistics	30.00
	Unpressurized Logistics	38.00
	Fluids and Propellants	3.50
	Total Logistics	71.50

Power and Propulsion:		
	Engines	.0593
	Fuel Tanks	.4212
	Oxidizer tanks	.9977
	Power and Propulsion Total	1.4782

Attitude articulation and control:	
Cold gas tanks	.1855
Environmental control and life support:	
Oxygen	.0395
Nitrogen	.0239
Water	.028
LiOH	.0461
Food	.125
Env. control and life sup. total	.2625
Total Volume Needed:	73.33

Once the volume is known, the shape can be determined.

The basic shape for the L.E.M.B.E.C is a cylinder with a rounded top and bottom as shown in Appendix I. The Logistics compartment is a cylinder with a radius of 2.4 m, and a height of 4m. It will be comprised of two levels, with the separating deck at a height of 1.85 m from the bottom deck. All the decks (bottom, middle , and top) of the logistics/crew compartment will have a hatch of dimensions 1.27m x 1.27m as specified if Ref 1. An additional compartment of radius 2.4 m and height .5 m (cylindrical in shape) will be below the logistics/crew compartment for fuel tanks , thrusters for attitude articulation and control, and any additional items that may be needed. A heat shield for reentry will surround the entire vehicle which on the bottom will extend past the bottom compartment a maximum of 5 meters at the center, and curve up as part of an ellipsoid to the sides. Above the pressurized compartment more fuel is stored and the engines for orbit transfer are mounted. Also mounted there are thrusters for attitude articulation and control. At the top of the pressurized compartment the heat shield curves up from the side of the pressurized compartment to a point that is 1.5 m from center and .5m above the logistics compartment, and then remains flat across the rest of the radius. An explanation of the shape of the heat shield can be found in the reentry and recovery analysts discussion. Calculations made to determined the shapes of the compartments are presented in Appendix XX.

The pressure vessel will be used to carry all the logistics, and also will be used as a crew compartment during emergency escape situations. For these reasons the design as shown in Figure 1 Appendix I was chosen. Note that the crew will be seated on the mid-deck. This decision was made basically for ease of departure when splashing down on earth. Therefore there will have to be a light supporting structure underneath the mid-deck to carry the load of the mid-deck. There is a second hatch to the outside on the side wall of the module. This would be used after the spacecraft has splashed down for ease of departure. A small ladder is attached to the wall to gain easy access to the hatch. The dimensions of the pressurized module are shown below (all cylinders). Refer to Figure 1 in Appendix I for more clarification.

Pressure vessel: $r = 2.4$ m, $h = 4.5$ m

Logistics/crew: $r = 2.4$ m, $h = 4$ m

- upper deck: $r = 2.4$ m, $h = 2.2$ m
- lower deck $r = 2.4$ m, $h = 1.8$ m

Fuel compartment $r = 2.4$ m, $h = 4.5$ m

A number of materials were considered for the pressure vessel. The material that would work the best would be one that had a low density, a high yield strength, and was inexpensive. The formulation for computing mass and thickness are given in Appendix II. Below is a list of four comparable materials, their density, yield strength mass, and cost for the geometry chosen.

Material	Density (kg/m^3)	Yield Strength MPa	Mass (kg)	Cost ($'86)
Aluminum Alloys				
7000 series	2.8 E3	600	205	293
5000 series	2.7 E3	300	395	514
Titanium	4.5 E3	170	1160	6983
Ti-6A14V	4.4 E3	900	214	1607

As can be seen, the 7000 series Aluminum has one of the lowest densities , and one of the highest yield strength for a very good price. The Ti-6A14V has a very high yield strength, but it is too costly for the materials and in terms of weight. Therefore 7000 series Aluminum will be used as the cylindrical shell for the L.E.M.B.E.C. . The computations used to determine the thickness is shown in Appendix II and the result is given below.

For 7000 series Aluminum to be used as a cylindrical shell, the thickness is:

$$t = 1.216 \text{ E-3 m}$$

This is using a safety factor of 3, and assuming a thin wall. A thin-wall approximation is good for $(t/r) < .1$. For this case $(t/r) = .000507$ which is much, much less than .1, and this should be a very good approximation. Note that the same wall thickness was used on all Aluminum.

The layout of the components is shown in Appendix I. One layout is for the logistics setup, and the other is for the emergency escape system. A list of the components, their mass, center of mass, and moment of inertia from the inert program are presented in Appendix III. The arrangement of the basic module (without logistics or crew furniture) is balanced using the inert program. The resulting inertia matrix is given below as well as the resulting center of mass.

$$\begin{array}{lll} I_{xx} = 14100 & I_{xy} = 34.37 & I_{xz} = .3633 \\ & I_{yy} = 14811 & I_{yz} = 0.00 \\ & & I_{zz} = 4408 \end{array}$$

c.m. $x = .002$ $\quad y = 0.00$ $\quad z = 1.5$

Note that not all the inertia cross products are equal to zero. This would seem a problem until it is compared to the actual percentage of this number as compared to the moments of inertia I_{xx} , I_{yy} , and I_{zz}. The value of the cross products is very, very small compared to the these values. It is assumed that when the logistics are loaded in racks, and some not in racks, that they will be distributed about the center of mass evenly, or at least about the z-axis. Also when the crew is on board, they will distribute themselves in the module about the z-axis so as not to create imbalance.

Certain requirements of the RFP have been addressed so far. These are (1) identifying the requirements,(2) determining the size of the module,(3) determining the shape of the module,(4) design of the pressure vessel, (5) the layout of the components, and the balancing of the load. Next to be considered is the material used. As stated, for the pressurized module, 7000 series Aluminum Alloy is being used. For the heat shield the reentry and recovery analyst has chosen Carbon-Carbon composites as the material. Figure 1 in Appendix IV is a graph of strength vs. temperature, and shows how durable Carbon-Carbon is (Ref 2). This is covered completely by the reentry recovery analyst elsewhere in this paper.

If these materials are going to be used, they will have to be fabricated somehow. Aluminum alloys are already being fabricated, so the pressure vessel will material will be fabricated by conventional means. The manufacturing of Carbon-

Carbon, although not wide spread today, is not a difficult task. With minor adjustments to existing manufacturing techniques, complicated shapes using Carbon-Carbon can be made. Already cylinders up to 1.22 m in diameter, and 2.5 cm thick have been made using existing equipment.(Ref 2)

Another concern is the protection of the spacecraft and crew from micrometeorite impact and radiation. To protect the crew and the spacecraft it has been decided to use a dual wall configuration. A dual wall is lighter, safer, and it prevents spalling. Some difficulties when using dual walls is that they are difficult to analyze.

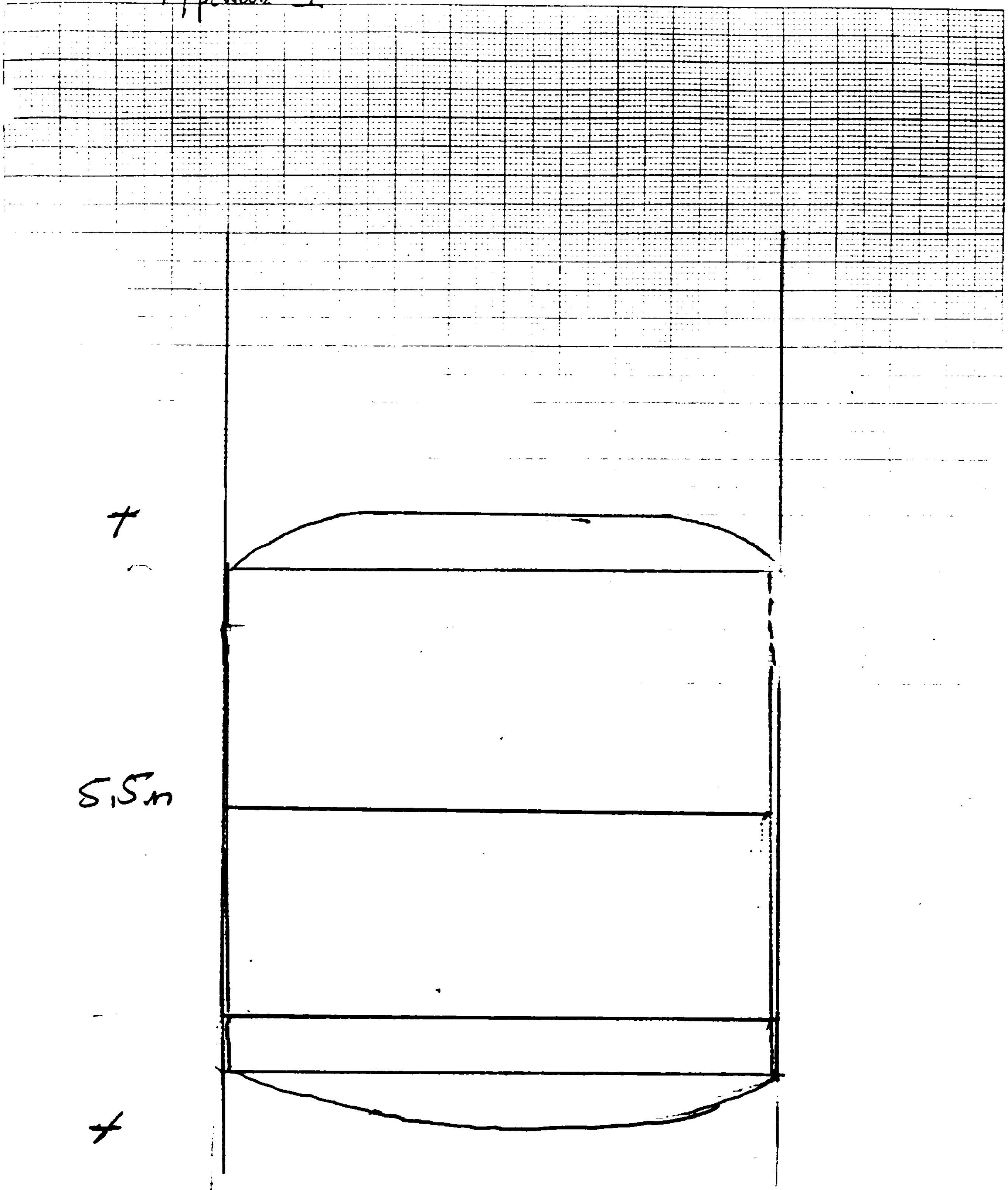
Thermal control is also a concern. Although it is listed as a structure analyst requirement, the environmental control analyst and the reentry and recovery analyst are the two that are really involved with thermal control. Therefore it will be discussed in their sections of this paper.

The safety factor chosen for the pressurized module was a safety factor of three (3) . This was primarily used when computing wall thickness for the vessel. It works out well. Finally a docking adapter must be considered. It was very difficult to find any literature on a docking adapter. Conceivably, for our design, it would have to stick up past the edge of the vehicle on the top.

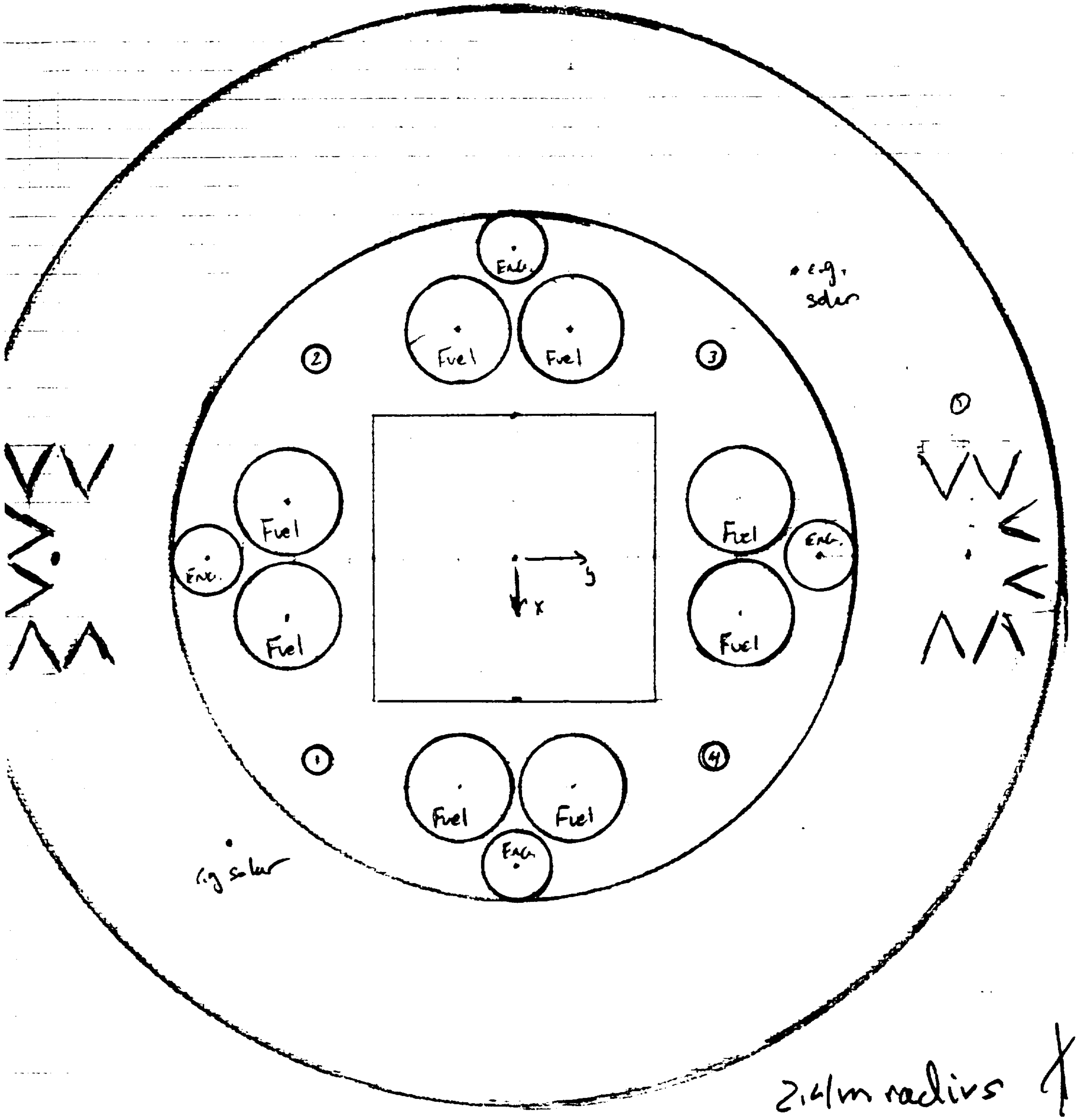
All the requirements have been addressed. First considered. of course, were the requirements themselves. Next the size and shape of the module were discussed. This all led to the pressure vessel design, in which a cylinder was chosen for the body , and it was sectionalized. The layout of the components and balancing them was next addressed, followed by considering the materials used, their fabrication, and protection against micrometeorites and radiation.
Thermal control was referred to environmental control as well as reentry and recovery. A safety factor of tree was chosen, and a docking adapter was discussed. Thus, as stated before, all the requirements have been met.

STRUCTURES

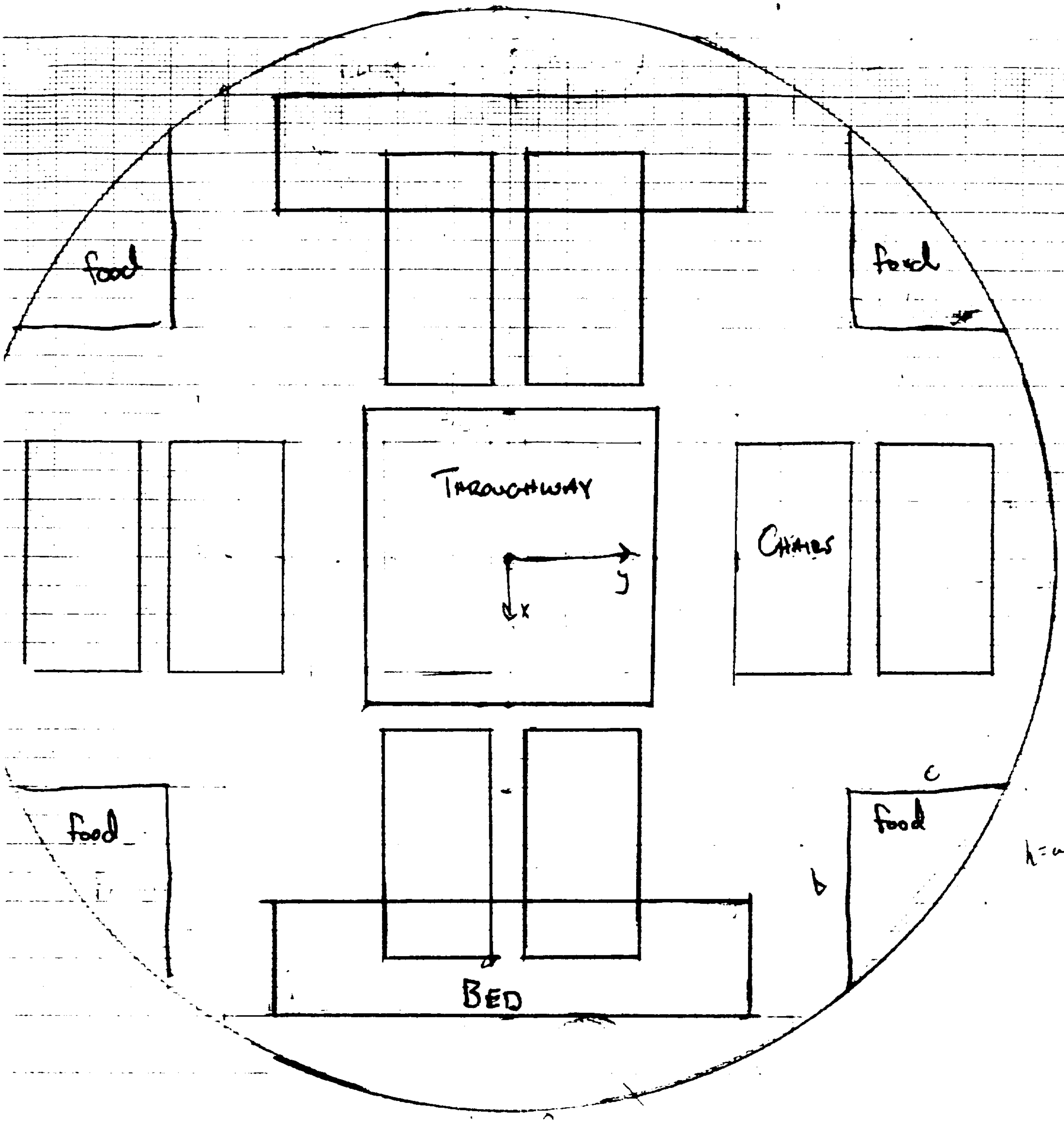
Appendix I



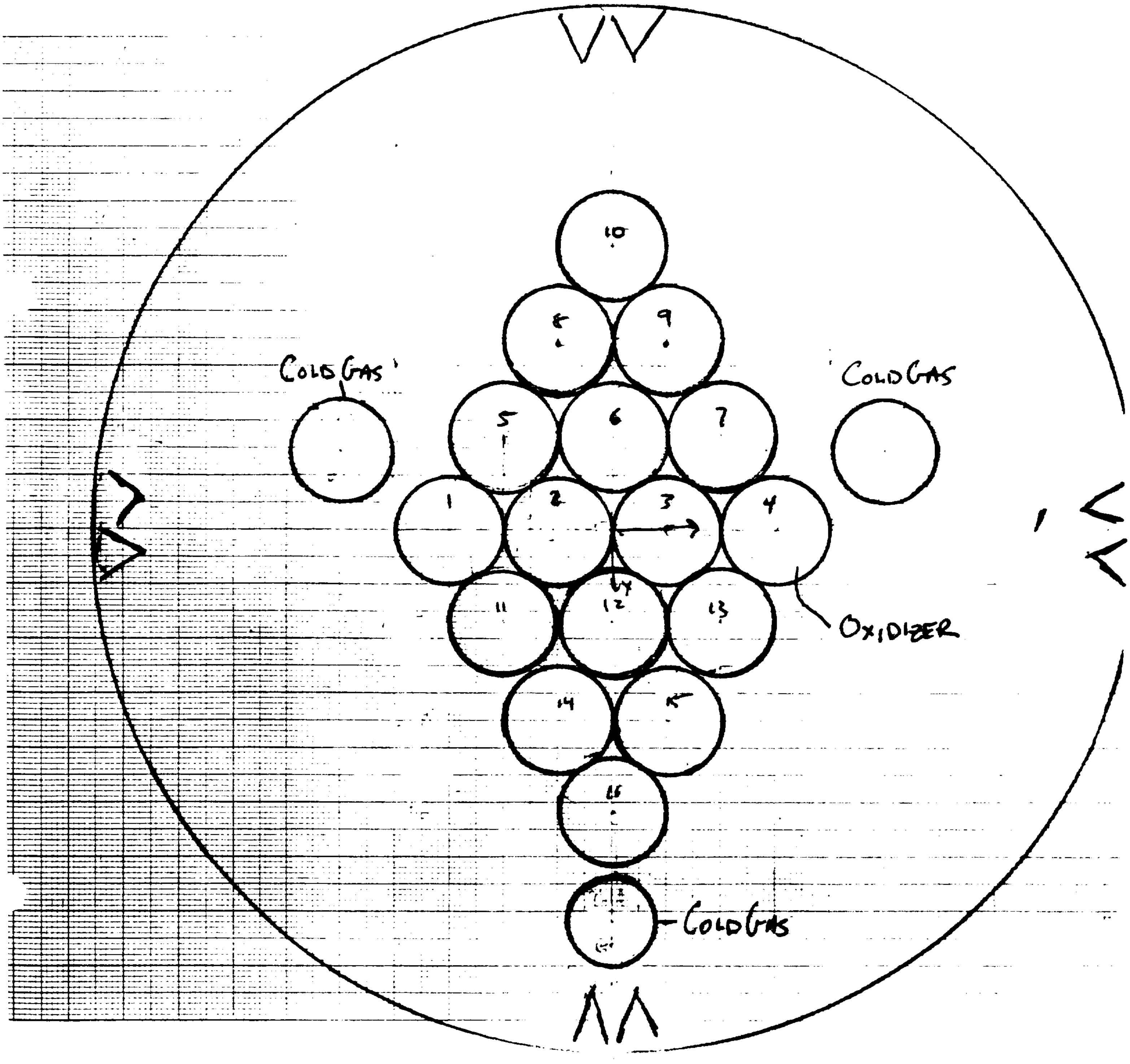
ORIGINAL PAGE IS
OF POOR QUALITY



ORIGINAL PAGE IS
OF POOR QUALITY


food
food
THROUGHWAY
CHAIRS
food
food
BED

ORIGINAL PAGE IS
OF POOR QUALITY



Appendix II

Formulae for calculations :

To determine thickness of vessel:

$\sigma_{tmax} = p(r_i + t/2)/t$ where p = pressure

r_i = inner radius

t = thickness

$(S.F.)(\sigma_{tmax}) = \sigma_y$ S.F. = safety factor

σ_y = Yield Strength

Sample calculation:

$\sigma_{tmax} = (101.325E3\ )(2.4 + t/2)/t$ Pa

$(3)(101.325)(2.4 + t/2)/t = 600E6$ for 7000 series

solve for t => $t = 1.2162E3$ m

Mass:

cylindrical shell: $m = 2\pi r_i t h \rho$ where ρ = density

disc: $m = \pi r_i^2 t \rho$

Moments of Inertia: (Ref 3)

cylindrical shell: $I_{xx} = I_{yy} = (1/2)mr^2 + (1/12)mL^2$

$I_{zz} = mr^2$ where m = mass

r = radius

L = length

solid sphere: $I_{xx} = I_{yy} = I_{zz} = (2/5)mr^2$

solid cone: $I_{xx} = I_{yy} = (3/80)(4r^2 + L^2)$

$I_{zz} = (3/10)mr^2$

solid disc $I_{xx} = I_{yy} = (1/4)mr^2$

$I_{zz} = (1/2)mr^2$

References

1 Lembeck, M. AAE 241 Logistics Requirements , Feb.23, 1989

2 Brahney, James H. . Heat Means Strength , Aerospace Engineering, June, 1987.

3 Ginsberg, J. and Genin, J. Statics and Dynamics, 1984, New York: John Wiley and Sons.

The Requirements

There are four main requirements which must be fullfilled by the reentry/recovery subsystem.

1) Protect the payload and crew from excessive G-forces and heat.
2) A return trajectory which will enable L.E.M.B.E.C. to land safely on Earth.
3) Dissipate the orbit energy in the atmosphere
4) A payload/crew pickup system.

These requirements are all interrelated. The return trajectory is constrained by decceleration and heating limits. The orbit energy will be dissipated mostly as heat , and the recovery system depends on the trajectory.

In addition to these requirements there are others stated in the Request for Proposal which states that the vehicle must stress reliability, endurability, simplicity and to use off the shelf technology.

The Reentry Enviroment

There are three different phases all recovery vehicles (RV) undergo[1]. There is the keplerian phase, the intermediate phase, and the Gas-dynamic phase.

In the keplerian phase forces due to the atmosphere are extremely small. A permanent orbiting satellite such as the space station would be affected by this drag but for an RV it can be ignored when it deorbits.

In the Intemediate phase aerodynamic forces such as drag cannot be neglected. For this report it is assumed that the intermediate phase begins at an altitude of 122 km. Other assumptions for this phase are;

1) Lift and Drag coefficients are independant of Mach number and Reynolds number.
2) The gravitational acceleration is constant.
3) The atmosphere is isothermal and decays exponentially with altitude.
4) Planetary rotation is negligible.

All the assumptions made in the intermediate phase are valid in the gas-dynamic phase as well. There are three major types of trajectory patterns possible in this phase;

1) Ballistic - Nonlifting vehicles with constant flight path angle.
2) Glide Trajectory - Lifting vehicle with zero initial flight path angle.
3) Skip Trajectory - Lifting vehicle with finite initial flight path angle.

Due to L.E.M.B.E.C. 's bullit shape a ballistic entry would seem the most logical choice. The RV will experiance no more than 4 g's and 2000 K temperature. A smaller RV with some lifting capability would suffer less g-forces but other considerations took precedance over the design.

The Keplerian Phase

In this phase L.E.M.B.E.C. is in orbit around the Earth at an altitude between 290 km and 430 km. All the calculations assumed the RV was originally in a cicular orbit at 290 km. The RV could have always made an orbit burn earlier to get to that position. When the RV is orientated in the correct position the main engines will fire to send the RV on a Hohmann transfer back to Earth. The velocity V of the vehicle any where on the transfer orbit is given by;

$$V=u^{.5}\ (2/r-1/a)^{.5} \tag{1}$$

where u = gravitational constant fo earth = 3.986E05 km/s

r = distance from center of earth to vehicle m

a = semi-major axis of the transfer ellipse m

but a is related to the flight path angle γ by;

$$a = r_a/2(\cos^2(\gamma)\ (r/r_a)^2 - 1)/(\cos^2(\gamma)\ (r/r_a) - 1) \tag{2}$$

where r_a = apogee of the transfer ellipse = 6668 km

The ΔV needed by the engines is simply the velocity of the RV in the initial circular orbit minus the velocity of the transfer orbit at that radius;

$$\Delta V = (u/r)^{.5} - u^{.5}(2/r - 1/a)^{.5} \tag{3}$$

When the RV reaches the outer limits of the atmosphere at r_e it will have a speed V_e given by equation (1). The maximum acceleration experienced by the RV is given by:

$$A_{max} = V_e^2 \sin\gamma/2eH \tag{4}$$

where H = scale factor = 6920 m

In figure 1 the ΔV required and acceleration are plotted as a function of γ_{e}. It seems the optimal point is at $\gamma = 0$ where the acceleration is zero but at that γ the vehicle would skip off the atmosphere and be lost. L.E.M.B.E.C. is going slow enough so that γ_e can be small as long as it is slightly negative.

Intermediate Phase

When the RV enters the atmosphere it will pass into the intermediate phase. There is not much of interest that occures in this phase. The RV will follow a constant flight path angle and continue to experience greater g-forces and heat until it enters the gas dynamic phase at about 90 km.

Gas-Dynamic Phase

It is in this phase that peak heating and acceleration occure. Consider the schematic in figure 2. Summing the forces gives;

$$MdV/dt = C_D\rho V^2S/2 \tag{5}$$

where the gravitational force is neglected since it is neglible compared to the drag force for hypersonic velocities. Equation (5) can be solved for V for a ballistic entry and an exponential atmosphere.

$$V = V_e\exp(H\rho_o e^{-(h/H)}/2B\sin\gamma_e) \tag{6}$$

where ρ_o = sea level density = 1,51 kg/m^3

h = altitude m

B = ballistic coefficient = M/C_DA

The acceleration can be found by substituting equation (6) into equation (5).

$$A = V^2\exp(H\rho_o e^{-(h/H)}/B\sin\gamma_e)\rho_o e^{-(h/H)}/2B \tag{7}$$

For a blunt body the coefficient of drag C_D is given by;

$$C_D = 1-\cos^4O \tag{8}$$

The angle O is shown in figure 3. Also is shown is the radius of curvature R_n which defines the area term in ballistic coefficient.

$$S = \Pi R_n^2 \tag{9}$$

For L.E.M.B.E.C.; M = 18908 kg (downmass)

C_D = .3263

R_n = 6m

B = 512 kg/m^3

The g-forces experienced by the crew during reentry is the most critical aspect of the subsystem. Since the RV is to be used in emergency situations it is desirable if the deceleration can be limited to about 30 m/s^2. Normal healthy humans can endure 4 g's up to an hour in the transverse direction(chest to back) but for an injured crew member even this might be too strenuous. L.E.M.B.E.C. will only experience 3.64 g's and only for a few seconds. The acceleration and velocity as functions of altitude are shown in figure 4. From figure 1 the initial flight path angle can be found at 1.25^{o}. This particular value was chosen because it was low enough to give a low ΔV and acceleration but not too low as to put L.E.M.B.E.C. in any danger of skipping off the atmosphere.

[illegible]

[illegible] the LRM and the moments of the articulated components. The [illegible] docking with the space station has yet to be fully developed but [illegible] be functional by 1994.

[Choosing] which system to control the attitude of the LRM depends on the maneuvers it must execute and the time that each one is allowed. [The LRM's] maneuvers can be separated into two categories: transporting the payload and platform missions. Transporting the payload consists of the [illegible] and the final launch from the space station. Platform [missions involve] leaving the space station to go to the platforms and return. The LEMBEC maneuvers will [be] discussed in more detail.

First the maneuvers for transporting the payload will be examined. When the LRM is approaching the space station to dock it will have to perform a decelerating burn with the main engines. When the LRM executes this burn the main engines might be pointed at the space station depending on the orientation of the structure and the approaching trajectory. The minimum distance between the space station and the LRM in which the main engines can be fired has been set at 500 meters. Inside this radius only the gas thrusters can be used to change the LRM's momentum. Since these thrusters are necessary for maneuvering inside the radius, an evaluation of the thrusters for three axis active control should only consider the weight derived by expanding the system for attitude control. After coasting across the 500 meters to the docking adaptor another smaller decelerating impulse is required to slow
the craft to docking speed. During the 500 meter coast the space craft will have to be slewed 180 degrees so the hatch will face the docking adaptor. The hatch and main engines are on the same side of the LEMBEC. When

ORIGINAL PAGE IS
OF POOR QUALITY

[illegible]

<u>[illegible]</u>
deceleration
slew maneuver

<u>[illegible]</u>
1 slew maneuver
reentry slew maneuver
departure impulse

<u>Platform Maneuvers</u>
4 slew maneuvers
1 departure impulse
1 deceleration

The thrust that would be necessary to perform the angular momentum maneuvers is only 15.3% of the thrust needed to separate and decelerate from the docking adapter. To expand the cold gas thrust system to accommodate this additional impulse would likewise lead to a 15.3% increase in tank and propellant mass. This comes out to be [illegible] kg. There is a need for 20 additional thrusters, [illegible] are for redundancy. The mass of the additional thrusters is [illegible] kg each for a total of [illegible] kg. This puts the total mass increase for the expansion of the cold gas system at [illegible] kg.

[illegible]

[illegible] The advantage of this solution is that there is literally nothing new. This advantage is realized in the form of no additional mass, complexity, and cost. All the platforms, however, present a different problem. The station crew would have to perform an extra-vehicular activity (EVA) maneuver to perform any maintenance, resupplying or data taking. This would require more necessary man hours from the crew, which is what we are trying to avoid. The problem of using automation to replace the man hours is that the diversity of the platforms and the jobs that must be done at them will require enormous amounts of artificial intelligence and robotic technology. An analysis of the actual man hours spent in addition to the usual load at the space station was done to determine if the savings by automation would justify the expense and complexity that the system would involve. The total transportation time to the platform and back to the station was calculated to be 62 minutes. If two crew members performed the platform mission at the rate of three missions over a 90 day period, it would amount to 150 hours over the six year life of the LEMBEC system. This total is compared to the 13 hours per week allotted to maintenance at the space station. The amount of extra time spent going to and from the space station by two astronauts represents an increase of [illegible] in the time spent performing EVAs. This small increase does not warrant the development and implementation of an automatic platform servicing system.

The automatic docking that the RFP calls for is a new area of technology for the United States that will require a new system to be developed. The system that will be supported is the Optoelectronic

ORIGINAL PAGE IS OF POOR QUALITY

[illegible] the antenna. [illegible] the orientation of the spacecraft. The antenna needs to [illegible] the communications satellite that it [illegible] messages through. This problem is resolved by using the identification signal that the satellite will emit to guide the direction of the antenna. For orientation of the vehicle, sun sensors will be used to find the angle between the sun and earth. [illegible] will be used to measure the disturbance torques that the LEMBEC [illegible] and also monitor the angular [illegible] of the LEMBEC as it performs its maneuvers. A fixed head star tracker [illegible] any drift that the gyros experience and be used to determine the attitude of the LEMBEC when the sun is blocked by the earth. A fixed head star tracker is chosen over other types because it does not have [illegible] and has no moving parts to limit its life span.

[illegible], James R., ed.
[illegible] Attitude Dete[illegible] and Control.
[illegible] D. Reidel
Publishing Company, 19[illegible].

[illegible], Michael F. and Jones, Da[illegible]
[illegible] Materials: [illegible] Properties
[illegible]
[illegible] Pergamon Press, [illegible].

[illegible]

[illegible], Donald T. [illegible]
[illegible] Jersey:
Prentice Hall Inc., 19[illegible]

Sorenson, Albert A. and Williams, I.J. "Spacecraft Attitude Control." TRW Electronics and Defense Quest, Summer 1982.

"Optoelectronic Docking System." NASA Tech. Briefs, February, 1987 P.30-31.

Klass, Phillip J. "Firms Research Fiber Optic Gyros as Successors to
Ring-Laser Systems." Aviation Week and Space Technology, February, 1989.
P.79-85

Ninomiya, Keiken. "High Precision Attitude Reference and Determination System
for Astronomy Mission Satellites." AIAA Paper 88-388CP

Bollner, M. "A Concept for Operations of the Rosat Attitude Measurement and
Control System." AIAA Paper 86-204

ORIGINAL PAGE IS
OF POOR QUALITY

FREEDOM'S RESUPPLY and EMERGENCY ESCAPE SYSTEM

FREES

FREEDOM RESUPPLY and EMERGENCY ESCAPE SYSTEM

AAE 241
Spacecraft Design
University of Illinois
Champaign, Illinois

May 2, 1989

Group Number 8	Social Security Num.	Subsystem
Aaron Fundich	[redacted]	Power/Propulsion
Stuart Greenfield		Mission Planning and Costing
Chris Higgins	[redacted]	Communications
John Kane		Attitude & Articulation Control
Tony Kim	[redacted]	Reentry/Recovery
Gerald Rauwolf		Life Support
Karl Stevens		Structures

Aaron E. Fundich   John Kane
Stuart Greenfield   Gerald A Rauwolf
Karl E. Stevens   Tony I. K
Chris Higgins

ABSTRACT

The permanent presence of humans in space will be established when NASA and its international partners complete the assembly of Space Station Freedom in the mid-1990's. When this objective is realized, a cost-efficient means of logistics resupply for the station will be imperative to its survival. Another primary requirement of the station will be to provide an emergency escape vehicle for its eight person crew. This paper will responds to a request for a proposal that combines a logistics resupply/crew emergency return system into one vehicle, in effect solving two problems simultaneously.

I. MISSION PLANNING

A. VEHICLE OVERVIEW

FREES has two main objectives: to resupply Space Station Freedom and to return its crew in an emergency situation. This dual-purpose design was chosen to reduce operational costs of the Freedom project and to free the Space Shuttle of resupply duties in order to perform its other responsibilities. FREES also has the capability of transporting equipment and supplies to other space platforms, as well as the ability to return waste products to Earth.

For resupply of Freedom, it was determined that 65,020 kilograms (143,044 lbs.) must be taken up annually (AAE 241, Noteset 238.06). Because the station is to be resupplied every 90 days, or four times annually, the payload was calculated to be nearly 16,255 kg for each service interval. Due to present launch vehicle limitations [see Section I.C-Launch] it was determined that two launches would have to be made every 90 days to resupply the station, each vehicle carrying approximately half the payload.

The vehicle was sized to meet payload and crew volumetric requirements. A maximum diameter of 4.40 meters was chosen in order to fit inside both a Titan IV's payload fairing and the Space Shuttle's cargo bay. It was then decided that the vehicle be cylindrical, allowing for a maximum gain in volume with additional height, and maintained simplicity in the vehicle production processes. Finally, the height of the vehicle was determined to meet volumetric requirements. The payload volume is estimated to be 276 cubic meters annually (AAE 241, Noteset 238.06), or 69 m^3 per service interval, assuming similar payloads each interval. Therefore, each vehicle had to have approximately 35 m^3 of open space available for the payload. Additional height was then added to accomodate tanks, thrusters, landing gear, and the power supply, giving a final height of slightly over 6.5 meters [see Section II-Structures]. Also, the space for payload was 45 m^3, which is about 30 percent greater than necessary, but allows for empty space between supplies and for possible transport of additional payloads if it is later determined to be necessary. This is important because the Japanese and Europeans are planning their own resupply vehicle, which would reduce the U.S. payloads by one-third to one-half, and could therefore be launched in one vehicle. Although this made the vehicle taller than it was wide, it was determined that stability during reentry could be maintained if the center of gravity was kept low.

Finally, the vehicle was designed with the capability of returning all eight crew members at one time. This was done for two reasons. First, the volumetric requirements to sustain eight persons were lower than those of the payload, and therefore all members could be returned in one vehicle. Because there will always be two vehicles at the station, one could return with two members (e.g. if one was injured) and the other would still be capable of returning the other six members if an emergency arose. Secondly, eight was chosen over six because, if the entire crew needed to be returned, all could do so in one vehicle, rather than be forced to return both.

B. ORBITS

Several orbit changes will be required to reach the Space Station, space platforms, and to return to Earth. The orbit of the Space Station ranges from 290 to 430 kilometers, at a 28.5^{o} angle of inclination (AAE 241, Noteset 238.06). A Titan IV launched from Cape Canaveral is capable of putting 18,180 kg into a 405 km orbit, and therefore only a small delta-V may be required to dock with the station [see Section III-Propulsion]. FREES is also capable of resupplying the co-orbiting free-flyers near the station. This too, however, requires only a miminal delta-V.

Another requirement of FREES was to resupply the polar platform orbiting at 824 km altitude, 98.7^{o} angle of inclination. In Appendix I-A , delta-V calculations were made to determine fuel requirements to reach the platform. It was found that a mass ratio of 12.74 was needed, assuming an I_{sp} of 360 seconds. Therefore, for a final mass of 5000 kg, for example, 58,700 kg of propellant would be required for this orbit transfer (note that this does not even include the return trip). Obviously this is not a feasible requirement, as it would require four Titan IV launches (including propellant storage tanks) just to get enough fuel to the station to take a small payload to the polar platform. Therefore, this requirement was removed from the system. It is recommended that payloads be launched directly to the polar platform on an ELV from Vandenberg Air Force Base. Although some type of robotic system would have to be developed to remove the payload, it would be much more economically efficient than transporting the payload from the Space Station.

Finally, an orbit transfer is required to return the vehicle to Earth. This delta-V transfer, from at most 430 km to a reentry altitude of 120 km, is calculated in Section III-Propulsion. Upon reentering the atmosphere, a parachute is deployed and it lands at Cape Canaveral (see Section VIII-Reentry). In case of poor weather conditions in Florida, the alternate landing site will be Edwards Air Force Base in California, although landing there would require a transport back to Cape Canaveral, possibly aboard a railroad car or inside a C-130.

C. LAUNCH

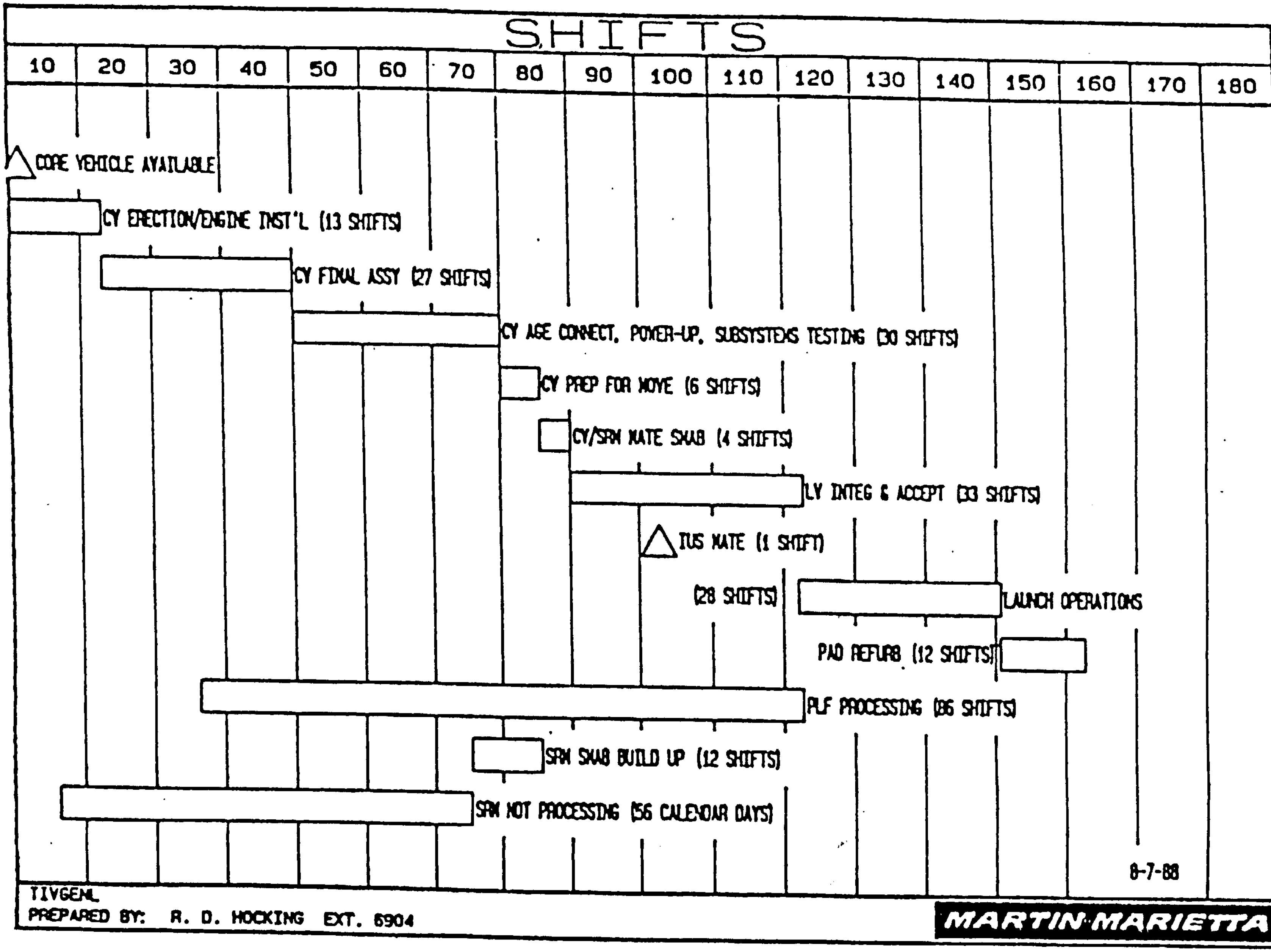
The use of an Expendable Launch Vehicle, or ELV, was a basic requirement of the system. The only available ELV capable of launching large payloads is the Titan IV. It is capable of putting 18,180 kg (40,000 lbs) into a 405 km orbit, with no upper stage required. The payload fairing is 17 meters tall and 5.1 meters in diameter. This is large enough for the resupply vehicle, which is less than seven meters tall and 4.40 meters in diameter (Neilon, p.53).

As stated earlier, two vehicles must be launched every 90 days for proper Space Station resupply. Both will be launched from Cape Canaveral: one from Complex 40, the other from Complex 41 (Neilon, p.53). Preparation of the Titan IV should begin approximately 150 shifts prior to its launch. Assuming two shifts per day, this should begin 75 days prior to launch [see Schedule I.1]. Also, thirteen days prior to a scheduled launch, the launch pads should be prepared according to Schedule I.2 (Neilon, p.50). It is also highly recommended that at the beginning of the project, a spare Titan IV be built and put into a rotation so that there will be an extra launch vehicle available in case of an emergency (a spare resupply vehicle will also be available). Finally, one Delta II launch vehicle will be used to carry the docking adapter to the Space Station.

As stated in the request for proposal, four systems are to be built, or eight vehicles. Two vehicles will be attached to the Space Station. After 90 days, two new vehicles will be launched to resupply the station. After these vehicles have docked, the other two will be loaded with manufactured products, faulty equipment, and waste products and returned to Earth. The vehicle rotation will follow Schedule I.3 . This type of rotation allows for a minimum of six months between launches to repair and refurbish the vehicles. Note that although eight vehicles must be built, only seven are used in the rotation; the eighth is to be used for ground testing and spare parts. This rotation can easily be altered if a vehicle is returned with an injured crew member or some other non-scheduled return.

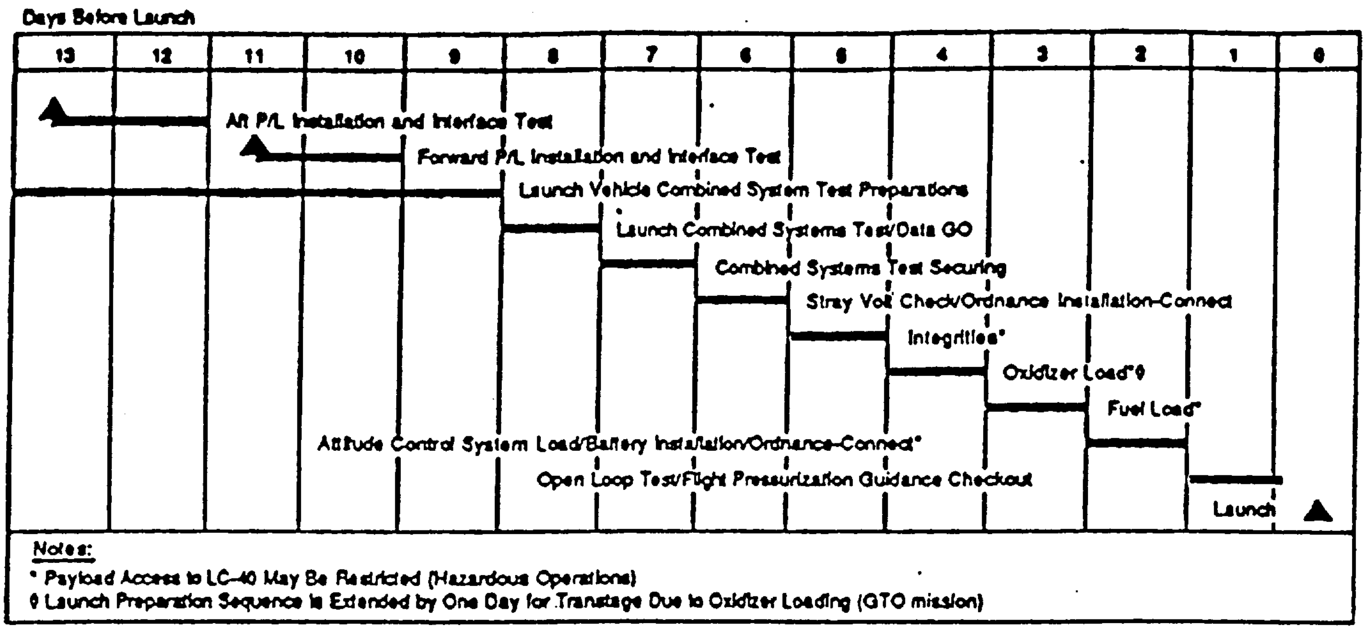
D. PAYLOAD INTEGRATION

FREES is equipped with four racks, identical to those on board the Space Station, and will be available for small payload items. Because the vehicle must be loaded through the top hatch, the bottom should be filled completely prior to filling the top level. The payload should be divided evenly between the two vehicles such that each carries approximately 7850 kg of payload. This value is lower than originally specified because it was determined that no live animals could be resupplied using FREES, which reduced the mass by about 280 kilograms per vehicle per launch (AAE241, Noteset 238.06).



SCHEDULE I.1 Titan IV Launch Schedule (Neilon, p. 55)

5



LC-40 Launch Preparation Schedule

SCHEDULE I.2 Launch Complex 40 Preparation Schedule (Neilon, p. 50)

TABLE I.1 Expendable Launch Vehicles (Neilon)

LAUNCH VEHICLE	PAYLOAD (LBS)	PAYLOAD (KG)
Delta II	8900	4045
Atlas I	12,500	5682
Atlas II	14,000	6364
Titan II	3500	1591
Titan III	22,100	10,045
Titan IV	40,000	18,182

Appendix I-A

SPACE STATION TO POLAR PLATFORM DELTA-V

Orbits: Space Station - 290 km minimum, 28.5^o inclination

Polar Platform - 824 km, 98.7^o inclination

Equations: $\text{Delta-V}_{cir} = [u_E/a]^{.5}$ (AAE 241, Noteset 238.12)

$\text{Delta-V}_{ell} = [u_E(2/r - 1/a)]^{.5}$

$\text{Delta-V}_{incl} = 2V\sin(0/2)$

Constants: $u_E = 3.986 \times 10^5\ km^3/sec^2$

$R_E = 6378$ km

Delta-V_1: 290 km to 824 km, at 28.5^o inclination

$= [u_E(2/6668 - 1/6935)]^{.5} - [u_E/6668]^{.5} = .147$ km/sec

Delta-V_2: Maintain 824 km orbit, 28.5^o inclination

$= [u_E/6935]^{.5} - [u_E(2/7202 - 1/6935)]^{.5} = .286$ km/sec

Delta-V_3: 28.5^o to 98.7^o, at 824 km

$= 2[u_E/7202]^{.5} \sin[.5(98.7^o - 28.5^o)] = 8.555$ km/sec

$\text{Delta-V}_{total} = .147 + .286 + 8.555$ km/sec $= 8.988$ km/sec

$= g\ I_{sp} \ln MR$

where $MR = 1 + [m_p/m_f]$, assume $I_{sp} = 360$ sec

$8.988 = (9.81)(.001)(360) \ln MR$

Solving: MR = 12.74 to transfer to the polar platform from the station

II. STRUCTURES SUBSYSTEM

A. REQUIREMENTS

The structures subsystem provides basic physical support for all of the components of the CERV/LRM system. The design is contingent upon the meeting of several requirements, both taken from the RFP and also derived. Common to all subsystems are a list of general requirements. They are as follows:

1.) The system will consist of three primary components: logistics resupply capsules, Space Station docking adapter, and orbital transfer propulsion subsystem.

2.) The design will stress simplicity, reliability, and low cost and use technology available before 1995.

3.) The system will have a design lifetime of six (6) years, but nothing in its design should preclude it from exceeding this lifetime.

4.) The system's components and payload will be delivered on an expendable launch vehicle. Vehicle components must be able to be returned to earth in the Space Shuttle bay.

There are also requirements that are more specific to the structures subsystem. They are as follows:

1.) Size of spacecraft must be large enough for both human and payload requirements.

2.) Shape of LRM must be compatible with launch vehicle and Space Shuttle, and re-entry considerations.

3.) Optimum material selection.

4.) Include a safety factor prior to design.

5.) Design should withstand regions of space environment, including micrometeorite shielding and thermal control considerations.

6.) Optimize the integration of the different subsystem components into the design.

B. METHOD OF ATTACK

The design process for the structure subsystem is a convoluted process entailing much interaction with all the subsystems. The process is most clearly described in the form of the following flow chart (Chart II-1):

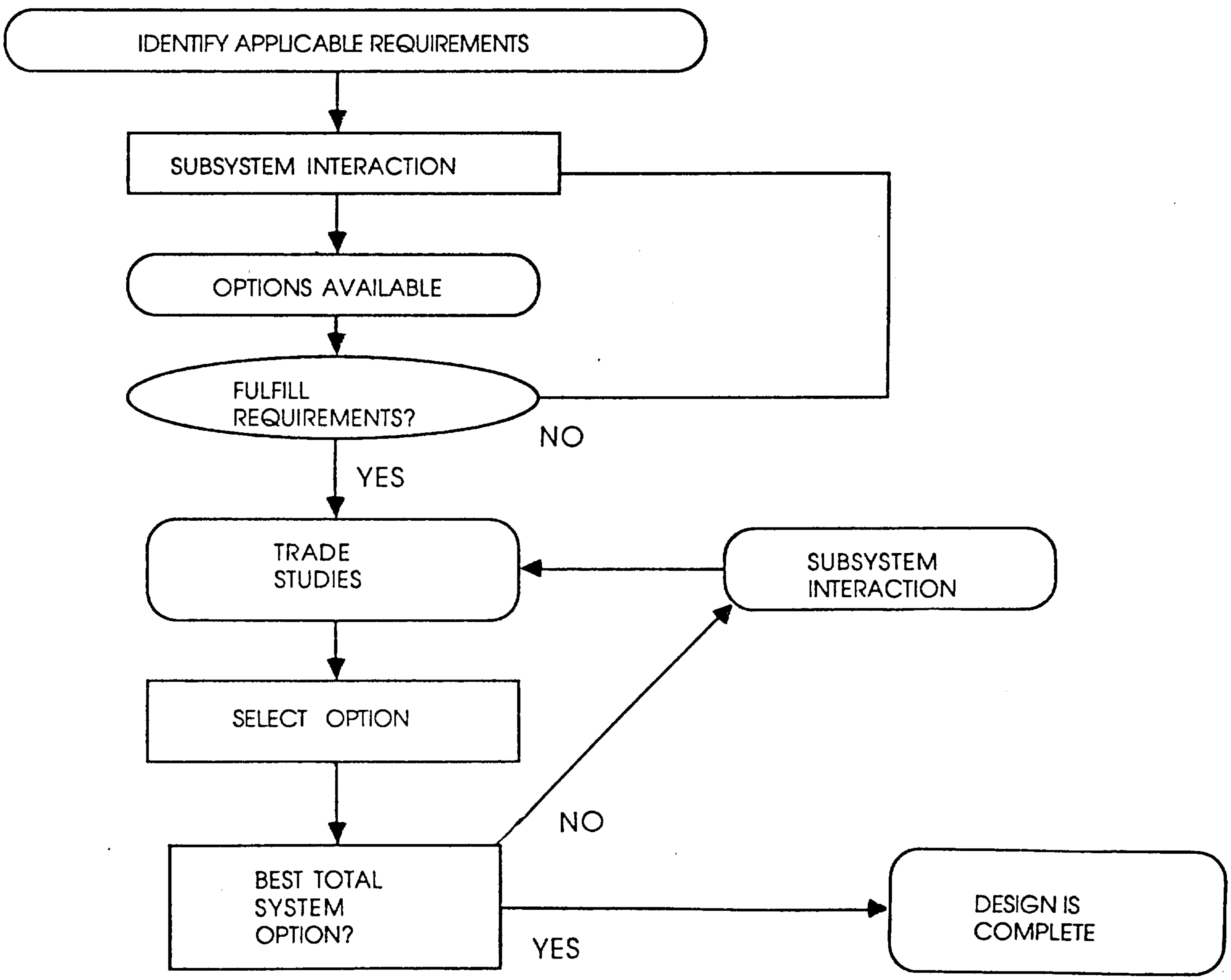


Chart II-1

C. SUBSYSTEM INTERACTIONS

Ineractions with the other subsystems was extensive mainly because of the iterative phase during the design process. The interactions are listed in Chart II-2.

SUBSYSTEM	INTERACTION
Mission Planning/Costing	Received information regarding Space Shuttle and Titan-IV launch vehicle in order to determine size and shape of FREES module. Gave mass breakdown for costing.
Propulsion and Power	Gave mass figures for propellant mass determination. Together determined thrusters and power system layout.
Attitude and Articulation Control	Together determined AACS component layout. Gave overall vehicle inertia tensor for control considerations.
Life Support	Determination of type of thermal control, sizing for human considerations, fire control and component layout.
Communications	Together determine best placement of antennae and computers.
Reentry	Determine best shape for reentry, also placement of landing gear and parachutes.

Chart II-2

D. MODULE SIZE AND SHAPE

After preliminary estimates of the module, it was determined that it was not feasible (see Section I.) to take up all the cargo in one vehicle. So, the next logical choice was to choose two equally sized vehicles for one resupply mission. A cylindrical shape was chosen as the vehicle shape for two main reasons: 1) Excellent compatibility with launch vehicle payload bays and 2) Interior volume shape next best to a sphere for odd-shaped cargo. Also, this shape fit with reentry considerations (Section VIII.) while still allowing a stable center of mass placement.

E. MATERIAL SELECTION

Aluminum alloy was selected as the main structure material because of two overriding factors: one, aluminum's performance record; and two, aluminum's low cost. It has been used extensively in past space missions and it has been found to be very reliable. Aluminum is relatively abundant and easily formed or machined into various shapes. This, coupled with the fact that much testing has been performed on aluminum leads to very low costs in both development and fabrication.

The trade study displayed in Chart II-3 lists several structural materials and their key properties in a ranking scheme. A score of ten (10) received is the best in the category while a score of one (1) is the worst.

MATERIAL	COST	DENSITY	YOUNG'S MODULUS	YIELD STRENGTH	STRENGTH / WEIGHT	COMMENTS
Ti alloys	4	3	7	8	6	very strong, too expensive
Al alloys	10	7	5	5	6	low density, good strength, proven
Steel	10	2	9	9	6	too heavy, other-wise good
Mg alloys	8	8	2	2	6	too weak
KEVLAR	- -	10	5	8	9	technology not ready
CFRP	1	10	8	7	10	too expensive

CHART II-3

Chosen

F. SAFETY FACTOR DETERMINATION

Before the final design was made, a safety factor had to be included into the calculation for module wall thicknesses. This was done by performing a trade study of safety factor vs mass. By choosing a vehicle minimum cargo mass capability of 9000 kg, a safety factor of 1.5 was chosen. This can be seen in Figure II-3. Equations appear in Appendix II.

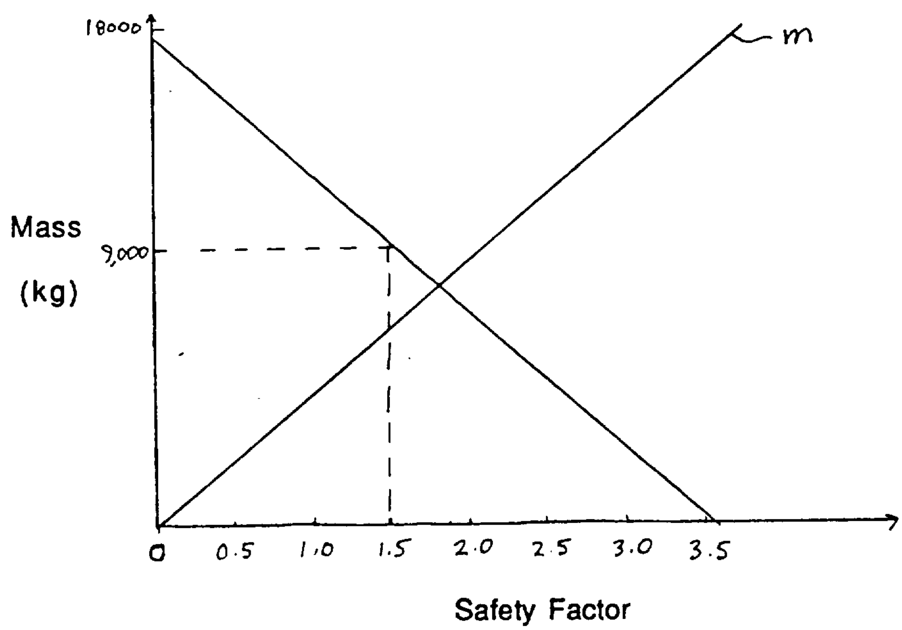


Figure II-3

G. PROTECTION FROM THE SPACE ENVIRONMENT

a.) Micrometeorite Shielding

Due to the presence of debris and micrometeorites in LEO (low Earth orbit), the FREES system had to be designed to withstand impacts of particles up to a 4.0 cm diameter and travelling close to 20 kilometers per second. The optimal design to handle the impacts is a dual wall-empty construction. (Koepke, 11) The shielding process is illustrated in Figure II-4.

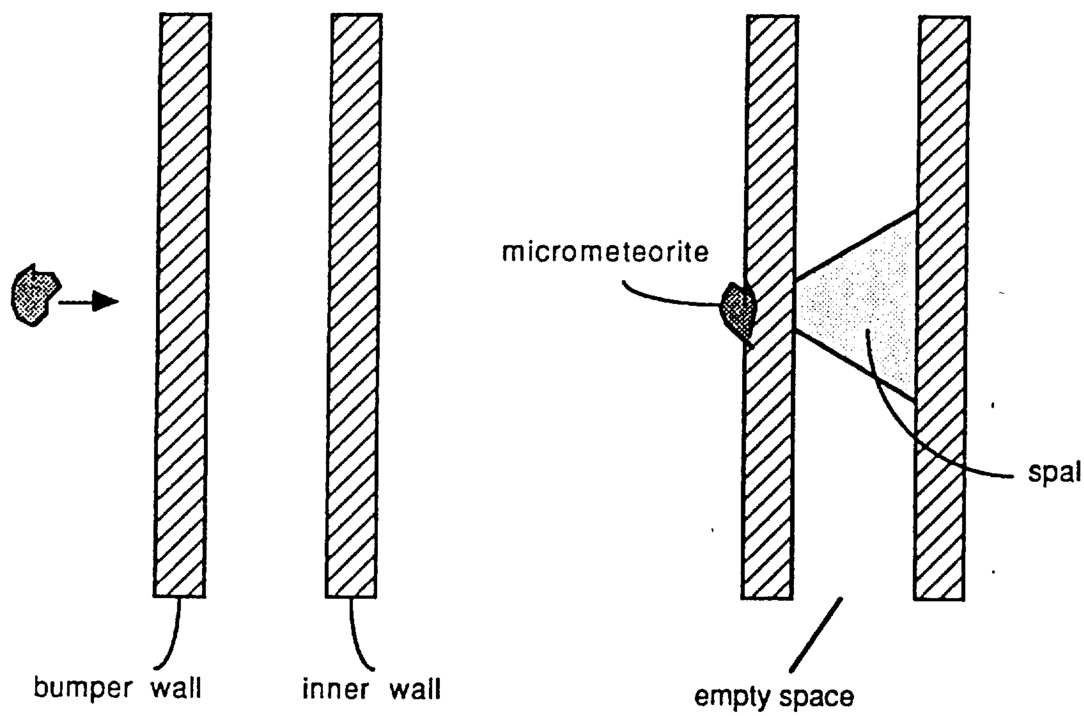


Figure II-4

The outer or bumper wall is impacted by the particle first. Upon collision, material from the inside surface is sprayed in a conical fashion. This condition is known as spalling. The addition of an inner wall prevents spalling from being a problem. Since the structure material used for the FREES module is aluminum alloy, the wall thicknesses and the space between them can be determined from the Nysmith Equation. (Koepke,16) The equation and the calculations are presented in Appendix II.

b.) Thermal Control

During a normal resupply mission the LRM will be subjected to temperatures as low as -460 ^{o}F in space and as high as a few hundred degrees during reentry. A thermal control system was chosen that would maintain the module's cabin temperature at or near 70 degrees.

The chosen system is listed, along with other options, in Chart II-4.

OPTIONS FOR THERMAL CONTROL		ADVANTAGES/DISADVANTAGES
Passive Thermal Control	Thermal Coating	Only good for reflecting or absorbing energy, simple, thin
	Thermal Insulator	Reduces rate of heat flow between two surfaces, light, simple
	Phase-Change Materials	Not applicable to FREES
Active Thermal Control	Examined in section V	

Chart II-4

The same material as the heat shield, REI-Silicate (properties in section VIII), was chosen as the thermal insulator. Using ITAS, a thermal analysis program, a thickness of 2.5 cm was determined to be sufficient. (Bain, 21) This insulation is applied directly to the outside of the bumper wall.

c.) Radiation Protection

The optimal design for protection against radiation in a space environment is a multi-walled filled structure. (Koepke, 11) Because a dual wall-empty configuration coupled with thermal insulation does stop most of the radiation, the addition of radiation insulation between the two structural walls would add unnessasary mass, making spalling a problem. Therefore, it was decided that any addition would do more harm than good.

H. COMPONENT LAYOUT

The placement of the components was a very important part of the design process. The layout was made as simple and rational as possible. The amount of wiring and propellant piping was kept to a minimum. The overall center of mass was kept as close to the geometric cylinder centerline as possible. Also, the center of mass was kept in the bottom half of the vehicle in order to promote stability. Figure II-5 through Figure II-9 show the final component layout of the LRM. Component coordinates used for the inertia resolving program INERT is included in Appendix II. (Lembeck, 2)

The final inertia tensor for the module is:

loaded with 9000 kg cargo

$$\begin{bmatrix} 403655.4 & -1414.7 & -3484.4 \\ -1414.7 & 367274.3 & -18751.1 \\ -3484.4 & -18751.1 & 76239.5 \end{bmatrix} \text{kg-m}^2$$

center of mass x = 0.067 m; y = 0.136 m; z = 3.14 m

total mass = 17650.1 kg

Side View

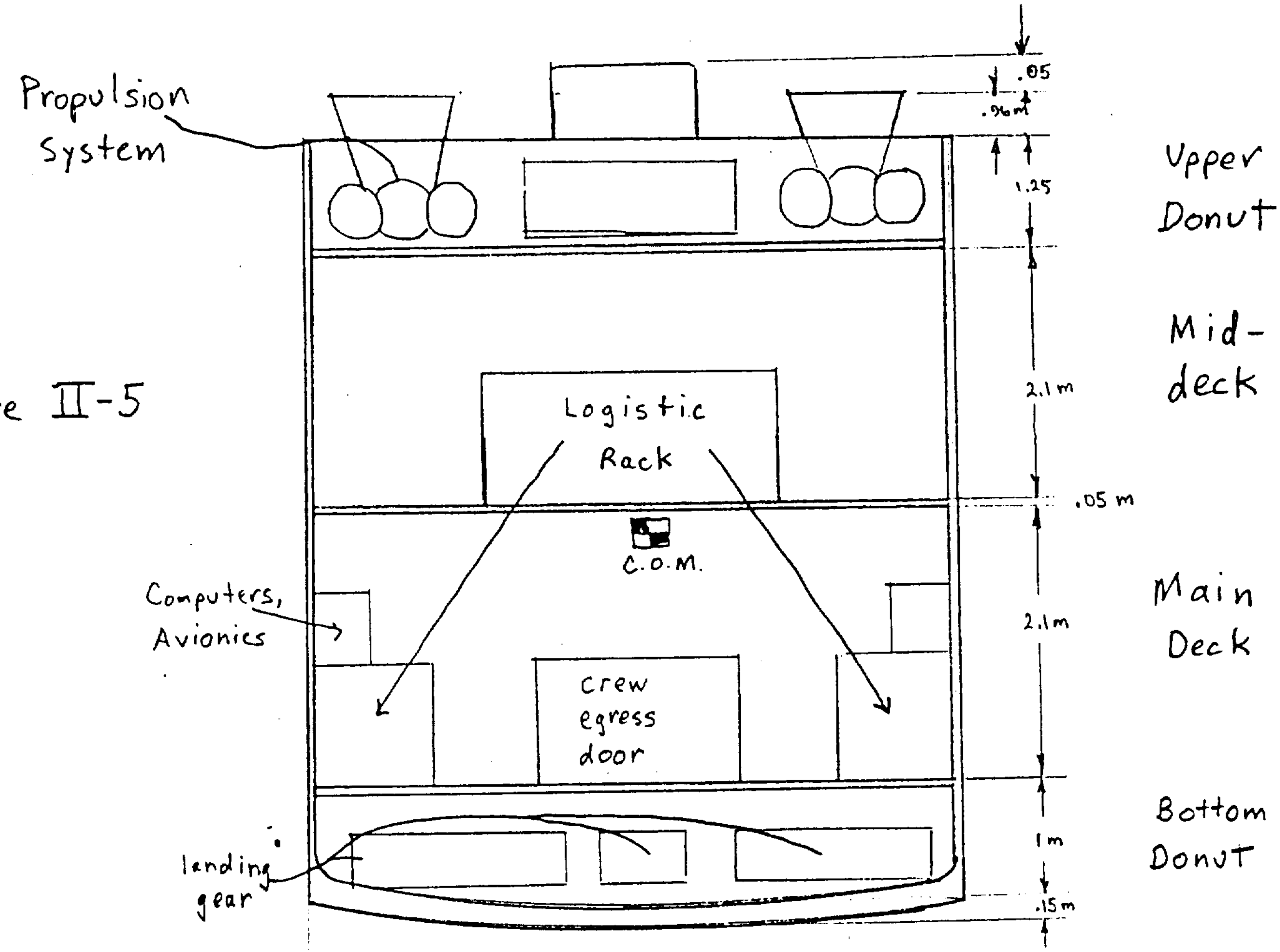


Figure II-5

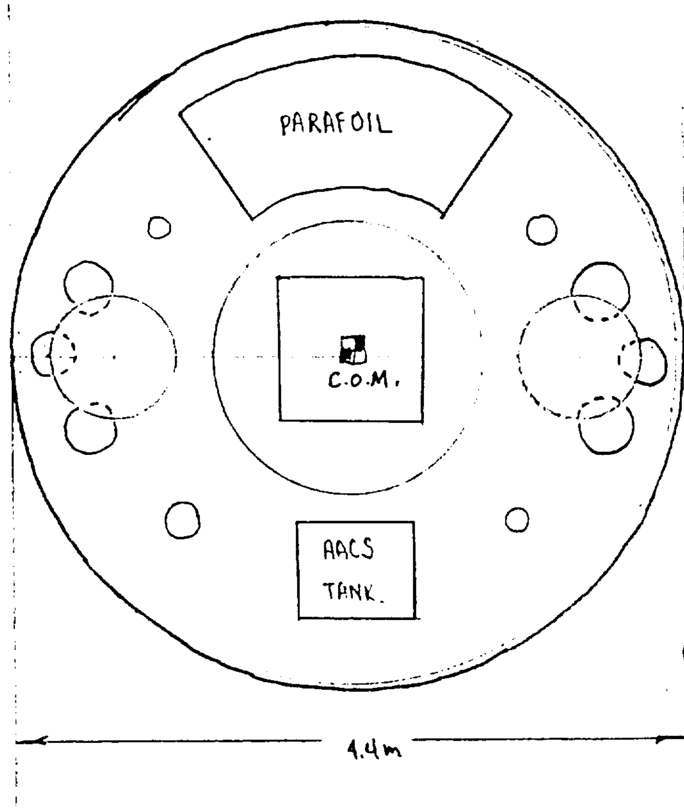


Figure II-7

ORIGINAL PAGE IS
OF POOR QUALITY

FIGURE II-7

Bottom Donut

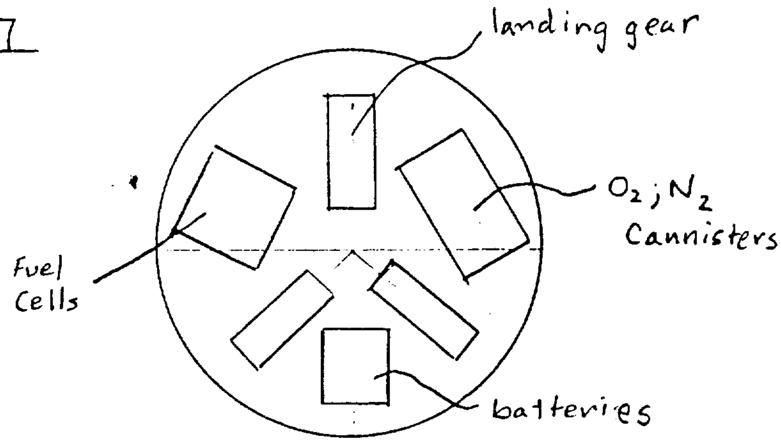


FIGURE II-8

Main Deck

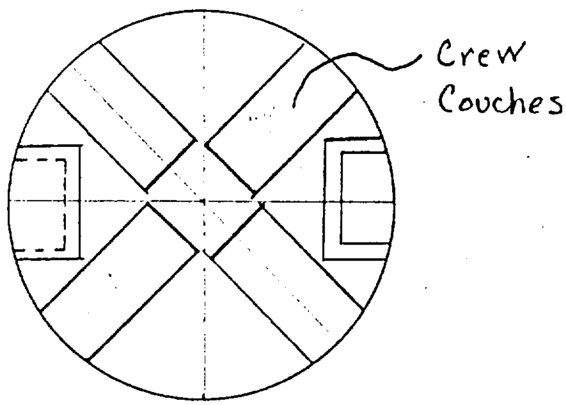
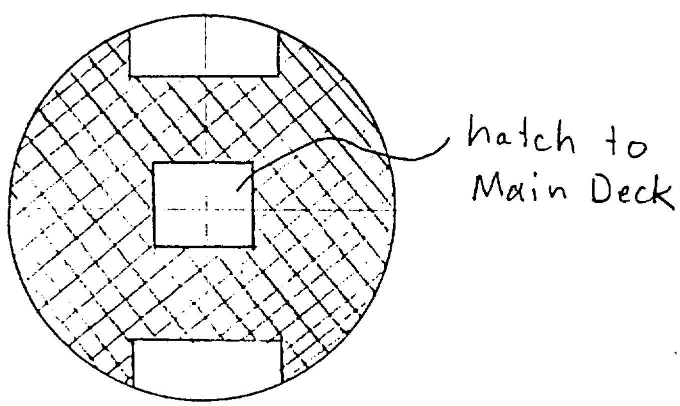


FIGURE II-9

Mid-deck Floor



no cargo	380775.7	-1255.9	-3154.5	
(in CERV mode)	-1255.9	344659.2	-18029.3	$kg\text{-}m^2$
	-3154.5	-18029.3	61246.3	

center of mass $x = 0.13$ m; $y = 0.277$ m; $z = 3.34$ m

total mass = 9850.1 kg

The z-component of the center of mass (c.o.m.) in both cases is below the volumetric center of mass and this should not be overlooked. When travelling from one platform to another or during the orbital maneuvering prior to reentry the c.o.m. will be situated closer to the leading edge. Normally, this would mean an unstable system but in this case it becomes irrelevant. The forces felt by the module during calm space flight are taken care of easily by the on board reaction control system. The c.o.m. is much more important during the rough reentry ride and the component layout of the FREES module leads to stabilization.

I. DOCKING ADAPTER

A very simple design was chosen for the docking adapter. It is basically a cube with six ports for docking. Mission Planning required that the adapter be less than 2.89 meters per side. Aluminum alloy was chosen for the adapter for the same reasons mentioned earlier. This allowed for an approximate mass of 2400 kilograms for the adaptor. The adapter was also designed for communication and power umbilicals between the Space Station and the LRM.

APPENDIX II

A. NYSMITH EQUATION

The Nysmith Equation for metals is given by

$$t_2 = 5.08 * D * V^{.278} * (D/t_1)^{.528} * (D/s)^{1.39}$$

where t_2 = bumper wall thickness, cm

t_1 = inner wall thickness, cm

V = velocity of particle, km/s

D = diameter of particle, cm

s = separation between the two walls

For a good design $s < 30 * D$ and $.25 < t_2/t_1 < 1$

For a value of $s = 15$ cm, $t_2 = 0.1$ cm and $t_1 = 0.56$ cm

But, after a determination of safety factor of 1.5 (in part b) below)

$s = 15$ cm, $t_2 = 0.3$ cm and $t_1 = 0.65$ cm.

B. SAFETY FACTOR EQUATIONS

$$S * t * S.A. = \text{mass} * \text{density}, \text{ with } t = t_1 + t_2$$

S.A. = surface area of spacecraft

This yields a relation of $S/m = 3032$

Also, $M_{cargo} = m - M_{comp}$

where M_{cargo} = cargo mass and M_{comp} = mass of components

Ideally , cargo mass should be around 9000 kg and S should correspond to this value.

C. INERT INPUT DATA FILE

```
:test.dat
nell
0600
0600
600
25
o00
at Shield
42.9
42.9
5.7
.15
6
rop-NTO1
.2
0
.0
.2
0
.2
.67
.67
03
4.7
op-NTO2
.2
0
.0
.2
0
.2
1.67
67
.03
4.7
op-MMH1
.6
.0
.0

28.6
0.0
28.6
-0.67
-1.67
6.03
268.13
y
1
1
Prop-MMH2
28.6
0.0
0.0
28.6
0.0
28.6
0.67
1.67
6.03
268.13
y
1
1
Prop-Hel1
0.22
0.0
0.0
0.22
0.0
0.22
2.0
0.0
6.03
5.23
y
1
1
Prop-Hel2
0.22
0.0
0.0
0.22
0.0
0.22
-2.0
0.0
6.03
5.23
y
1
1
Top Struc
984.2
0.0
0.0
984.2
0.0
1970
0.0

0.0
6.5
836
y
1
1
Bot Struc
984.2
0.0
0.0
984.2
0.0
1970
0.0
0.0
0.15
836
y
1
1
MidFloor
471
0.0
0.0
471
0.0
941.2
0.0
0.0
3.3
400
y
1
1
AACS Prop
2.65
0.0
0.0
9.66
0.0
9.66
1.67
0.0
6.03
100
y
1
1
ECLSS O2 1
0.56
0.0
0.0
0.19
0.0
0.56
-1.17
1.33
0.65
15
y

1
1
ECLSS O2 2
0.56
0.0
0.0
0.19
0.0
0.56
-0.83
1.23
0.65
15
y
1
1
ECLSS O2 3
0.56
0.0
0.0
0.19
0.0
0.56
-0.4
1.13
0.65
15
y
1
1
ECLSS O2 4
0.56
0.0
0.0
0.19
0.0
0.56
0.10
1.03
0.65
15
y
1
1
ECLSS N2 1
4.15
0.0
0.0
1.14
0.0
4.15
-0.99
1.33
0.65
43
y
1
1
ECLSS N2 2
4.15

0.0
0.0
1.14
0.0
4.15
-0.60
1.23
0.65
43
y
1
1
ECLSS N2 3
4.15
0.0
0.0
1.14
0.0
4.15
-0.24
1.13
0.65
43
y
1
1
ECLSS N2 4
4.15
0.0
0.0
1.14
0.0
4.15
0.0
1.03
0.65
43
y
1
1
Fuel Cells
55.2
0.0
0.0
25.0
0.0
55.2
-0.75
-1.17
0.65
800
y
1
1
Batteries
28.3
0.0
0.0
5.7
0.0

32.6
1.33
0.0
0.65
375
y
1
1
LandGear 1
3.2
0.0
0.0
34.3
0.0
34.3
-1.33
0.0
0.65
200
y
1
1
LandGear 2
34.3
0.0
0.0
3.2
0.0
34.3
0.667
1.17
0.65
200
y
1
1
LandGear 3
34.3
0.0
0.0
3.2
0.0
34.3
0.667
-1.17
0.65
200
y
1
1
Parachute
18
0.0
0.0
18
0.0
18
-1.56
0.0
6.03

400
y
1
1
CarRack u1
29.5
0.0
0.0
27.5
0.0
12.3
1.67
0.0
3.85
75
y
1
1
CarRack u2
29.5
0.0
0.0
27.5
0.0
12.3
-1.67
0.0
3.85
75
y
1
1
CarRack d1
27.5
0.0
0.0
29.5
0.0
12.3
0.0
1.67
1.75
75
y
1
1
CarRack d2
27.5
0.0
0.0
29.5
0.0
12.3
0.0
-1.67
1.75
75
y
1
1

Engine 1
42.5
0.0
0.0
42.5
0.0
25.2
0.0
1.5
6.23
140
y
1
1
Engine 2
42.5
0.0
0.0
42.5
0.0
25.2
0.0
-1.5
6.23
140
y
1
1
Catch-All
1849
0.0
0.0
1849
0.0
2354.5
0.0
0.0
3.25
500
y
0
```

INERT
Input Data File

ORIGINAL PAGE IS
OF POOR QUALITY

III. PROPULSION

The propulsion subsystem of FREES has one major requirement: to enable the module to get where it needs to go. Desirable characteristics of this subsystem include low mass and volume, simplicity, reliability, and low cost.

Of the four basic propulsion system categories (solar electric, electric, chemical, and nuclear), chemical propulsion is the only one that fits the needs of FREES. The other three are more suitable to long term missions; due to this and various other drawbacks they were quickly dismissed as propulsion system candidates.

A. PROPELLANT

The propellant choice for FREES was the first objective to be completed in designing the propulsion subsystem because it is the design driver in terms of subsystem mass and volume. The first step toward completing this objective was to decide whether solid or liquid chemical propellants would be used.

Propellant consumption will occur at three times during a typical FREES mission: rendezvous with the Space Station after escape from the ELV, trips made to and from the co-orbiting platforms in the 28.5 inclination, and during the return to Earth from the station. The Δ v's required to perform these operations have been calculated and are tabulated in Table III.1. These calculations are shown in detail in Appendix III-A.

MANEUVER	MAXIMUM	V REQUIRED (m/s)
Station Rendezvous		12.38
Platform Trips *		28.50
Return to Earth		119.12
Subtotal		160.00
Safety Factor		x 1.10
TOTAL ΔV REQUIRED		176.00

TABLE III.1 - TOTAL ΔV REQUIREMENTS OF FREES

Due to the varying nature of the thrusts necessary to perform the above maneuvers, a liquid propellant system will be used in the propulsion system design. Solid propellant systems simply do not provide the throttling and start/stop capabilities that are necessary for FREES. A study was conducted to determine which liquid bipropellant system would best apply to the module and its missions. Desirable propellant characterisics include low density, low toxicity, high performance (I_{sp}), good storability, and low cost. Some properties of liquid fuels and oxidizers are shown in Table III.2 and a cost comparison between several propulsion systems is shown graphically in Figure III.1. The propellant that was eventually chosen was a monomethylhydrazine - nitrogen tetroxide combination (MMH/NTO).

Although the I_{sp} of MMH/NTO is lower than some other propellants, other important considerations played a role in its selection. At the top of the list are its excellent storability

PROPELLANT	LIQUID FLOURINE	HYDRAZINE	LIQUID HYDROGEN	MONOMETHYL-HYDRAZINE	NITROGEN TETROXIDE	LIQUID OXYGEN	ROCKET FUEL RP-1
FORMULA	F_2	N_2H_4	H_2	CH_3NHNH_2	N_2O_4	O_2	Hydrocarbon $CH_{1.97}$
MELTING/FREEZING POINT (°K)	53.7	274.5	14.0	220.7	261.5	54.4	225
DENSITY (g/cm^3)	1.66	1.01	0.071	0.879	1.45	1.14	0.58
VAPOR PRESSURE (MPa)	0.158	0.0014	0.0083	0.0069	0.0958	0.0052	0.002
MOLECULAR MASS	38.0	32.05	2.02	46.08	92.02	32.00	175
TOXICITY	HIGH	HIGH	LOW	HIGH	HIGH	LOW	HIGH
STORABILITY	BAD	GOOD	BAD	GOOD	EXCELLENT	GOOD	GOOD
PERFORMANCE	EXCELLENT	GOOD	HIGH	GOOD	GOOD	GOOD	GOOD

TABLE III.2 - PROPELLANT PROPERTIES (SUTTON, pp 170-182)

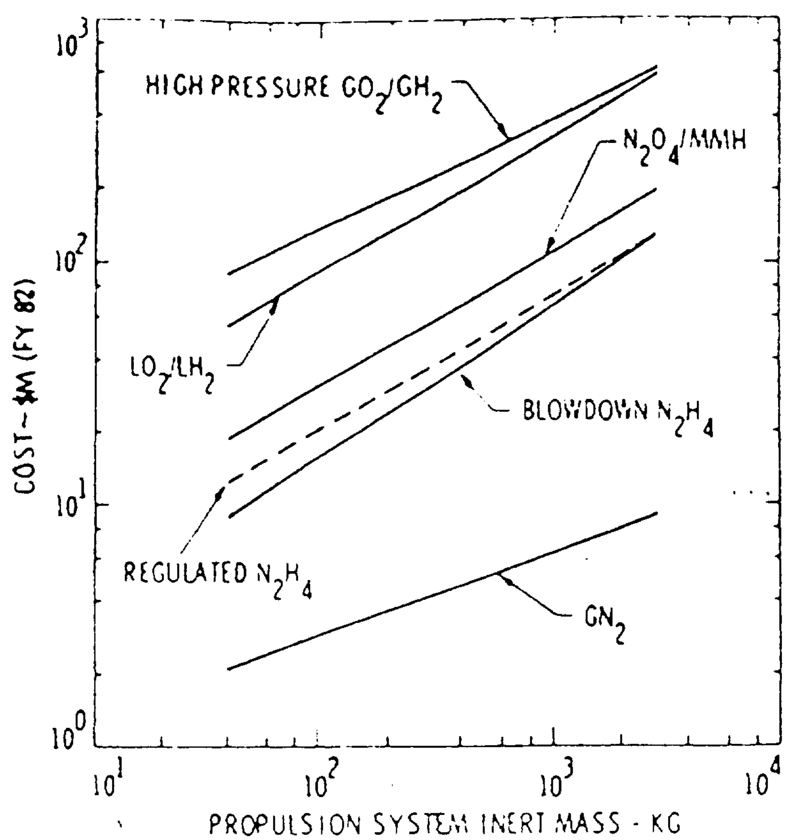


FIG. III.1 - Propulsion Systems Design and Development Cost Estimates (RAULT, p.5).

properties. With the module rotation system (see Section I - Mission Planning) leaving vehicles docked in space for up to three months at a time, it is important to have a propellant that will not evaporate too quickly. Another good property of the MMH/NTO propellant is the hypergolicity between the fuel and oxdizer, which prevents the need for a complicated ignition system, thereby helping to meet the overall design requirement of simplicity. Another favorable characteristic about the chosen propellant is that it has been used successfully in the Space Shuttle Orbital Maneuvering System, which performs duties very similar to those of the propulsion system of FREES (Sutton, p. 199).

After choosing the propellant type, the next step in the design was to compute the amount of propellant needed to provide thrust for the required orbit velocity changes. Plugging the total Δv required of 176 m/s into Tsiolkovsky's Equation

$$\Delta v = g I_{sp} \ln (M_i / M_f)$$

with I_{sp} = 313 sec and M_i = 18000 kg yields a value of M_f = 16997.0 kg. The mass of propellant needed is therefore 1003 kg. After multiplying by a safety factor of roughly 1.15, the final design mass of 1150 kg of MMH/NTO propellant was decided upon.

The optimum oxidizer to fuel mass ratio for MMH/NTO is about 1.65 (Sutton, p. 199). The subsequent final masses of the propellant components are 734 kg of the oxidizer nitrogen tetroxide and 434 kg of monomethyl-hydrazine.

The next task to be completed was to size the spherical tanks in which the fuel and oxidizer would be contained. The tanks will be constructed with a high strength titanium alloy (ρ = 4.7 g/cm^3, Ashby, p. 52) each equipped with a microgravity propellant management device (Rault, p. 6). There are two fuel tanks and two oxidizer tanks, each having diameter 80 cm and mass 46.67 kg. All of these tanks will be 5 mm thick, easily able to contain the liquids inside them. Calculations for these tanks are shown in detail in Appendix III-A.

B. ENGINE CONFIGURATION

The main engines that will be used in the propulsion system of FREES are the Orbital Maneuvering Engines (OMEs) that are used on the Space Shuttle. This was decided upon long after the decision was finalized to use MMH/NTO as the propellant for the subsystem. When it was realized that the Δv required by the Shuttle is in fact quite similar to that which will be used by FREES, the OMEs immediately became the primary choice for the main thruster system.

The similarities between the total mass of FREES during reentry (about 18,000 kg) and the return mass of the Shuttle (payload 11,360 kg) made the final decision to use the OMEs on FREES an easy one. The OMEs are perhaps a bit large for use on the logistics module - the Shuttle total mass upon reentry is approximately 50,000 kg (Sutton, p. 16). However, in the event of a catastrophic

50,000 kg (Sutton, p. 16). However, in the event of a catastrophic emergency aboard the station, these thrusters could be depended upon to quickly provide large amounts of thrust and enable the crew to safely escape.

The use of already proven Shuttle technology will result in a cost savings of tens of millions of dollars in the design and production of FREES (see Section IX - Mission Costing). The OMEs, when designed for the Shuttle, are designed for 100 flight missions, at least 500 starts, and a service life of ten years (Sutton, p.199). These parameters easily exceed the minimum design lifetime of six years established for FREES.

A simplified half-section of the thrust chamber of one of the two engines to be employed in the FREES propulsion system is shown on the following page in Figure III.2. The operating pressure in the chamber is 128 psia and the nozzle area ratio is 55 : 1. Each engine has a mass of 120 kg and develops a vaccuum thrust of about 26.7 kN when fed the MMH/NTO propellant (Sutton, p. 199).

The engines and the propellant tanks will be placed in the upper compartment of the FREES vehicle. The thrusters wil be aligned symmetrically on both sides of the cargo hatch/docking adaptor on top of the module. The propellant tanks will feed the engines in a redundant propellant feed system similar to that of the Shuttle. In case of failure of one of the OMEs, one engine is capable of providing sufficient thrust for the total required Δv during reentry by itself. The layout of the tanks, engines, and interaction with other components in the top room of the vehicle is shown in the structures subsystem (Section II).

Two small helium tanks have been designed to carry the high pressure inert gas that is used to push the fuel and oxidizer to the engine. These tanks will each carry 2.23 kg of helium. The tanks will be constructed of prestressed composite kevlar (ρ = 1.44 g/cm^3, Zweben, p. 4). Calculations for this data are shown in Appendix III - A.

A schematic drawing of the propellant feed system is shown below in Figure III.3. The valves, pumps, piping, and wiring are estimated to have a mass of about 40 kg.

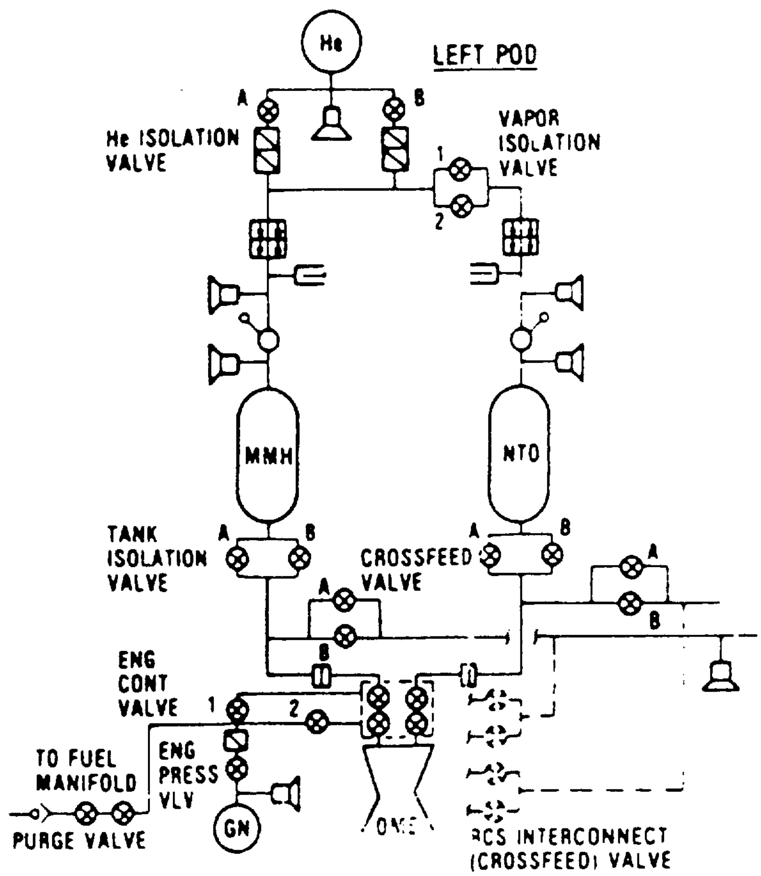


FIGURE III.3 - PROPULSION SUBSYSTEM SCHEMATIC DRAWING

The main thrusters will operate during operations when substantial Δv 's are necessary, as mentioned before. However, when the module is in the proximity of the station (within 5 km), these engines cannot be allowed to run due to the toxicity of the exhaust. Monomethylhydrazine is a toxic fuel, and its fumes would surely affect Station experiments in an adverse manner, as well as causing possible contamination to the station atmosphere. During these maneuvers near the station (including docking) the attitude and articulation control system and its cold nitrogen gas thrusters will be used exclusively. This subsystem will be discussed in the next section of the report.

COMPONENT	QUANTITY	UNIT MASS (kg)	TOTAL MASS (kg)
PROPELLANT	1	1150.00	1150.00
MMH TANKS	2	46.67	93.34
NTO TANKS	2	46.67	93.34
HELIUM TANKS	2	3.00	6.00
HELIUM GAS	2	2.23	4.46
MAIN ENGINES	2	120.00	240.00
MISCELLANEOUS			40.00
TOTAL			1627.14 kg

TABLE III.3 - PROPULSION SUBSYSTEM FINAL MASSES

Appendix III-A

ΔV REQUIREMENTS

- FROM ELV TO 430 km:

$r_1 = (408 + 6378) = 6786$ km

$r_2 = (430 + 6378) = 6808$ km

$a = \frac{r_1 + r_2}{2} = 6797$ km

$\Delta V_1 = V_p - V_{c_1}$

$= \sqrt{\mu\left(\frac{2}{r_1} - \frac{1}{a}\right)} - \sqrt{\frac{\mu}{r_1}}$

$= \sqrt{3.986\times10^5 \frac{km^3}{s^2}\left[\frac{2}{6786} - \frac{1}{6797}\right]} - \sqrt{\frac{3.986\times10^5\ km^3/s^2}{6786\ km}}$

$\Delta V_1 = 7.6703 - 7.6641 = 0.0062$ km/s

$\Delta V_2 = V_{c_2} - V_a$

$= \sqrt{\frac{3.986\times10^5\ km^3/s^2}{6808\ km}} - \sqrt{3.986E5\left(\frac{2}{6808} \cdot\right.}$

$\Delta V_2 = 7.6517 - 7.6455 = 0.00618$

$\Delta V_{TOT} = \Delta V_1 + \Delta V_2 = 6.2 + 6.18 = \boxed{12.38}$

- FROM STATION TO ATMOSPHERE:

ONLY ONE THRUST REQUIRED: $\Delta V = \sqrt{\frac{3.986\times10^5\ km^3/s^2}{6808\ km}} - \sqrt{3.986\times10^5\left[\frac{2}{6808} - \frac{1}{6643}\right]}$

$r_1 = 6808$ km

$r_2 = (110 + 6378) = 6488$ km

$a = \frac{6808 + 6498}{2} = 6643$ km

$\Delta V = 7.6517 - 7.5326$

$\boxed{\Delta V = 119.1\ m/s}$

NOTE: THE ΔV REQUIREMENTS FOR REACHING THE CO-ORBITING PLATFORMS ARE SMALL; A TOTAL ΔV REQUIREMENT OF 28.5 m/s WAS DECIDED UPON FOR EACH VEHICLE. THIS AMOUNT SHOULD BE LARGE ENOUGH TO ALLOW FOR 3-5 ORBIT TRANSFERS PER VEHICLE EVERY 90 DAYS, WHICH SHOULD BE MORE THAN SUFFICIENT.

PROPELLANT TANK SIZING

$\rho_{TANK} = 4.7 g/cm^3 = 4700\ kg/m^3$

- MONOMETHYLHYDRAZINE TANK:

$\rho = 0.879\ g/cm^3$

$V_i = \frac{M}{\rho} = \frac{434000g}{.879 g/cm^3} = 493793\ cm^3$

Divide into 2 tanks → $\frac{V}{2} = 246871\ cm^3 = \frac{4}{3}\pi r_i^3$

$r_i = \sqrt[3]{\frac{246871(3)}{4\pi}} = 38.92$ cm

Tank thickness chosen arbitrarily to be 5mm thick. Then choose r_o of tank to be 40 cm.

$V_{TANK} = \frac{4}{3}\pi\left[(.400m)^3 - (.395m)^3\right]$

$V_{TANK} = 0.009928\ m^3$

$m = \rho V = \left(4700 \frac{kg}{m^3}\right)(.009928\ m^3)$

$\boxed{m_{MMH} = 46.67\ kg}$

- NITROGEN TETROXIDE TANK:

$\rho = 1.45 g/cm^3$

$V_i = \frac{716000g}{1.45\ g/cm^3} = 493793 cm^3$

$\frac{V}{2} = 246871\ cm^3$

$r_i = \sqrt[3]{\frac{246871(3)}{4\pi}} = 38.92$ cm

$V_{TANK} = \frac{4}{3}\pi\left[(.400m)^3 - (.395m)^3\right]$

$V_{TANK} = .009928\ m^3$

$m = \rho V = (4700\ kg/m^3)(.009928\ m^3)$

$\boxed{m_{NTO} = 46.67\ kg}$

- HELIUM TANK:

Need 2.23 kg Helium for propellants

$\rho_{KEVLAR} = 1.44 g/cm^3$

$\rho_{He} = 0.038 g/cm^3$

$V_i = \frac{2230g}{.038 g/cm^3} = 58617 cm^3$

$\frac{V}{2} = 29308\ cm^3$

$r_i = \sqrt[3]{\frac{29308(3)}{4\pi}} = 24.10$ cm

$r_o = 24.60$ cm

$V_{TANK} = \frac{4}{3}\pi\left[.246^3 - .241^3\right] = .002083\ m^3$

$m = \rho V = (1440\ kg/m^3)(.002083\ m^3)$

$\boxed{m = 3.00\ kg}$

IV Attitude and Articulation Control System(AACS)

A. SYSTEM REQUIREMENTS

1. Provide 6 degree of freedom control and stability for FREES.
2. Provide guidance and navigation for FREES.
3. Have speed and accuracy for autonomous rendezvous.
4. Operate under positive Space Station control.
5. Examine the loading and unloading of payload

*6. Control of pointing devices (scan platforms, antennae)

*FREES will have no scan platforms and the antennas do not need pointing devices.

B. SYSTEM OVERVIEW

In order to comply with all of the above requirements the following system has been constructed. The following is a list of the system's needs and components used to meet those needs.

1. Rotational control: 4 (250N) control moment gyroscopes(CMG) .
2. Translational control: 12 cold gas 111N thrusters, 2 nitrogen tanks, valves and piping.
3. Inertial guidance: 3 fiber optic gyroscopes(FOG) and 3 fiber optic accelerometers(FOA).
4. Celestial guidance: 1 standard star tracker(SST) and 1 conical earth sensor(CES).
5. Automated docking: 2 guide beacons.
6. 1 computer to integrate the AACS with the other systems of FREES and the Space Station.
7. Payload loading and unloading plan.

The total system weight is 186 kg.

The average system power requirement will be 300 W.

These totals can be broken down in the following fashion:

Component	Weight(kg)	Power(W)
4 CMG	40	360*
12 Thrusters, tanks, fuel	126	5
3 FOG	3	6
3 FOA	3	6
Star Tracker	8	7
Earth Sensor	2	2
2 beacons	4	60
Computer**		

* This is the power that would be required if 3 CMGs were used at maximum power. This would be a very unusual case and is not used in computing the average power required.

** The computer was not included in AACS weight and power calculations. It will be accounted for in the Command and Data Control system.

This was an overview of the AACS and its components. Figure IV.1 (placed at end of section) shows where these components will be located on the FREES vehicle. Now a more detailed look at each of the components.

C. SYSTEM COMPONENTS

Rotational Stability:

This is accomplished through the use of the CMGs. They are placed so that there is a CMG on each of the geometric pitch, roll, and yaw axes. The fourth CMG will be placed at 30 degree angle to the three axes. This will allow the system to still function if one of the CMGs ceases to operate. A combination of the skew CMG and one of on axis CMGs will take the place of the malfunctioning CMG. Figure IV.2 show the arrangement that was just discussed. Also Figure IV.3 gives an example of how the on axis CMGs are integrated into the rest of the system.

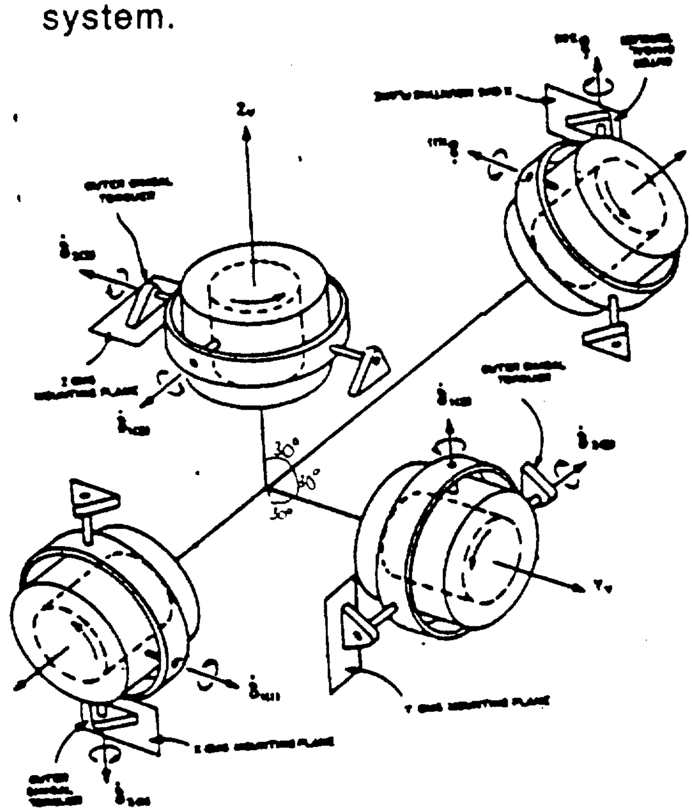


Figure IV.2

(Chobotov, p.30)

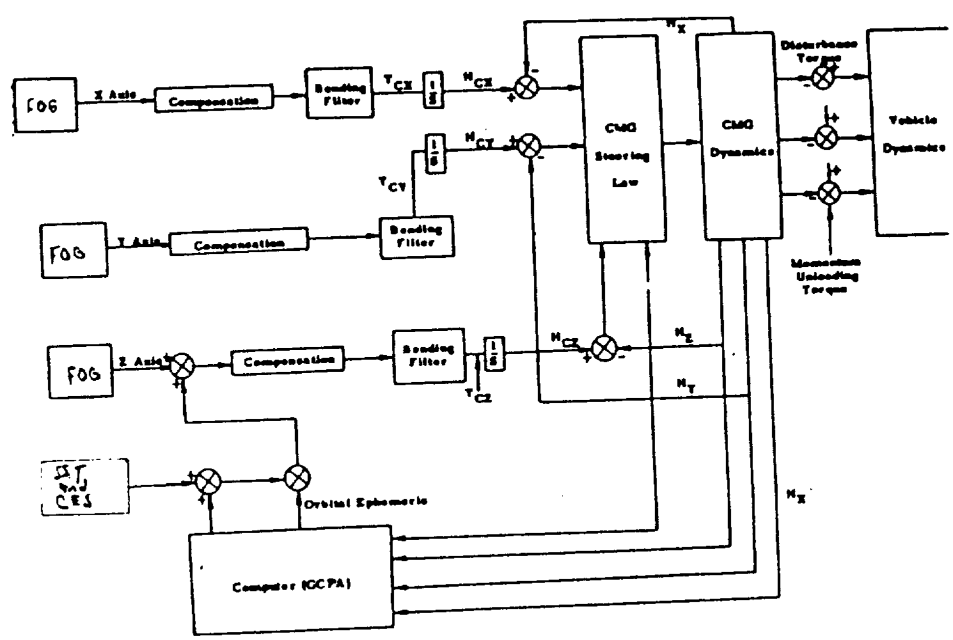


Figure IV.3

(Chobotov, p.31)

The reasoning behind using CMGs is their large torque and low power as compared to reaction wheels, this is show in Figure IV.4. The CMGs also have a smaller weight and longer lifetime than a mass expulsion system.

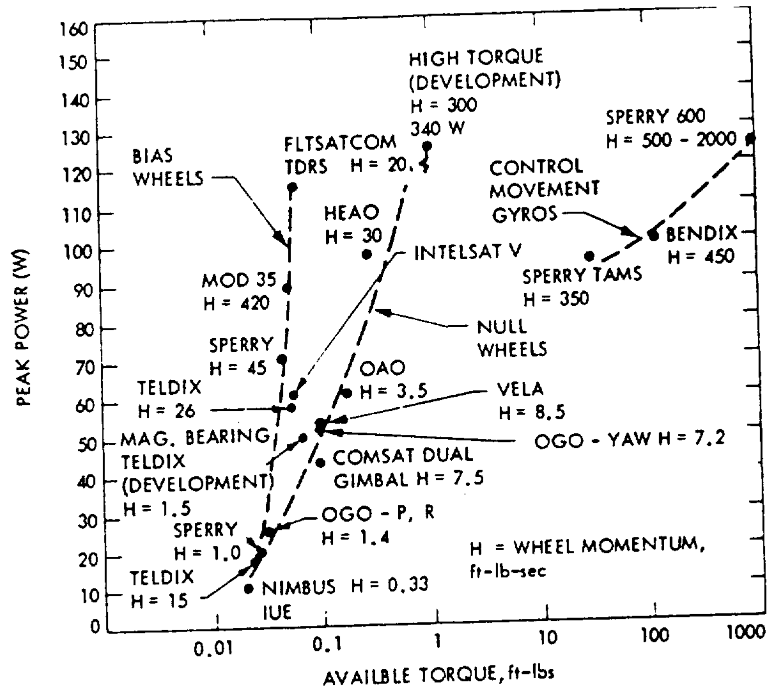


Figure IV.4

(Journal of Guidance and Control 1980, p. 259)

In order to establish what the actual torque needed, the following set of equations were used. These equations gave an approximation of the vehicle's angular acceleration and time it would take to rotate the vehicle 180 degrees.

$A = T/I$ where: A is angular acceleration of the vehicle.
T is the torque applied by the CMG.
I is the moment of inertia of the vehicle.

$t = 2(pi/A)^{.5}$ where: t is the time it take the vehicle to rotate 180 degrees and have an angular velocity of 0 m/sec.
pi= 3.141529
A is angular acceleration of the vehicle.

The moments of inertia were approximated to be
$I_x = I_y = 88113.75$ kg-m^2 $\qquad I_z = 43560$ kg-m^2

Taking a range of torques for 10N to 1000N the following plot (Figure IV.5) was created. Similar plots about the pitch and yaw axes will yield the same results.

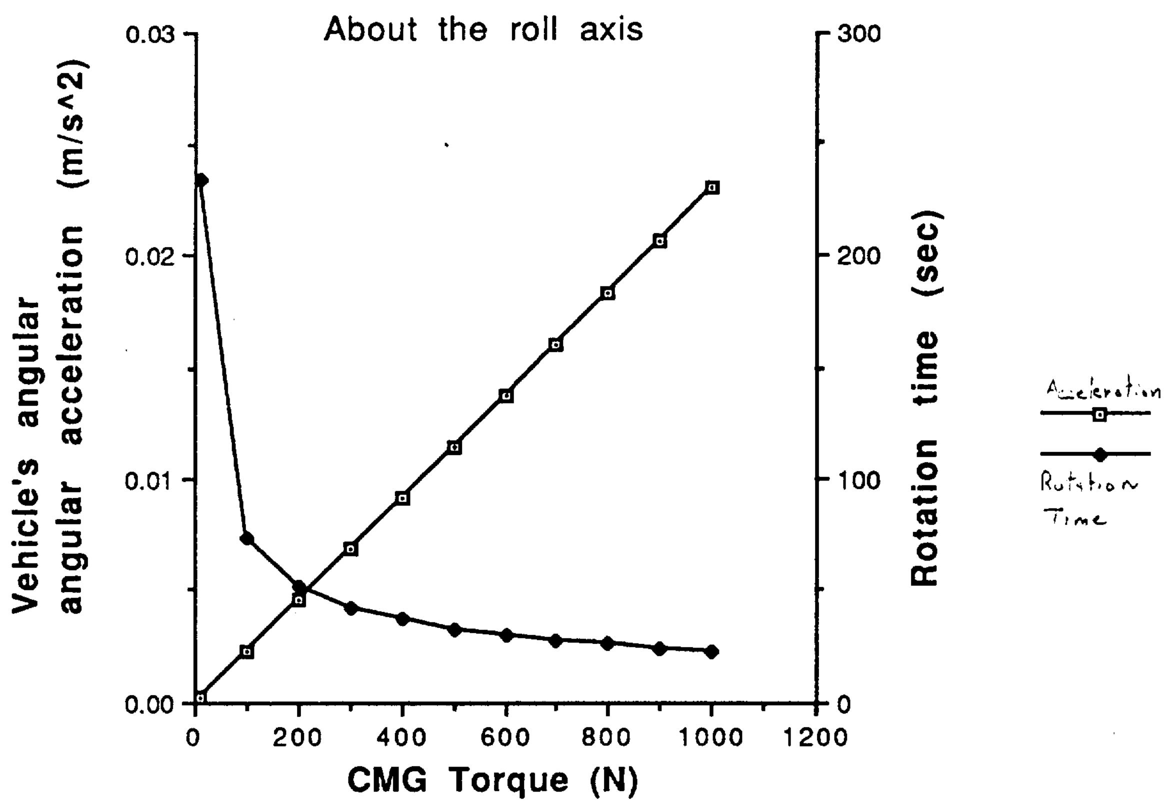


Figure IV.5

The intersection of the graph provides the location of the best time with the lowest angular acceleration. This was used to pick the CMG torque with a torque capability of 250 N and thereby created a more efficient system. It should be noted that the vehicle's rotation time more than meets the time requirements of automated docking and reentry.

Translational Stability:

Translational stability is accomplished through the use of 12 cold gas thrusters. The positions of which were shown in Figure IV.1. The system will use cold gas in compliance with a Space Station requirement. A combination of cold and hot gas will not be used so that the system complexity can be kept somewhat low. This system will keep the vehicle at the proper orientation. main engines. The following plot in Figure IV.6 shows the relationship between several cold gases densities, Isp's, and atomic weights.

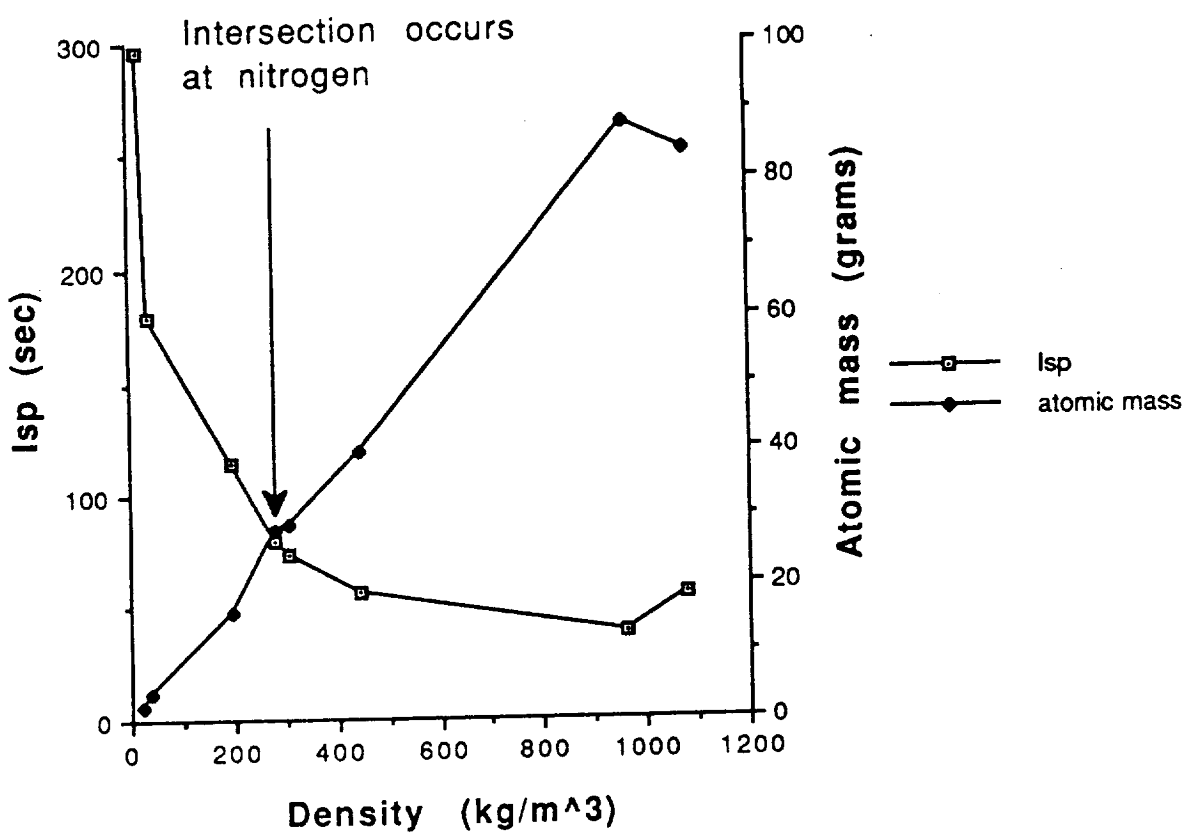


Figure IV.6

By examining the plot it can be seen that the nitrogen point is at the intersection of the two curves. This means that nitrogen will give the highest Isp at a high density. The second advantage of using nitrogen is that it gives the system the ability to use the same tanks as ECLSS.

Next using the equation massflow = F/gIsp Figure IV.7 was created. where: F is thrust in newtons

$g = 9.8 m/sec^2$

Isp= 80 sec

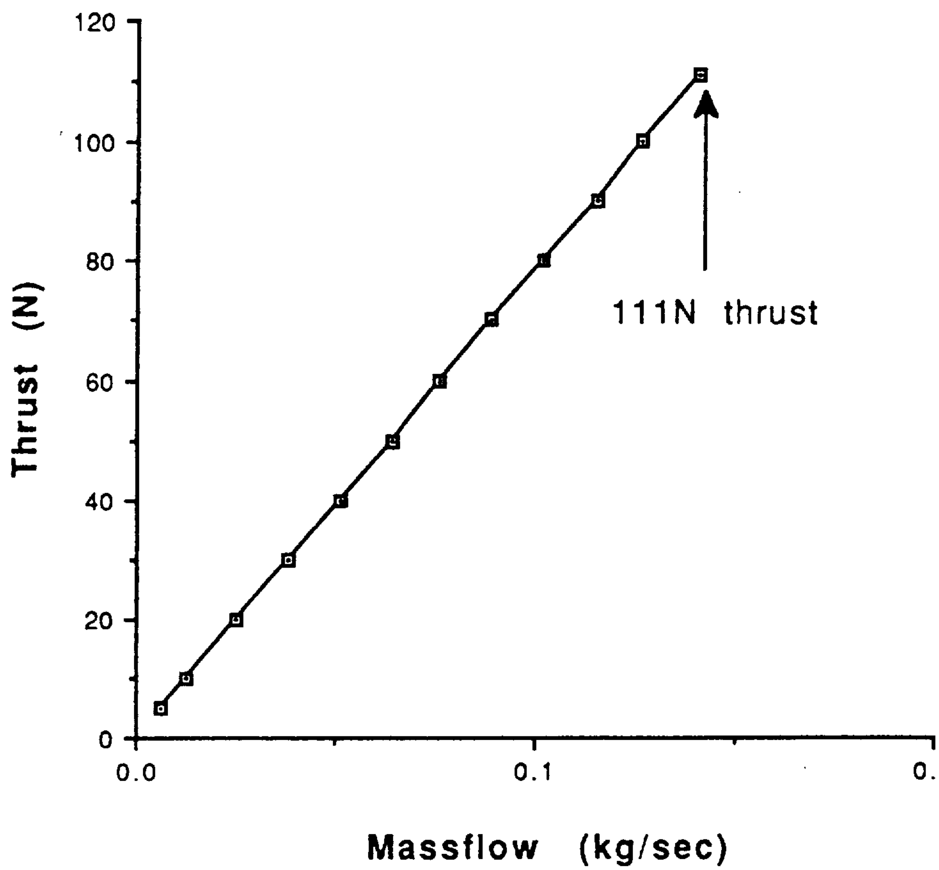


Figure IV.7

The 111 Newton thruster was chosen because of its ability to provide the needed delta v to maintain the vehicle's attitude. It is also the same thruster as the vernier thruster that is used by the Space Shuttle. The use of this thruster will help to decrease the cost of the system and will also provide FREES with a flight proven system. Figure IV.8 shows the relationship between time the the delta v achieved and the amount of fuel required by the system.

t(sec)	delta v (m/sec)	mass of fuel (kg).
1	.00615	.1413
5	.031	.7065
10	.0616	1.413
20	.123	2.83
30	.1846	4.24
40	.2462	5.65
50	.3078	7.07
60	.369	8.49

The fuel lines and valves have been installed so that no one failure will cause the entire system to fail. Also with the thrusters positioned in the way shown in Figure IV.1 they provide yet another redundancy for the CMGs along the pitch and yaw axes. Should all CMGs fail the thrusters would still be able to position FREES for reentry.

Inertial Sensors:

The inertial sensor package is divided into two parts. First there are the 3 fiber optic gyroscopes to measure rotational motion. Second the 3 fiber optic accelerometers to measure translational motion. A FOG and a FOA will be placed on each of the geometric pitch, roll, and yaw axes to measure rotation and translation both about and along the axes.

Although FOGs have not been flight proven they will be the gyroscope of choice by the mid 1990's. The FOG offers the advantages of high reliability, low cost, small size, low weight, long life, and high accuracy. Current ring laser gyroscope technology cannot compare to what FOG will offer.

In a similar fashion the FOA promises to offer several distinct advantages over currently used accelerometers. The FOA's light weight, small size, long life, and high accuracy makes it the best choice for what is required by the FREES vehicle. The long life and high accuracy will be need so that the FREES system can dock and release quickly and also so that it will be able to last for the required amount of 6 years.

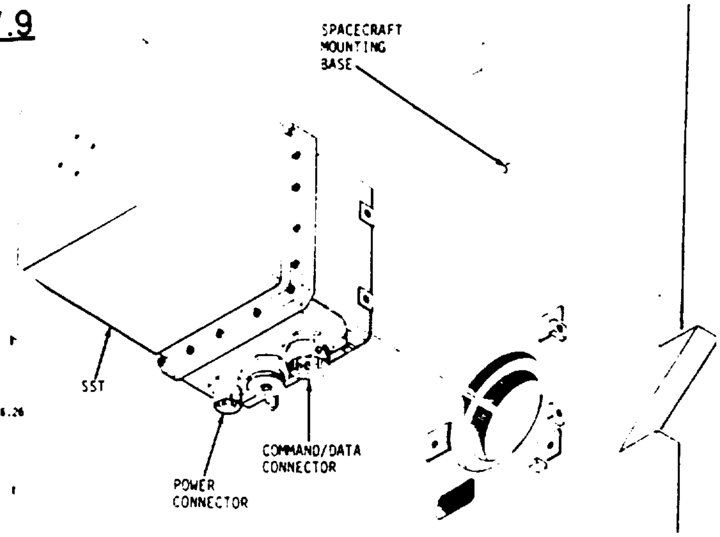
Celestial Sensors:

The celestial sensor package consists of a star tracker and an earth sensor. These sensors are used to provide the inertial sensors with their initial values. After that the celestial sensors will serve not only as a backup for the inertial sensor package but also as a constant reference point for them.

The star tracker chosen was the BASD Standard Star Tracker (SST). The SST shown in Figures IV.9 and IV.10 has been qualified for use by the Space Shuttle. Figure IV.11 shows the operating parameters of the SST.

Figure IV.9

SST/Spacecraft Interface



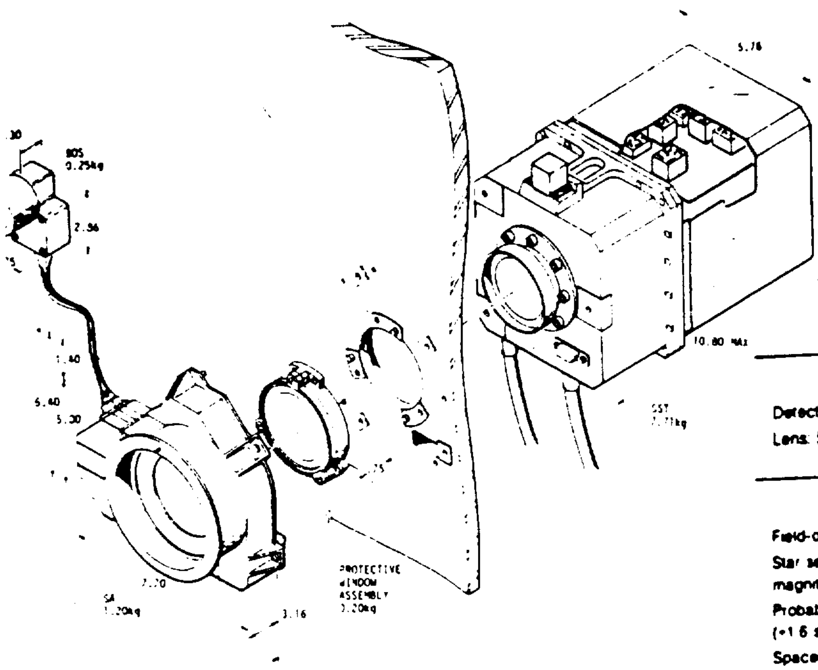


Figure IV.10

ELECTRO-OPTICS

Detector: Two element silicon diode array
Lens: 50 mm, f/1.4

PERFORMANCE

Field-of-view: 12° elevation
Star sensitivity: +1.6 to -1.4 silicon magnitude (8 commandable thresholds)
Probability of Detection: 0.99 (+1.6 silicon magnitude)
Spacecraft Spin Rate: 12 to 8 rpm
Sensor Optical Axis: 14° from vehicle spin axis
Sun Shade Protection: ±70°
Power Consumption: Less than 2.0 Watts from 28 VDC

Accuracy

Position: ±10 arc minutes (1σ)
Star Intensity: ±0.25 magnitude

MECHANICAL CONFIGURATION
(including sunshade)

Size: 15" × 14" × 7.875" (381 mm × 356 mm × 200 mm)
Weight: 6 lbs (2.7 Kg)

Figure IV.11

An earth sensor will be used in connection with the SST. The reasoning behind this is that for LEO the earth is the second brightest object in the sky. FREES will use a conical earth sensor (CES) which will be able to be used in any orbit required of the vehicle. Figure IV.12 gives a specification summary of the CES and Figure IV.13 shows the scan pattern of the CES.

APPLICATION	Two axis attitude determination for spacecraft		
ALTITUDE RANGE	100 km to super-synchronous		
ACCURACY	Geosynchronous < .05° Low Orbits* < .10°		
	SENSOR	ELECTRONICS	TOTAL
SIZE	4.0" x 3.0" dia.	6" x 7" x 3"	
WEIGHT	2.1 lbs (0.95 kg)	3.4 lbs. (1.55 kg)	5.5 lbs (2.5 kg)
POWER	4 Watts	4 Watts	8 Watts

Figure IV.12

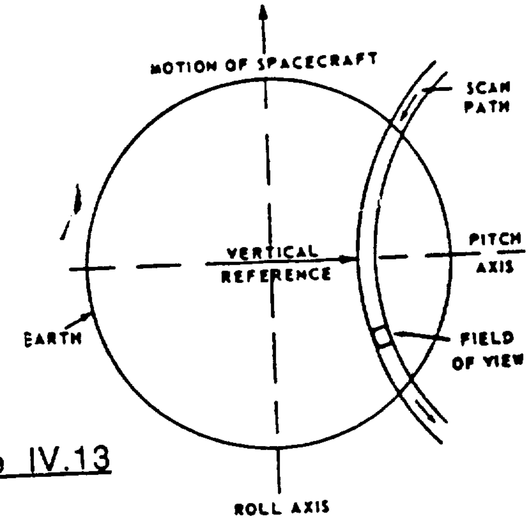


Figure IV.13

The celestial sensors play a major role in the control of FREES's attitude. At least one of the two will remain active while the vehicle is docked to the Space Station. So when the vehicle releases from the Space Station it will already have an accurate understanding of its orientation. This will allow for quicker missions to the free flying platforms and for returns to earth.

Guide Beacons:

These are discussed by Command Data Control system.

Payload loading and unloading:

As the system requirements stated AACS was charged with the responsibility of creating a system for loading and unloading the payload. Due to the way in which the payload will be distributed throughout the FREES vehicle, see the structures section for a description of this, and also due to the weight and volume limitations it was decided to use a completely manual system in space and a manual with machine assisting on earth. This decision was made based on examination of data and drawings of the FREES craft. It is suggested that further studies be undertaken when an actual prototype vehicle has been built.

Do to the weightlessness of space an entire manual system presents no real problems. In fact, it offers several weight and power advantages over any other system. On earth, where there is gravity, a system which uses heavy lifting equipment must be use. This equipment will be used to place the heavier payload into the vehicle where ground personnel will make final placement adjustments.

AACS Computer:

While the specifications for the AACS computer will be given in the Command and Data Control system here is a summary of why the computer will be needed. The computer will serve as the hub of AACS, see Figure IV.14 All of the sensors and attitude control devices will feed information into the computer and receive commands and data from it. The computer will serve as the means of integration between AACS the other systems and the Space Station.

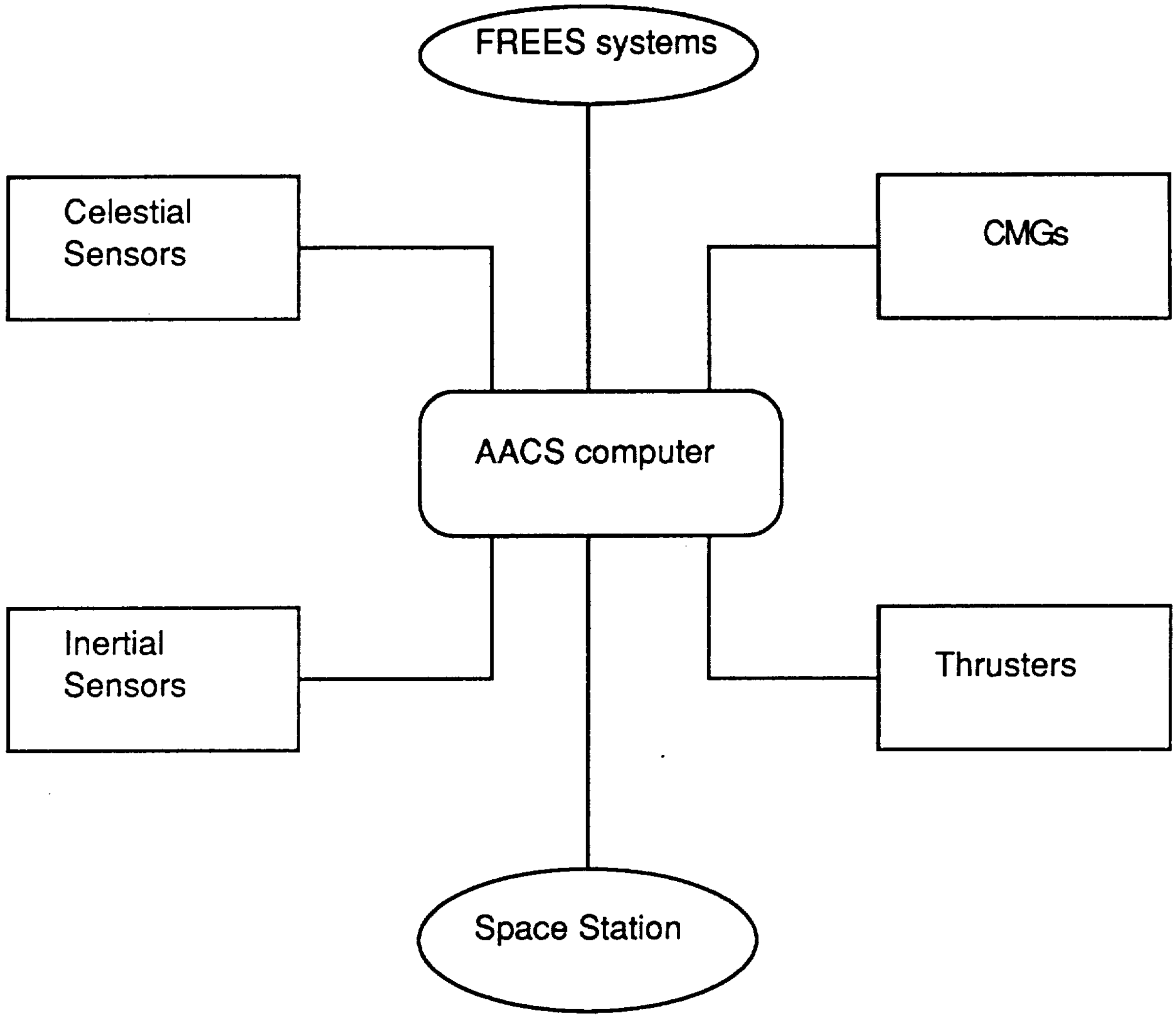


Figure IV.14

D. INTEGRATION WITH OTHER FREES SYSTEMS

The computer section mentioned that AACS will be connected to FREES' other systems. AACS will have to interact with these systems on several different parts of the mission. The following is a list of the systems which AACS must interact with during the mission.

1. Power and Propulsion: It is from this system that AACS will receive its power. Also, AACS and PPS will work together in

providing the necessary delta v for vehicle adjustments during orbit.

2. Command and Data Control: AACS is linked to this system through the computer and must remain linked to receive data from all the other systems. This link also provides AACS with the ability to contact the Space Station during the mission.

3. Reentry and Recovery: During the descent and reentry of the vehicle AACS must remain in contact with RRS in order to keep the vehicle at the proper attitude for use of its heat shield.

4. Emergency Crew Life Support: AACS is connected to ECLSS through the cross feeds of the nitrogen tanks. These cross feeds allow both systems to borrow nitrogen from each other if the need should ever arise.
5. Structures: The entire attitude and articulation control system must be integrated into the structure of FREES. It had to be placed in such a way so as not to weaken the structure or place any excess strain on it.

E. CONCLUDING REMARKS

This attitude and articulation control system meets all of the requirements that were presented to it by the request for design. The system which consists of 4 CMGs, 3 FOGs, 3 FOAs, 1 SST, 1 CES, 2 guide beacons, 12 cold gas thrusters, and a computer that is capable of high accuracy. It also meets all time requirements presented to it by FREES's other systems. The payload loading and unloading scheme that has been studied needs to be examined further when a prototype vehicle is constructed. This system will help bring in a new generation of U.S. space travel.

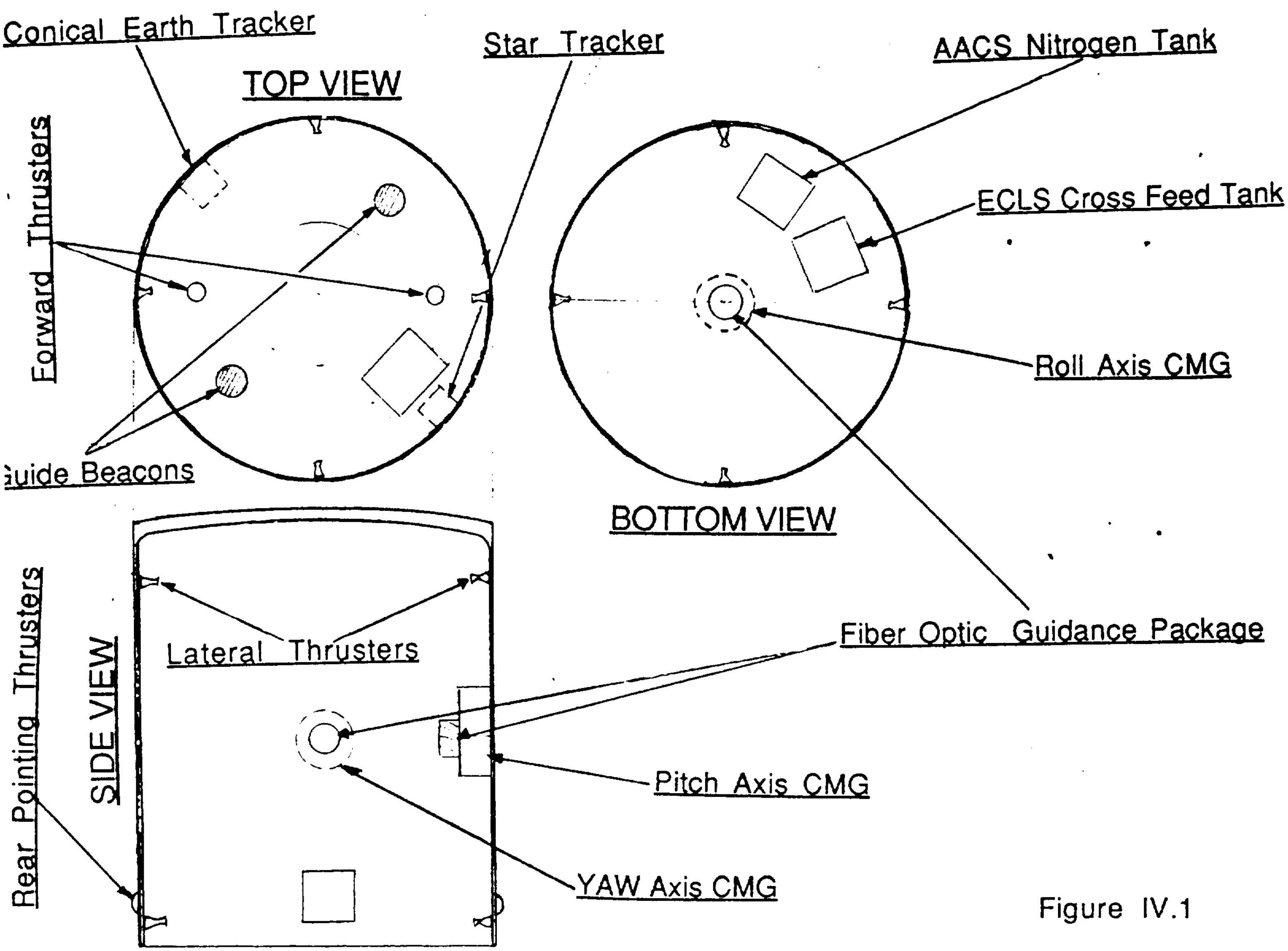


Figure IV.1

V. ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM

A. EXPLICIT AND DERIVED ECLSS REQUIREMENTS

The ECLSS system is responsible for the life of the crew. It also needs to provide for pressurized cargo transport. ECLSS should provide living space, consumables, temperature control, humidity control, and fire suppression for FREES. These are the major requirements of this system, however there are also many derived requirements that need to be filled.

The derived requirements come from many sources. One is the need for an easy low-g entry because of the possibility of bringing back an injured crew member. The environment of the module must be compatible with the Space Station and no EVA should be required to enter the module from the station. Pressure suits of some kind should be available to the crew. Also, the atmosphere of FREES needs to be recyclable. Finally, the module should be able to rapidly separate with the station.

B. CAUSES FOR USING **FREES** AS A MANNED VEHICLE

The primary manned use of FREES will be to travel to various platforms and perform maintance duties. However, the module will also serve the crew of the station in an emergency. There are many potential events that could cause this utizilation of the module. For example, a fire may cause the evacuation of the station. Another threat would be an injured or ill astronaut, or a loss of pressure in the station. Other threats include an out of control astronaut or an out of control Space Station (Lembeck, Noteset 238.04).

These events are what FREES needs to be capable of responding to and they define the need for a simple and nearly fail-safe system. In a sense the ECLSS allows the module to act as a lifeboat for Space Station Freedom.

C. SIZING

The Logistics Resupply Module needs to be properly sized for both payload and the crew missions. The primary mission of the LRM is to deliver payload to the station, however it will also be required to perform several manned missions. Some of these missions include carrying payload to free-flying platforms and acting as a lifeboat for the Space Station.

From Sections I-Mission Planning and III-Propulsion it was determined that the return trip to Earth will take a maximum of 6 hours. Transfers to the nearby free-flyers will require much less time to perform; hence the ECLSS design is basically independent of these trips since tanks can be refilled at the station following these missions. The maximum mission time would occur if the crew were forced to evacuate the station during an atmosphere-purging event such as station contamination. This time is estimated to be about 18 hours.

Summarizing, it is apparent that the longest required mission would take 18 hours. This mission requires FREES to support eight crewmen while the Space Station's atmosphere or other error is being corrected. A safety margin of 6 hours is added to this mission length to enable the crew to return to Earth if the Space Station is still uninhabitable after 18 hours. Therefore, a mission length of 24 hours has been selected as a sizing parameter.

Another parameter to be decided is the required volume and its most efficient usage. Figure V.2 shows a graph of the payload volume and the volume required for acceptable crew performance against the number of crew members. The payload volume per module was determined in Section I by Mission Planning and is half of the total payload volume. The crew volume required was calculated using the acceptable crew volume equation given in AAE 241 lecture (Lembeck, Noteset 238.05). A mission time of 24 hours, as found above, was used and the crew size was varied from two to eight members. As the graph shows, the payload volume is much larger than the greatest required crew volume. Thus, it would be a most efficient use of space to design the module to support eight men. A distinct advantage of this design choice is that for a total evacuation of the Space Station, only one vehicle is required. Also, if the Space Station is upgraded to support a larger crew, two vehicles would be able to maintain the emergency return capability.

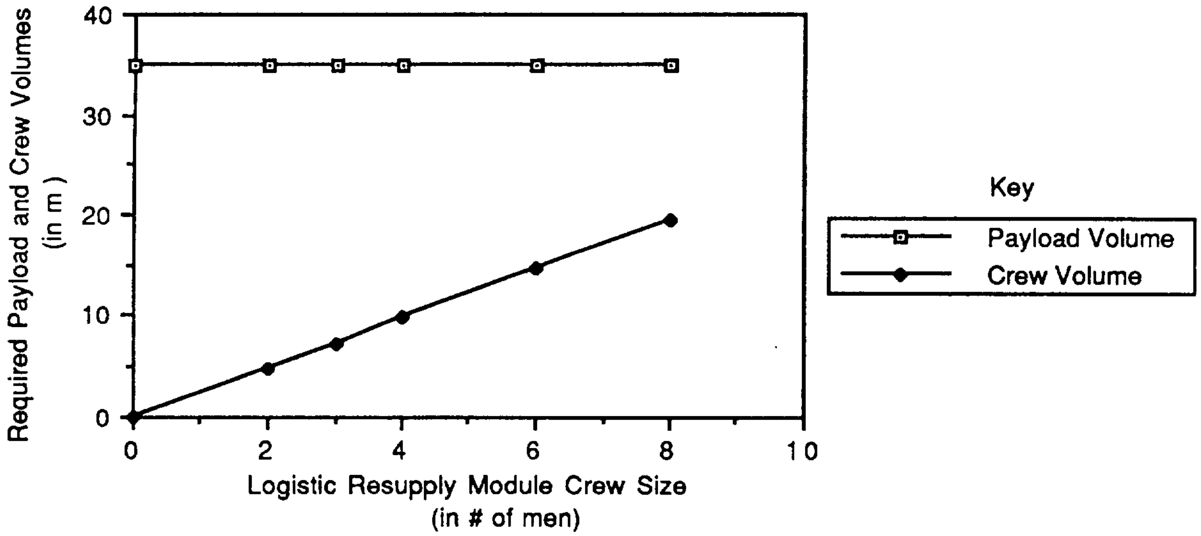


Figure V.1, A Study of Crew and Payload Volumes

B. ATMOSPHERE SUPPLY AND CONTROL SYSTEM

The Atmosphere Supply and Control System, or ASCS, is the source for the atmosphere in the module. The ASCS provides an atmosphere of nitrogen and oxygen which resembles what is found on Earth. The atmosphere is maintained at a pressure of 14.5 to 14.9 psia with an O_2 partial pressure of 2.83 to 3.35 psia (Humphries, p. 3). This is roughly the 21% O_2 rich atmosphere of Earth. The system will monitor the O_2 partial pressure and regulate it by leaking an N_2-O_2 mixture into the cabin. This system also provides for the repressurization of the cabin three times when filled to capacity.

The N_2 and O_2 will be stored in "Kevlar" 49 tanks at 3000 psia. They will be fed through high pressure regulator valves into a mutual tank at 100 psia. The gases will be leaked from this lower pressure mixing chamber into the Atmosphere Resupply System's ducts and sent into the cabin. Four such N_2-O_2 assemblies will be located in the vehicle; two on each side. This provides for redundancy on both sides of the system. Also, the tanks will have crossfeeds with back-flow check valves to keep the tank levels equal and to allow a tank which has been shut off from the system, due to a fault, to drain into another tank.

These tanks were sized according to the procedure given in AAE 241 lecture (Lembeck, Noteset 238.05). The detail calculations of this procedure are found in Appendix V-A. Note that, due to the units in the equations, the analysis was done in English units and converted to metric. The results are listed in Figure V.1.

The 100 psia mixing tank will be one foot in diameter and also made of "Kevlar" 49. Due to its much lower pressure it will have a thickness of 1.27 mm.

FREES is also capable of rapidly being depressurized and repressurized. It uses a release valve with a pump, that is located about at the middle of the module, to vent the atmosphere. The module is then repressurized using the oxygen and nitrogen tanks of ASCS. The approximate rates for decompression and compression are 4.0 psi/min (Lembeck, Noteset 238.05). The crew will be provided with oxygen masks to use during this time. The tanks for the oxygen masks are sized in Appendix V-A and are 12.7 cm in diameter and 30.35 cm high.

The ASCS system is used on nearly all manned spacecraft including Skylab, Salyut, Soyuz, and the Space Shuttle (Bolger, pp. 90-3). Figure V.2 shows a schematic layout of half of the system. The other half would be housed in the opposite wall of the vehicle.

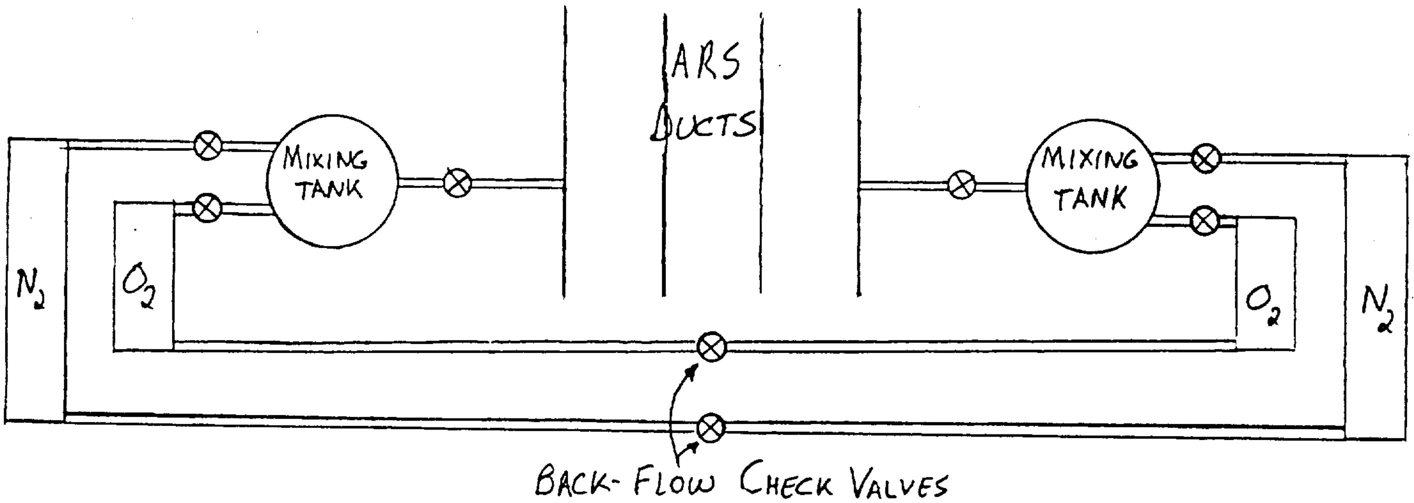


Figure V.2, ASCS System Schematic

	O_2	N_2
WEIGHT OF TANKS (kg)	2.93	5.91
WEIGHT OF GAS (kg)	13.19	38.10
TOTAL FULL TANK WEIGHT (kg)	16.12	44.01
ASCS PIPING WEIGHT ESTIMATE	42.41 kg	

Figure V.3, ASCS Tank Sizing Summary

Figure V.3 is a summary of the results of Appendix V-A. This system is proven reliable and has several safety features. One feature is power close O_2 valve and power open N_2 valve. This feature, which requires power to close the O_2 valve, ensures an oxygen supply even if all power should fail. Also, oxygen partial pressure sensors are placed throughout the cabin and Atmosphere Resupply System's vents.

D. ATMOSPHERE REGENERATION SYSTEM

The Atmosphere Regeneration System, or ARS, will process the atmosphere of the cabin and remove water vapor and carbon dioxide. It will also filter contaminants from the air and regulate the temperature of the cabin. The ARS consists of four fans to circulate the air and send it through the ducts to be processed. Other components are filters, LiOH cannisters, heat exchangers, and condensers.

The system is diagrammed in Figure V.4. Note that the fans are positioned so that the forces they exert on the spacecraft cancel each other out. The air will be drawn from the cabin into the ducts in the wall of the module. There, the airflow will be divided in half and sent through nearly identical processing. They will be passed through contamination control filters which will filter particles 0.5 microns or larger (Bolger, p. 90). The air is then passed through LiOH cannisters and one of the ducts passes through a heat exchanger. Both ducts pass through a condensor to remove the excess water vapor and another contamination filter before being returned to the cabin. Figure V.4 also shows the points at which the ASCS leaks its oxygen-nitrogen mixture into the tank.

The LiOH cannisters weigh 5.45 kg apiece and are 21.6 cm in diameter and 33.35 cm high. This weight was determined from an equation given in an AAE 241 lecture (Lembeck, Noteset 238.05). As shown in Appendix V-B, the equation reveals that 10.91 kg of LiOH are necessary for the operation of an eight man vehicle for 24 hours. This figure was doubled for safety and to account for a possible fan failure one of the ARS processing loops. If one half of the ARS system failed there would still be enough LiOH in the other half for 24 hours. Also, two spare LiOH cannisters will be kept on board to replace used or faulty ones. This will allow for possible uses during a visit of the module to another platform.

The heat exchanger will be a closed loop-liquid system. It will fan out into capillaries in the duct and as the air is blown through these tubes, heat will be exchanged. The pipes will be sent to the space between the dual walls and fan out to absorb or radiate heat. The outer dual wall will be louvred with one side capable of absorbing heat and the other capable of radiating it. The weight of this assembly is 20 kg and it is roughly 30 cm by 70 cm by 12 cm.

The system will contain four spare filters, two spare fans, and the two replacement LiOH cannisters in a compartment next to one of the ARS duct networks. All the main components of ARS are accessible to the crew via doors in the inner wall.

Systems very similar to this one are and have been in use nearly exclusively in space. The system varies slightly from program to program, but the major components are always similar. FREES is highly compatible with the Space Station and the only major

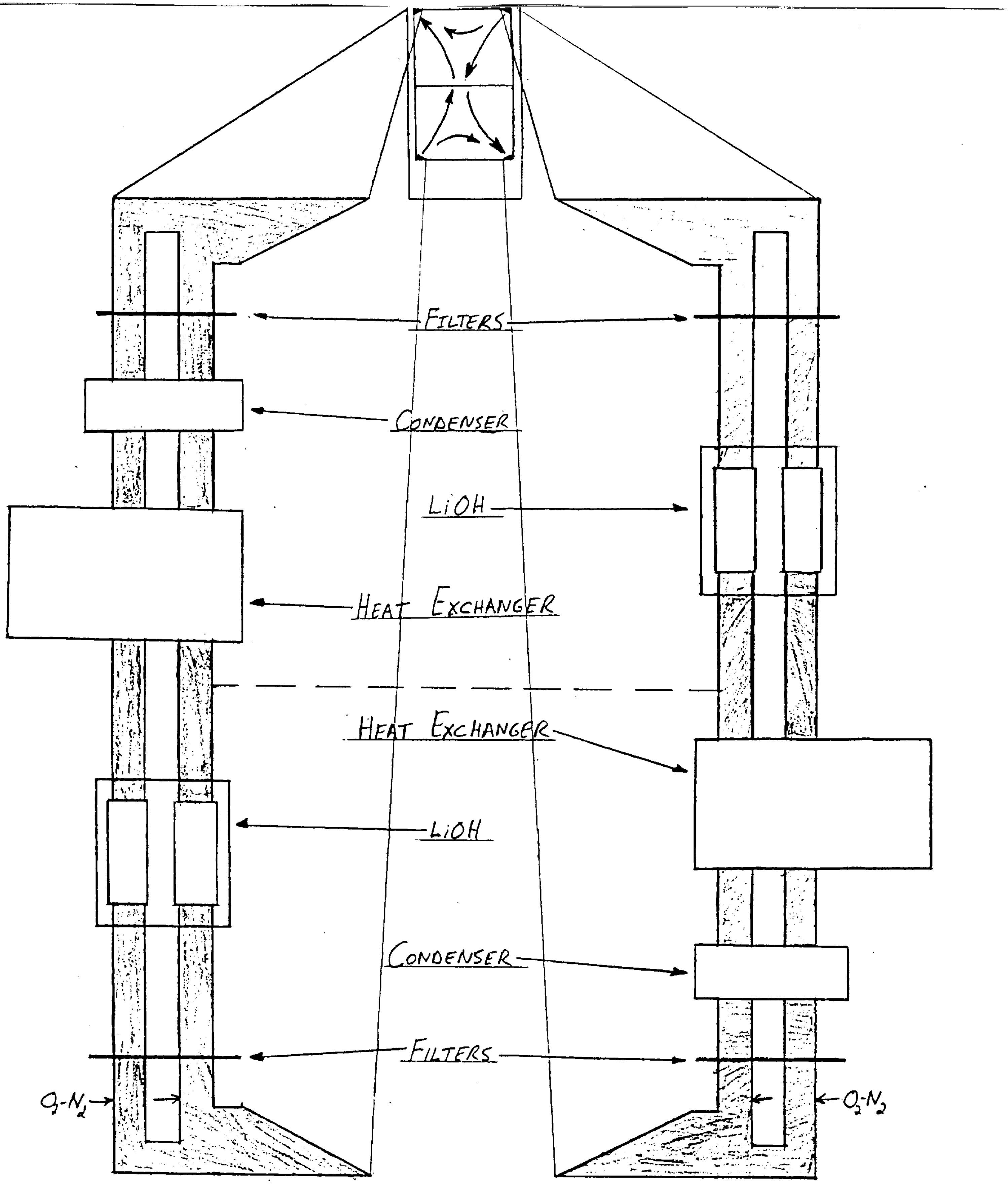


Figure V.4, ARS System Schematic

difference between the two is that the oxygen and water loops are open on FREES. This system is reliable and proven in space (Bolger, pp. 89-93).

E. FIRE DETECTION AND SUPPRESSION

The Fire Detection and Suppression System, or FDSS, is responsible for detecting fires and taking measures or providing means to fight the fire. It consists of a network of heat and smoke detectors and a set of fire extinguishers.

The detectors are connected to the Caution and Warning System, which is designed in conjunction with Command and Data Control. If a detector is triggered, a visual and audible alarm is given in the cabin. Also, the information given by the detector is sent to the ground and to the Space Station. The first response option to a fire is for the crew to attempt to control and extinguish the fire with chemical extinguishers. To aid this effort, the oxygen partial pressure in the cabin can be reduced, the circulation provided by one or two of the fans can be reduced or stopped, and any electrical equipment which may be shorting out can be shut down. Also, the crew may use the oxygen masks for depressurization to protect the crew from fumes and to provide adequate oxygen. If this fails, the last option is to depressurize the cabin.

There are eight chemical fire extinguishers located in the module, one on each couch. There will also be a network of roughly 15 smoke and heat detectors placed strategically throughout the ducts and cabin. FREES' system of chemical extinguishers and detectors is in common use as the simplest and most reliable system available.

F. CABIN SUPPLIES

The cabin has two decks with four couches on each level (see Section II-Structures). The couches are stored upright in the walls of FREES and in storage the front of the couch faces the inside of the cabin. They are pulled from the wall at the bottom and the top slides down a track and spring pins lock it in place. The bottom is the locked in place with spring pins on the floor. Each couch is equipped with an emergency oxygen tank and a fire extinguisher. Also, partial pressure suits will be stored in a compartment in the wall of the module and will be used by the crew during reentry. Figure V.5 shows a couch and the positions of the oxygen tank and fire extinguisher.

The cabin will also provide 30 meal packets for the crew for extended or emergency missions. They will be resupplied at need from the Space Station. Water will be available to the crew from a tank just under the lower deck and urine will be stored for disposal at the Space Station in a tank and pump system located on the lower level in the wall. These systems are also shown in Figure V.6. The water tank shown in the bottom of the module is the same one used for ARS.

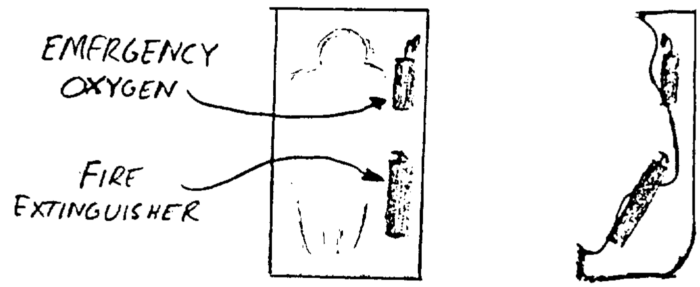


Figure V.5, Crew Couch With Accessories

G. MEDICAL SUPPORT

The medical needs that should be addressed for FREES are very similar to those of a modern ambulance. The two purposes it serves are analogous to that of an ambulance. First, FREES would be required to recover an injured astronaut and return him to the Space Station and secondly, to transfer a stabilized injured astronaut to Earth. Therefore, the module should be supplied like an ambulance.

FREES will contain a respirator, an IV system, a Heart Monitor, a portable suction unit, a medical kit, a litter and other minor medical supplies. The respirator weighs approximately 9.0 kg and the IV system weighs about 4.5 kg. The Heart Monitor weighs 8.5 kg, the suction unit is about 5.5 kg, and the medical kit is 11.5 kg. Also, the litter weighs about 13 kg. Spare supplies such as extra solutions for the IV system and extra medicines will also be stored in compartments in the wall (O'Donnell, paramedic).

H. CAUTION AND WARNING SYSTEM

The Caution and Warning System, or CWS, is a network of sensors which monitor oxygen partial pressure, carbon dioxide partial pressure, air flow, humidity, temperature, smoke, and other parameters. The sensors are connected through Command & Data Control's computer to a control panel located on top of the racks on the lower level. This panel will alert the crew to a problem both visually and audibly. It will also send such information to the ground and the Space Station.

ORIGINAL PAGE IS
OF POOR QUALITY

APPENDIX V-A

TANK SIZING

Consider the total volume of $= 165\,m^3 = 5830\,ft^3$

Volume O_2 $(21\%) = 1224.4\,ft^3$

$+$ vol. O_2 consumption $=$ $16.64\,lb \Rightarrow 204.9\,ft^3$ @ 14.7 psia

total vol. O_2 @ 3000 psia $= 7.0036\,ft^3$

Volume N_2 $(79\%) = 4606.0\,ft^3$

$+$ vol N_2 leakage $= 79.2\,lb \Rightarrow 111.4\,ft^3$ @ 14.7 psia

total vol. N_2 @ 3000 psia $= 23.115\,ft^3$

Design four equal tanks of each

vol. O_2/tank $= 1.7509\,ft^3$

vol. N_2/tank $= 5.7788\,ft^3$

Use cylindrical tanks of radii 6 in. for O_2 and 9in. for N_2

$vol = \pi r^2 h = 1.7509\,ft^3$

$h = 2.23\,ft$

$= 68.0\,cm$

$vol = \pi r^2 h = 5.7788\,ft^3$

$h = 3.27\,ft$

$= 1.00\,m$

Calculate maximum tangential stress

$$\sigma_{t_{max}} = \frac{3000(6)}{t} + 1500$$

$$\sigma_{t_{max}} = \frac{3000(9)}{t} + 1500$$

Using a safety factor of 3.

Yield Strength (Kevlar" 49) $= 300{,}000\,lb/in^2$ (Zweben, p. 6)

$3(\sigma_{t_{max}}) = 300000\,lb/in^2$

$t = .061''$

$3(\sigma_{t_{max}}) = 300{,}000\,lb/in^2$

$t = .091''$

Compute weights

$$Vol_{tank} = h\pi\left[(r+t)^2 - r^2\right] + 2\pi(r+t)^2 t$$

$\rho_{Kevlar\ 49} = .052\,lb/in^3$ (Zweben, p. 6)

O_2 tank weight $= .052\,lb/in^3\,(124.09\,in^3)$

$= 6.453\,lb = 2.93\,kg$

N_2 tank weight $= .052\,lb/in^3\,(250.20\,in^3)$

$= 13.010\,lb = 5.91\,kg$

WEIGHT OF O_2 GAS = 29.01 lb = 13.19 kg

weight of N_2 Gas = 83.83 lb = 38.10 kg

Size of emergency O_2 tank

Assume radius = 2.5 in

length of operation = 52 minutes

mass of O_2 = .0753 lb

Vol @ 14.7 psia = .9279 ft^3

Vol @ 100 psi = .1364 ft^3

vol = $\pi r^2 h$ = .1364 ft^3

h = 1.0 ft = 30.35 cm

Size of Aluminum piping

two sizes

$r = 4''$

$V = \ell \pi ((4.0625)^2 - 16)$

$= 1.5831 \ell$

$V = 35 (1.5831) \frac{12 \text{ in}}{1 \text{ ft}}$

$= 664.9 \text{ in}^3$

$r = 5.66''$

$V = \ell \pi ((5.7225)^2 - (5.66)^2)$

$= 2.2349 \ell$

$V = 10 (2.2349)(\frac{12 \text{ in}}{\text{ft}})$

$= 268.2 \text{ in}^3$

weight = ρ (vol)

$= 1 \frac{\text{lb}}{\text{in}^3} (933.1 \text{ in}^3)$

$= 93.31$ lb = 42.41 kg

APPENDIX V-B

LiOH CALCULATIONS

$$3.0 \frac{lb}{man\cdot day} = \text{LiOH use}$$

FOR 8 MEN, FOR 1.0 DAYS

$$\text{LiOH use} = \left(3.0 \frac{lb}{man\cdot day}\right)(8\ \text{MEN})(1.0\ \text{DAYS})\left(\frac{1.0\ kg}{2.2\ lb}\right)$$

$$= 10.91\ kg$$

VI. COMMAND AND DATA CONTROL

Antennae

Two sets of antennae are used to provide communication to the ground and the Space Station simultaneously. Ground communication will be conducted using S-band frequency via the Tracking and Data Relay Satellite System (TDRSS). Ku-band frequency via TDRSS will be used for communication to the Space Station. A configuration of four flush mounted, parabolic antennae is used for each frequency. The antennae are located at a separation of approximately ninety degrees in two separate rows. Selection of the proper antenna to be used is controlled by the CERV's main computer with a two degree dead band between the quadrants to avoid switch chatter. Physical characteristics are shown in Appendix VI-A.

Appendix VI-A

Size : 0.60 meters diameter

Material : 0.01 meters thick aluminum

Mass : 3.5 kilograms total each

Power: 25 Watts transmitting

Ku-band frequency : 13.4-14.9 GHz

S-band frequency : 1.55-5.2 GHz

Automated Rendezvous and Docking

The automated rendezvous and docking system consists of three video cameras located at each port on the docking adapter and two different colored guide beacons located on the top of the CERV. The cameras will be interfaced to a computer on the Space Station. One camera is used to locate the CERV as soon as it is in view. Data from the camera is then sent to the computer which will determine the velocity, trajectory, and attitude of the CERV. This data is then compared to pre-programmed flight data and any necessary corrections are sent to the CERV's AACS computer. This will get the CERV aimed in the general direction of the docking adapter.

As the CERV draws closer to the Space Station, the other two cameras are used for fine tuning the rendezvous and docking sequence. Each light will focus on a different guide beacon. The computer will programmed to set up a grid for each beacon and compare the actual position with the necessary position as a function of distance from the Space Station. Information is then sent to the CERV's computer to correct the attitude and trajectory of the vehicle to ensure proper alignment with the docking adapter. The beacons are different colors so that the cameras can be programmed to focus on the same beacon every time. This system was chosen because the CERV can be located by the camera when it is still far from the Space Station. This allows for plenty of time to correct any errors in trajectory and attitude slowly and accurately.

Computer System

The computer system consists of two main computers and three smaller computers for the subsystems. Two main computers are used for redundancy. The three subsystem computers monitor life support systems, power and propulsion control, and AACS operations. These computers send the data to the FREES Onboard Computer (FOC) which controls power switching and other necessary subsystem functions. The FOC also controls antenna selection, downlinks to the ground and Space Station, and uplink commands from the ground and Space Station.

The design of this computer system closely resembles the AP101S General Purpose Computer (GPC) used on the Space Shuttle. This is a conceptual design that performs similar functions as the AP101S but uses advanced technology hardware available today. The two main considerations in designing this system were speed and memory expansion. The FOC will be able to handle five million instructions per second (Mips) using parallel processing, pipelining, and high speed magneto-optical memory systems. It will be microprogrammable with eight megabytes of RAM expandable to one gigabyte using single error correction - double error detection (SEC-DED) coding. The optical memory was chosen for it's weight savings over high-density modular core memory or the semiconductor memory and the fact that it is virtually unaffected by electro-magnetic radiation (Storrie-Lombardi pg.39). The large, expandable memory allows for future hardware and software improvements. A block diagram of the system is shown in Figure IV.1 (Norman pg. 317).

The central processing unit (CPU) and the input/output processor (IOP) are combined into one line replaceable unit (LRU) linked together by the high speed bus. Although they are physically one unit, they are functionally separate. The CPU will be described first.

The CPU is connected to the memory management unit (MMU) and the

IOP by the synchronous high speed bus. It is capable of high speed processing and the execution of macro instructions. The instruction unit prefetches instructions from main memory. The instructions are then put in a FIFO file. The instruction unit provides the logical address to the MMU which converts this to a physical address before the prefetch occurs. The effective address (EA) unit decodes the instruction to determine the addressing mode and the effective logical address of the operand. The MMU then translates this to a physical address and the operand is fetched. The EA unit then sends the operand and decoded instruction to the execution (EX) unit. The EX unit executes the instruction via microprogramming. The microcode provides signals to control the flow of data through hardware.

The MMU provides several other functions other than main memory management. These include controlling all timing and sequencing to the high speed bus and main memory, handling memory faults, and directing I/O commands between units in the FOC (Norman pg. 313).

The IOP is a digital, microprogrammed, time shared processor that controls commands and requests from the three subsystems. The primary functions of the major IOP components are as follows (Norman pg. 313-314):

- AGE and Discretes	Receives and transmits discrete inputs and outputs, which are single control or status lines that interface with the subsystems.
- Flow Top/Flow Bottom	Contains the general registers associated with each arithmetic and logical functions, and contains the working registers necessary to process

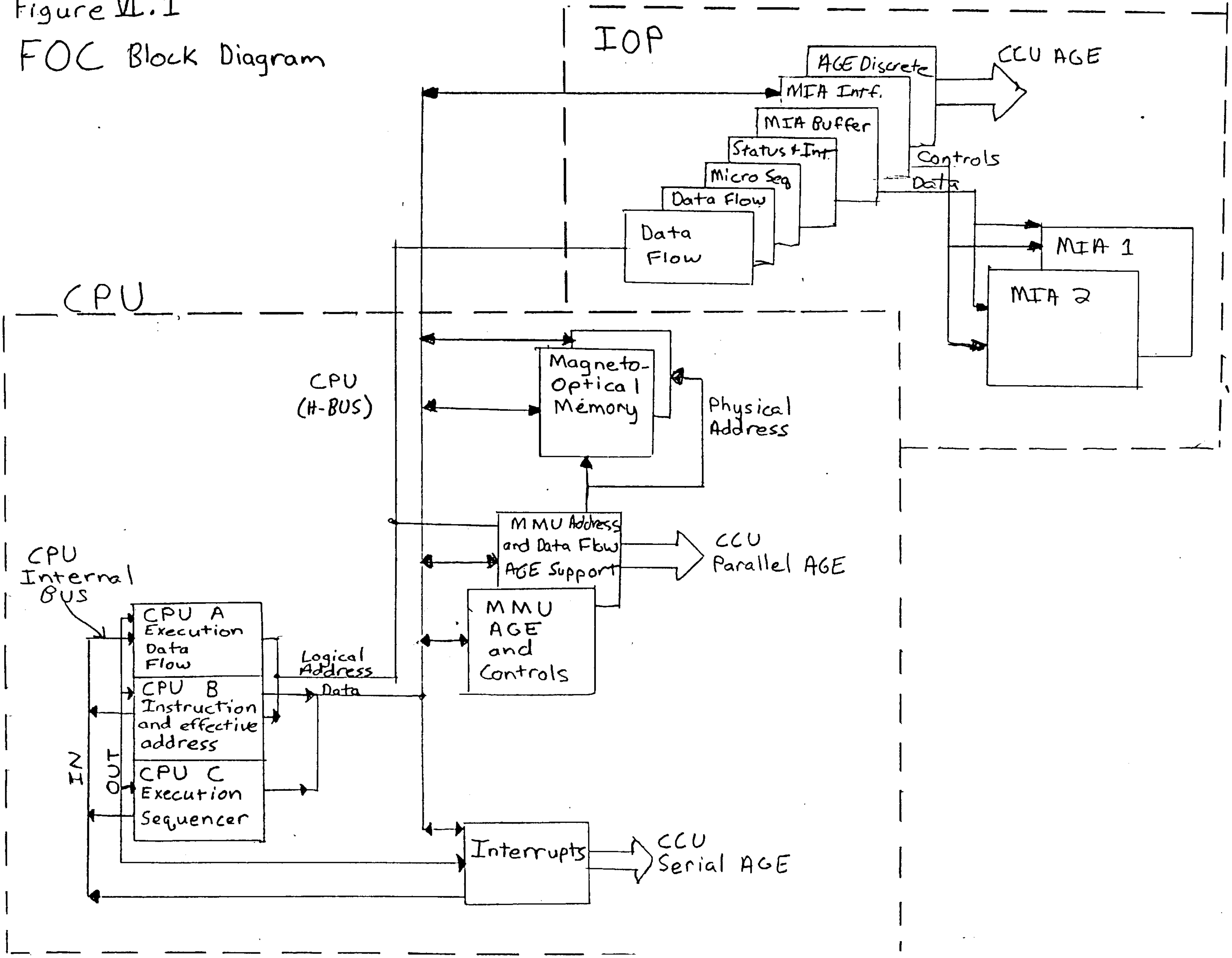
	and control data flow. Contains the Direct Memory Access (DMA) queue which is the FIFO type RAM device that handles instructions and data requests from main memory.
- Micro Sequence and Control	Contains the basic processor time sharing, read-only store for the microcode, micro sequencing, and branch logic, DMA transfer controls, stop/step controls, and local store and queue controls.
- Interface and MIA control	Interfaces the synchronous HBUS and the asynchronous IOP.
- Status and Interrupts	Performs redundancy management functions, reports status of each processor, and generates interrupts to the CPU.
- I/O Buffer	Provides the interface between the data flow section of the IOP and the three MIA pages.
- MIA	Receives and transmits data over the 10-MHz, serial, digital data buses.

The FOC will be placed on the top shelf of one of the payload racks. This requires that the unit size be approximately (0.15 X 0.5 X 0.5) meters. Each FOC will have a mass of around 15 kilograms and consume 300-325 Watts of power (the same as todays smaller high speed computers). The subsystems will be scaled down versions of the FOC with smaller memory and less speed.

Crew System Avionics

A keyboard and monitor will be attached to one of the FOC. Preprogrammed sequences for various platform missions or emergency escape will be able to be started from this terminal. Manual thruster control will also be available but every planned use for the CERV will be preprogrammed and manual control will not be necessary except under extreme circumstances. Also a small radio transceiver will be interfaced with the FOC for voice transmission. The pressurized suits will be equipped to use radio contact with the CERV for EVA missions.

Figure VI.1
FOC Block Diagram



VII. POWER

The power subsystem of FREES will be required to supply large amounts of power to several other vehicle subsystems, and it must be dependable, especially in an emergency. Since the module is a manned vehicle the power demands imposed are much greater than those of most other spacecraft.

The requirements of the power subsystem are to provide an uninterrupted flow of power to each subsystem as well as any payload if necessary. The system must protect the power sources from overloads and similarly protect the users from source malfunctions. The power subsystem in FREES will stress simplicity, reliability, safety, and low cost.

A. POWER REQUIRED

As mentioned above, the manned vehicle requires much more power than an unmanned one. The life support requirements alone require huge amounts of power. Communications are much more important and much more frequent when a crew is aboard a spacecraft. The power source must be reliable - a breakdown could be catastrophic if it were to occur while returning a decapacitated crew member to Earth.

The overall system design requirement of an almost totally automated spacecraft is another factor that makes the power demands as great as they are. The computer power demands, especially during a complicated maneuver such as docking with the Space Station, must be met flawlessly.

Table VII.1 shows the estimated power requirements of each individual vehicle subsystem. These are rather coarse estimates; hence the safety margin of 1000 Watts.

SUBSYSTEM	POWER REQUIREMENTS (W) MAXIMUM	EMERGENCY
AACS	300	300
STRUCTURES	100	50
PROPULSION	300	150
REENTRY	400	400
COMMAND & DATA	800	500
LIFE SUPPORT	1000	300
MISCELLANEOUS	100	0
SUBTOTAL	3000	1300*
SAFETY FACTOR	X 1.33	X 1.15
TOTAL	4000 W	1500 W

TABLE VII.1 - POWER DISTRIBUTION ESTIMATES

* NOTE - IN THE CASE OF AN EMERGENCY, THERE IS NOT ANY ONE MOMENT WHEN THE LANDING GEAR OF THE REENTRY SUBSYSTEM AND THE PROPULSION AND AACS SYSTEMS WOULD REQUIRE POWER SIMULTANEOUSLY.

B. POWER SUPPLY

Four power sources were investigated while designing the power subsystem for FREES: batteries, fuel cells, solar cells, and RTGs. The RTGs were eliminated very quickly because the maximum mission duration was determined from mission planning to be no more than 24 hours, clearly out of the domain of RTGs and nuclear reactors. Solar cells were also dismissed as power source candidates due to their large mass and volume. The payload fairing dimension restrictions (especially radius) referred to in the mission planning section make solar arrays for FREES an impossibility.

This left batteries and fuel cells as the two power sources for FREES. Referring to "Power Source Domains" on page 15 of AAE 241 Noteset #13, one can see that the fuel cells must be chosen as the main power source. Batteries simply cannot provide the 96 kW-hr in total energy that could possibly occur in a worst case scenario for the vehicle. Usually when the energy level exceeds 10 kW-hr, fuel cells are more efficient per unit mass than batteries (Corliss, p. 6).

The next step in the design process was to determine which type of fuel cell will best suit the power needs of the module. In complying with the overall system requirement of using proven technologies to increase reliability and reduce costs, a cryogenic hydrogen-oxygen fuel cell was chosen. The H_2/O_2 cell has been used successfully in manned reentry vehicles such as Gemini, Apollo, and more recently the Space Shuttle (Gehrke, p.5). It is worth noting at this juncture that many new fuel cell investigations are occuring in industry today and that perhaps within a decade aluminum or lithium cells may dominate the industry. However the reliable H_2/O_2 cell will be used in FREES.

In the fuel cell, the hydrogen and oxygen react to produce water, heat, and electricity. Care must be taken to prevent overheating in this area of the craft. The H_2 fuel and the oxidizer are supplied continuously to form these products. The gases are pressure-pumped into metal electrodes, where water forms and is drained away to the water storage tank of the environmental control subsystem (see Section V - ECLSS).

At the hydrogen anode electrons "are freed and travel through the external electrical load," (Corliss, p. 6). Then at the cathode (O_2 electrode) the electrons are returned to the cell and the process continues as long as the fuel and oxidizer are supplied to the system.

A schematic drawing of the fuel cell system is shown in Figure VII.1. It will be very similar to that of the Space Shuttle. The total system mass was estimated to be 800 kg with an additional 100 kg of wiring which will run throughout the vehicle supplying electicity to the various subsystems.

The secondary source of power that will be utilized should there be a breakdown of the fuel cell system are Ni-Cd batteries. These batteries were chosen due to their relatively low cost and simplicity compared to other batteries. The fact that Ni-Cd batteries have been used for 20 years in spacecraft applications made the choice to use them an easy one. If in fact there is a power shortage and the secondary source is called upon, reliability is their number one objective and these batteries have exemplified this over the years.

Referring back to Table VII.1 in this section, the emergency power required estimates are significantly lower than the maximum values. It was assumed that if this power emergency were to occur, the main power source would be repaired within 6 hours or that the module would be capable of returning to the Station or to Earth within the same time span. Consequently, the battery system was designed for a 11.25 kW-hr system. Calculations for the battery pack sizing are shown in detail in Appendix VII-A. There it is shown that 375 Ni-Cd batteries operating at 80% depth of discharge are necessary to provide the required power. The total mass of these batteries came to roughly 375 kg, since the stored energy per cell is equivalent to the stored energy per kilogram according to the data given in AAE 241 Noteset #13.

It may be possible to utilize the batteries as a primary power source during short trips to the co-orbiting platforms so that cell fuel and oxidizer can be conserved. The batteries can then be recharged when the vehicle returns to the station.

The power system components are located in the bottom section of the module, below the living quarters. The detailed layout of these components was discussed in Section II of this report.

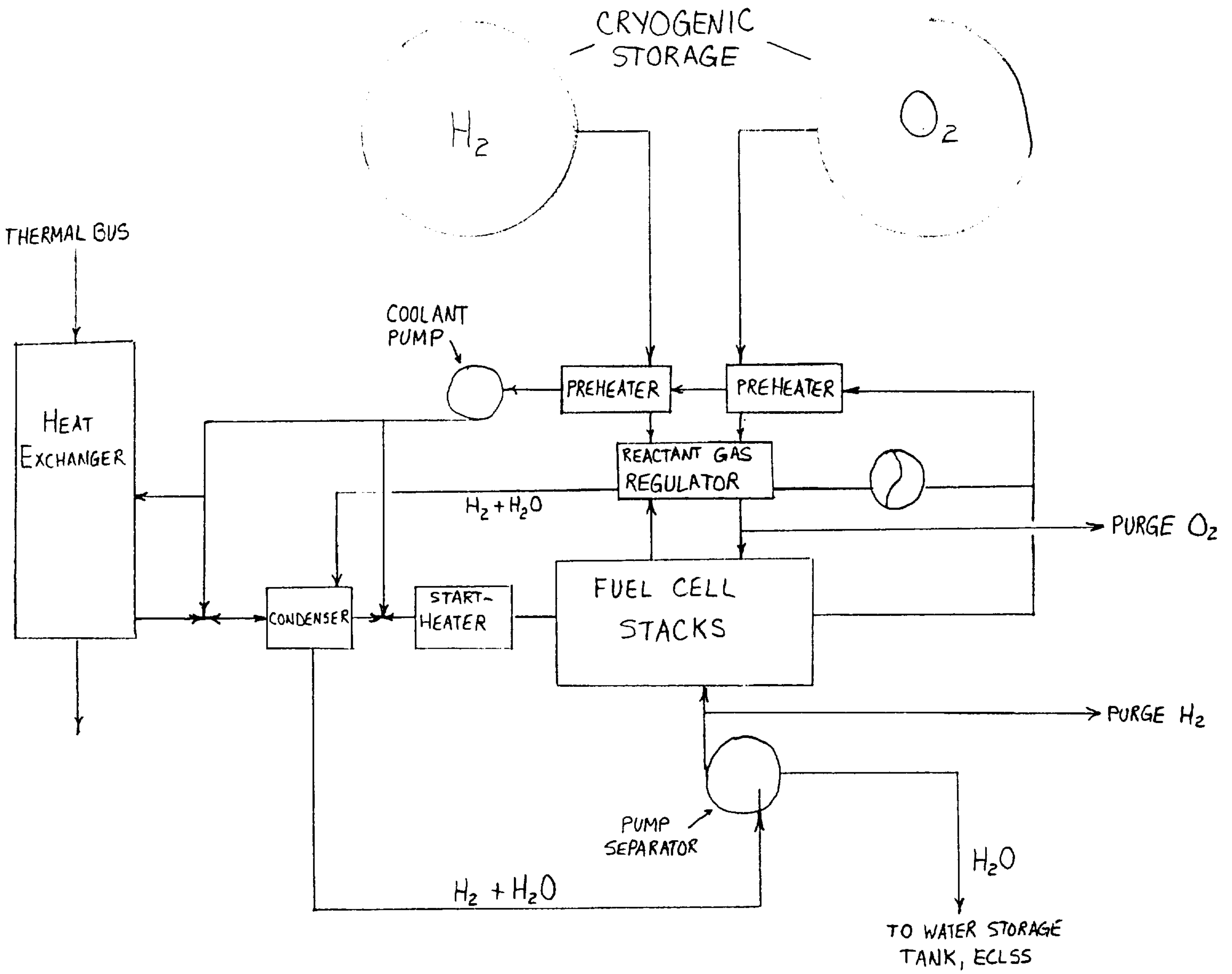


FIGURE VII.1 - CRYOGENIC H_2/O_2 FUEL CELL SCHEMATIC (GEHRKE, p. 7).

Appendix VII-A

BATTERIES: For Ni-Cd batteries, $\frac{\text{W-hr}}{\text{cell}} = \frac{\text{W-hr}}{\text{kg}} = 30$ (AAE 241 Noteset #13)

POWER REQUIRED USING BATTERIES AS A SECONDARY POWER SOURCE: 1500W for 6 Hours.

$$\#\text{ of cells} = \frac{P_L T_E / DOD}{\text{w-hr/cell}} = \frac{1500\,(6)}{0.80\,(30)} = \boxed{375 \text{ cells}}$$

$$\text{BATTERY WEIGHT} = \frac{P_L T_E / DOD}{\text{W-hr/kg}} = \frac{1500\,(6)}{0.80\,(30)} = \boxed{375 \text{ kg}}$$

VIII. Reentry and Recovery

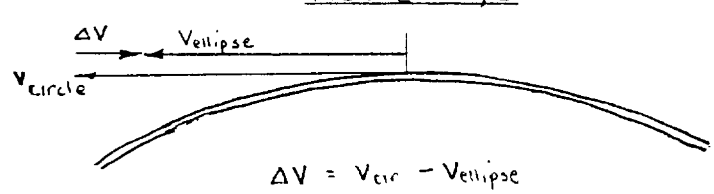
Three main requirements for reentry and recovery are to:

1) dissipate orbit energy in the atmosphere
2) protect module from heat/excessive g's
3) pick up of logistics module and crew

The module is a cylindrical body that will make a high ballistic entry into the atmosphere. Its base is 4.4 meters, and its height is 6.65 meters. Although the height of module is larger then the base, all the heavy components will be placed on the bottom to give the module a low center of gravity. See Figure VIII.1 at the end of section for the shape of the module.

Reentry starts at the Space Station. The station is located at 290 km to 430 km depending on the atmosphere. The fluctuation is to keep the distance between atmosphere and the Space Station the same. The delta v , the change in velocity, needed for reentry in to the atmosphere needs to be considered. For delta v, consider the most extreme case of 430 km altitude of the space station. The delta v can be solved by considering a simple diagram.

FIGURE VIII.2



The module will be traveling at the same speed as the station and the module will need to get into an elliptical path for reentry. The delta v is needed to slow down the module to set it into the elliptical path. The delta v was solved by using the equations from the class notes (AAE 241). Propulsion will have to give a delta v of .1191 km/sec.

eqn. 1

$$\Delta V = V_{cir} - V_{ellipse}$$

$$V_c = \sqrt{\frac{\mu_\oplus}{r}} = \sqrt{\frac{3.986\times10^5}{6808}} = 7.6517 \text{ km/sec}$$

$$V_e = \sqrt{\mu_\oplus\left(\frac{2}{r} - \frac{1}{a}\right)}$$

$$= \sqrt{3.986\times10^5\left(\frac{2}{6808} - \frac{1}{6604}\right)} \qquad a = \frac{r_a + r_p}{2} = \frac{6808 + 6400}{2} = 6604 \text{ km}$$

$$= 7.5326 \text{ km/sec}$$

$$\Delta V = 7.6517 - 7.5326 = .1191 \text{ km/sec}$$

See figure VIII.3 at the end of this section for a diagram of reentry trajectory from the Space Station to the upper boundary of the atmosphere.

The time that it takes for the module to reach the upper atmosphere from the Space Station can be calculated with this equation given in the class notes for AAE 306(Orbital Mechanics by Conway). eqn. 2

$$T = 2\pi\sqrt{\frac{a^3}{\mu_\oplus}}$$

$$T/2 = \pi\sqrt{\frac{(6604)^3}{3.986\times10^5}}$$

$$= 44.51 \text{ minutes}$$

Because the nozzle for thrust is located at the top of the vehicle, the module will have to be turned 180 degrees, so that the module's heat shield will be in proper position to encounter the atmosphere. The attitude and control will be able to orient the module in 35 seconds with the use of control moment gyros. This gives the module plenty of time for adjustments.

The trajectory for the module was calculated so that the module will only experience up to 3 g's upon reentry. This allows only a small corridor for safe reentry.

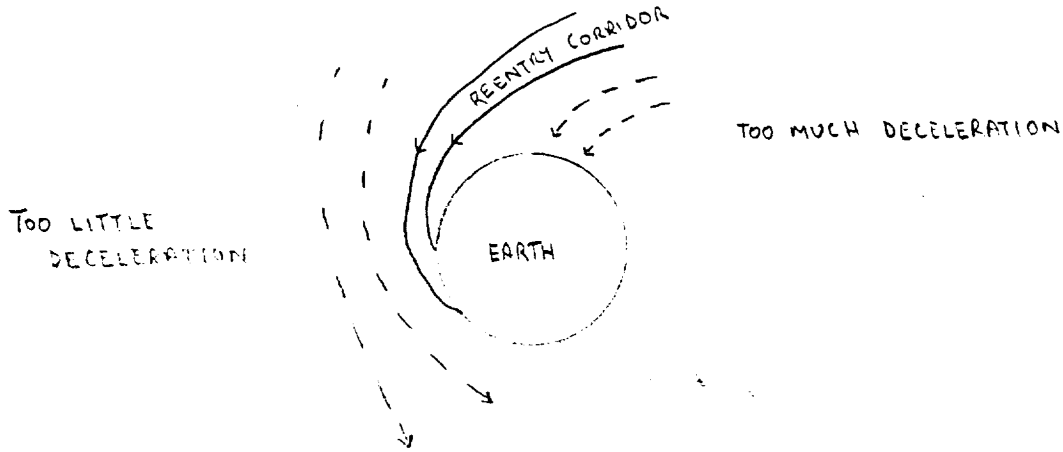
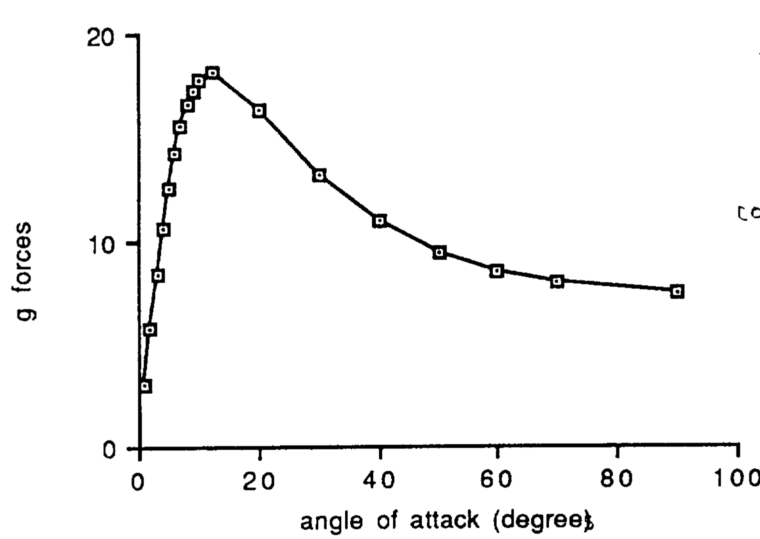


FIGURE VIII, 4

By using a simple Fortran program with the equations below, the entry angle was calculated to be 1.036 degrees, which will give 3 g's to the module upon reentry. See Figures VIII.5 and VIII.6 for g forces vs. entry angle.

G Forces vs. Angle of Attack(figure 8.5)



eqns. used:

$$a = \frac{r_a + r_p}{2}$$

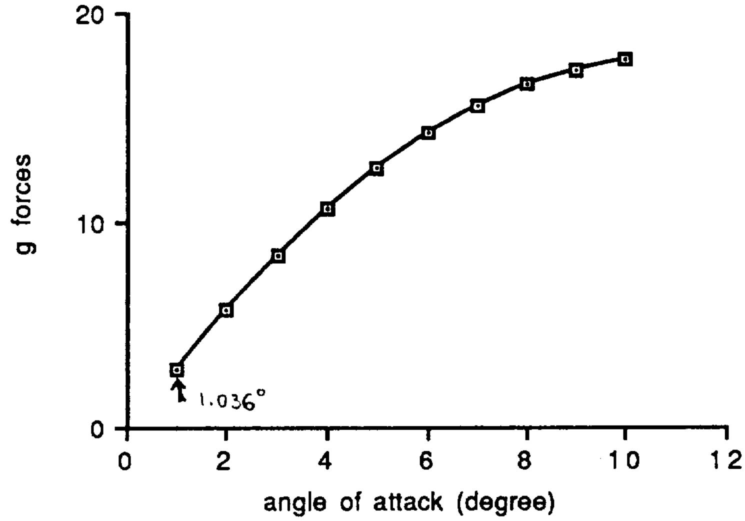
$$V_e = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

$$\cos \gamma_E = \sqrt{\frac{a^2(1-e^2)}{r(2a-r)}}$$

$$a_{max} = \frac{V_E^2 \sin \gamma_E}{2 e a H_s}$$

ORIGINAL PAGE IS OF POOR QUALITY

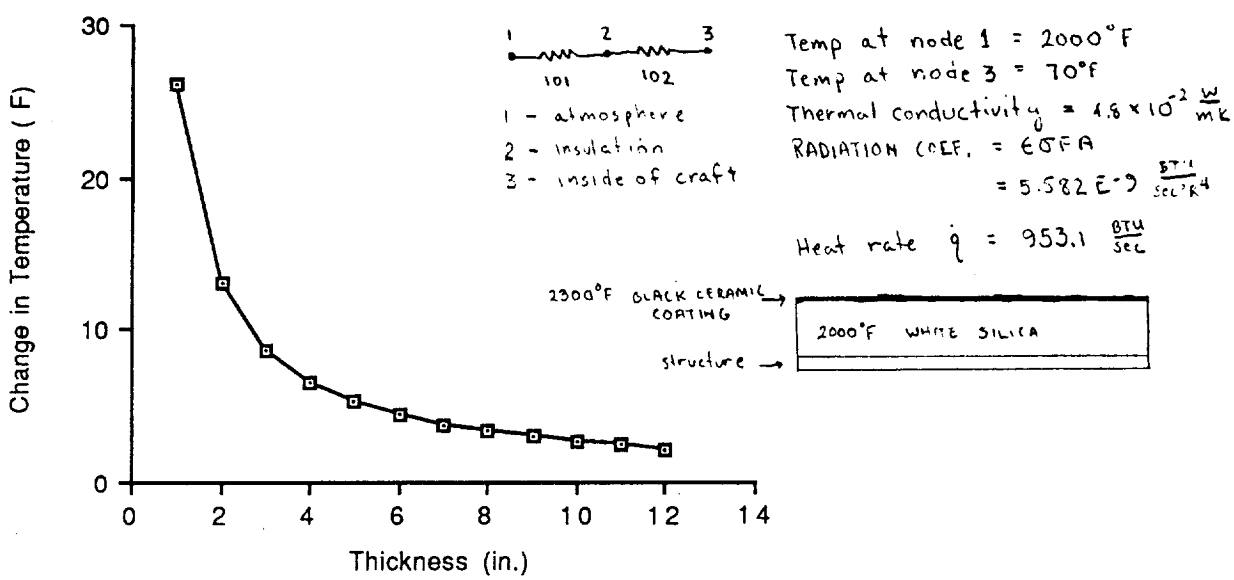
G Gorces vs. Angle of Attack(fig 8.6)



Thermal protection is an important part of reentry. The material for thermal insulation for the module will be REI(reusable external insulation) - silica. This material was chosen because of its low density(8.5 lb/ft^3). The material is based on rigidized ceramic fiber monofilaments. This amorphous(vitreous) silica has low solid thermal conductivity(< 4.8E-2 W/m^{o}K), and low coefficient of thermal expansion. The low coefficient of thermal expansion results in excellent thermal shock resistance and thermal induced strain. It is also very cheap and easy to produce. (Space Shuttle Materials, Pg. 445). This material, with a combination of a ceramic coating such as the carbon carbon composite, has been used extensively on the shuttle, and it will be sufficient for the module's needs. The REI-silica can be used for up to 100 plus missions. This fulfills the requirement of multiple missions. Some other materials were considered, such as, REI-mullite, REI-zirconia, TdNiCr thermal protection system. These other materials were not as light and were more expensive than REI-silica.

The module will experience temperatures of 2000 ^oF at the leading base of the module. This is where most of the heat dissipation will occur. The sides will also need thermal protection due to the nature of the design, but not as much as the bottom of the module. By using ITAS, a program used for a class assignment, a graph comparing the thickness of the insulation vs. the change in temperature of the inside of the module spanning four minutes was calculated. See Figure VIII.7 for graph of temperature change vs. thickness. The thickness of six inches was selected for the module because the change in temperature was only 2^{o}F, and the temperature change tapered off from there. By increasing the thickness from 6 inches, the inside of the module will only notice a slight change in temperature. REI-silica will be much like the shuttle's insulation, and there will not be any new technology involved.

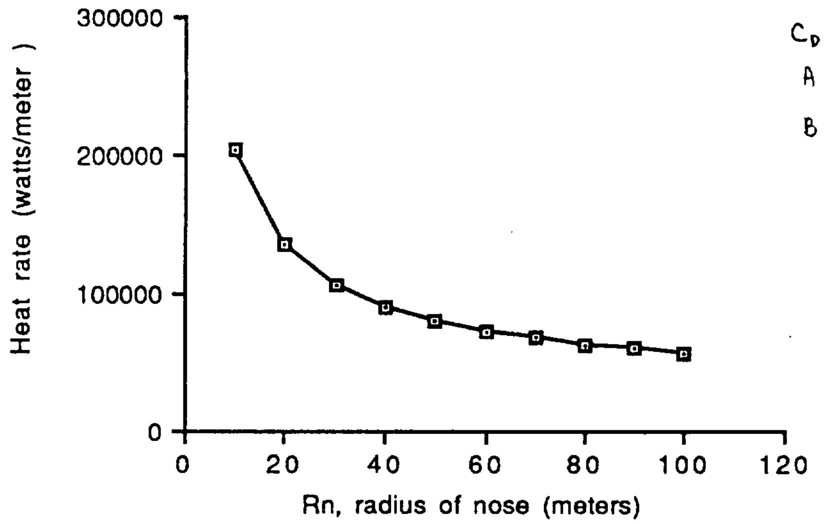
Delta T vs. Thickness (figure 8.7)



ORIGINAL PAGE IS
OF POOR QUALITY

The shape of the bottom layer was calculated to be a very flat curve. The Rn, the nose radius, was chosen to be 100 meters. See Figures VIII.8 and VIII.9 for graphs of heat rate vs. Rn. As the Rn becomes greater, the heat rate decreases. The decreasing heat rate tapers off at Rn = 100m. The total heat load also tapers off at Rn = 100m.

Heat rate vs Rn (figure 8.8)



EQNS. USED:

$m = 18000$ Kg

$C_D = 1 - \cos^4\theta$

$A = \pi R_n^2$

$B = \frac{m}{C_D A}$

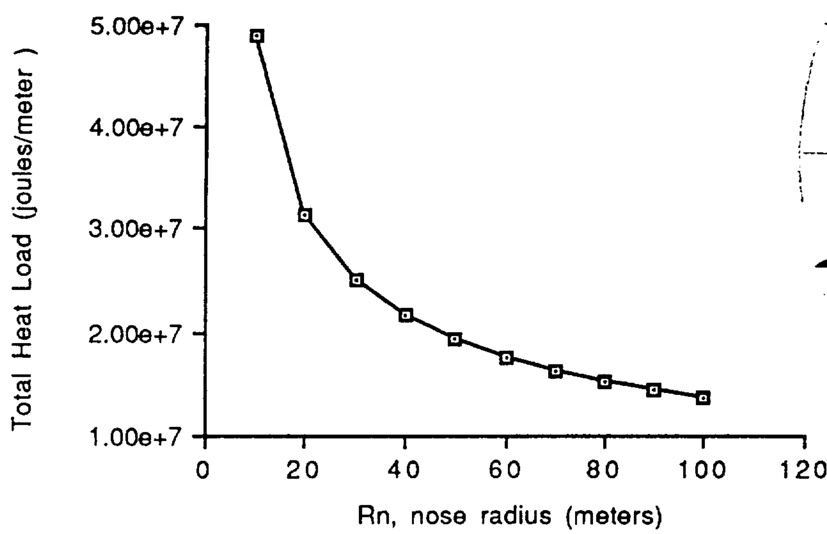
$$\dot{q} = \frac{3.5 \times 10^7}{\sqrt{K R_n}} \left(\frac{V_E}{V_0}\right)^3 \ \frac{W}{m^2}$$

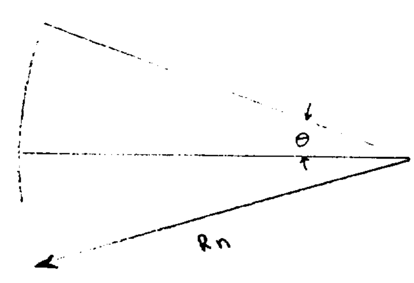
$$Q_0 = \frac{10^8}{V_0} \left(\frac{V_E}{V_0}\right)^2 \left\{ \frac{m}{C_D A \rho_0} \left(\frac{H\pi}{R_n \sin\gamma_E} \right) \right\}^{\frac{1}{2}} \ \frac{J}{m^2}$$

$$K = \frac{C_D A \rho_0 H}{m \sin\gamma_E}$$

(from [illegible] class notes)

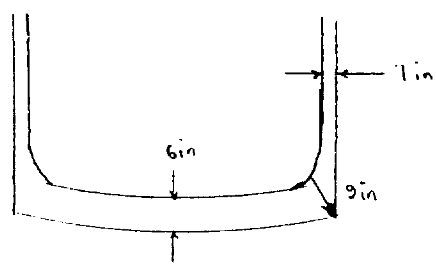
Total Heat Load vs. Rn (figure 8.9)





The corners of the module will receive a great amount of the heat load, therefore the thickness at the corners will be increased by rounding off of the edges of the inside structure. Although it will not be crucial, the life support can protect the crew with thermal suits as a precautionary measure.

FIGURE VIII, 10



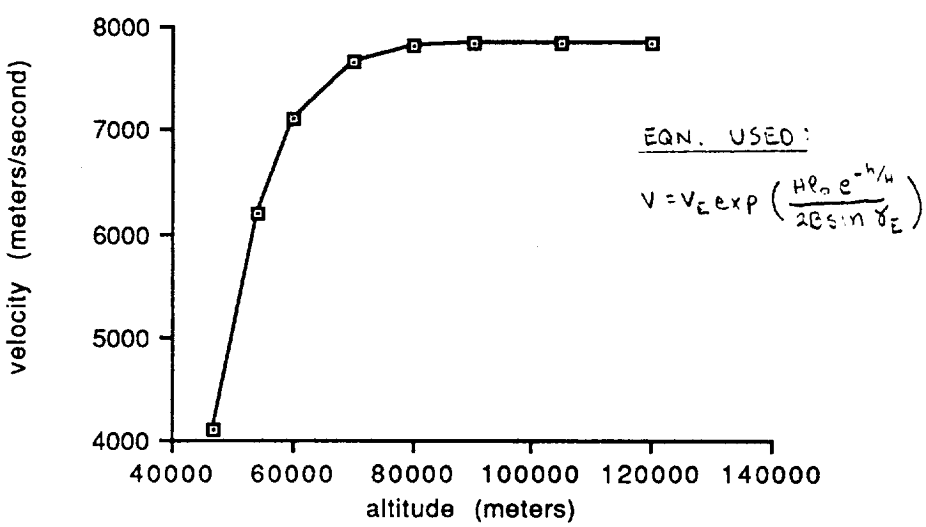
A comparison study was done for different types of chutes

	control	deployment	stability	vertical
parachute	none	good	good	30
parasail	ok	good	good	25
paraglider	good	not good	unknown	0
parafoil	good	good	good	0-5
rotors	unknown	unknown	unknown	0

For the landing system, the parafoil seems to be the obvious choice because of its controllability, and its soft landing capability. The Pioneer Systems Inc. in Melbourne, Florida is developing a system called ARS (Advanced Recovery System). This system, when developed, will be sufficient to support the module of 18000 kg. The proposal from Pioneer Systems Inc. stated that the parafoil will be ready for production in 1994. See Figure VIII.11 for picture of the parafoil. The dimensions of the parafoil were scaled down from the model drawing for module mass of 60000 lb on pg. 105 of the ARS study by Pioneer Systems Inc. The weight distribution for the components of the parafoil can be seen on Figure VIII.11. The total weight of the parafoil package will be 400.5 kg. The total area needed for the parachute and accessories will be 5.0 ft^3.

The deployment of the parafoil is determined by the velocity of the module. See Figure VIII.12 for a velocity vs. height diagram. The parafoil deployment will be initiated at 18.3 km altitude. The altitude vs. time relationship of the module can be seen on Figure VIII.13 at the end of this section.

Velocity vs. Height (figure 8.12)



The equations did not include a gravity term and where the Figure VIII.12 shows the greatest deceleration is where the craft will reach terminal velocity. The module reaches terminal velocity at 47 km.

Control of the parafoil will be the job of ground command. The crew will not be able to control the parafoil. By controlling the parafoil, the module can land on most solid flat surfaces with sufficient area. A clear field of radius of 1 mile will be more than sufficient for landing purposes. The parafoil will bring the craft to the ground with very little or no vertical velocity. Also, by adjusting the parafoil just before landing, the module can attain zero horizontal velocity. In the case where the module might have horizontal velocity due to severe wind conditions, the landing pads will be made of skid type material, and the landing gear will be placed such that the module can make a landing much like an airplane. This will give the module a very soft landing with negligible g forces.

The weather conditions upon landing is very important. The parafoil can only be used in dry weather because excessive moisture could severely retard its functions.

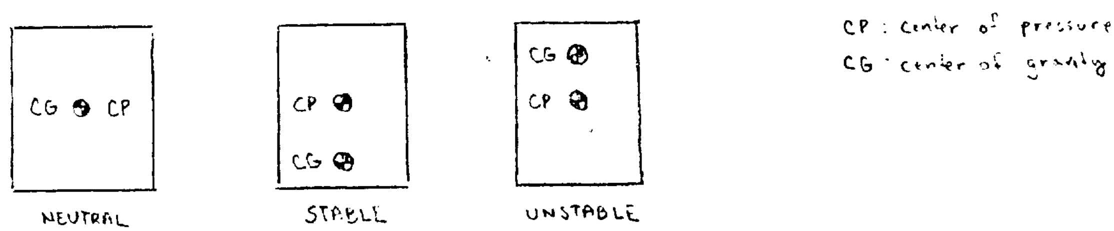
The landing gear will be made of an aluminum-lithium alloy. This material has low density for a metal and good strength. It is relatively cheap compared to other high-tech composition metals. Some alternatives are CFRP (carbon fiber reinforced polymer), and GFRP (graphite fiber reinforced polymer). These polymers are very strong and can take on great loads, but these materials cost much more then the aluminum-lithium alloy.

The landing gear system and its size can be seen in Figure VIII.14. The landing gears will be deployed by hydraulics. There will be a spring and damper system to absorb most of the shock of landing. There will be three 'legs' on the module protruding at an angle of 45°. The pads will be located .8 meters out from the edge of the module, and this will give it stability. The pads will be a circular disk made of material with a moderate friction coefficient. This will allow the module to have a landing with some horizontal speed without tipping over. The low center of gravity of the module will also help the stability of the craft upon landing.

The advantage of landing on land is the on-scene assistance, and with the accurate reentry guidance and control capability supported by a zone landing, the crew will receive assistance right from the point of touch down. If there are any injured crew members, they will be treated right away.

Some problems may arise during development. The landing gear hatches will be a weak point in the heat shielding. This may cause some problems. The deployment of landing gear from the side of the module can be considered. The high ballistic entry velocity of the module may need to be retarded by a very sturdy chute before the main parafoil is deployed because of the high velocities involved with a cylindrical type of body. Due to the module's cylindrical shape with its base smaller than the height, the module's center of gravity will have to be very low so that the center of pressure will be significantly above it. This will keep the module orientated in the right position with heat shield facing entry.

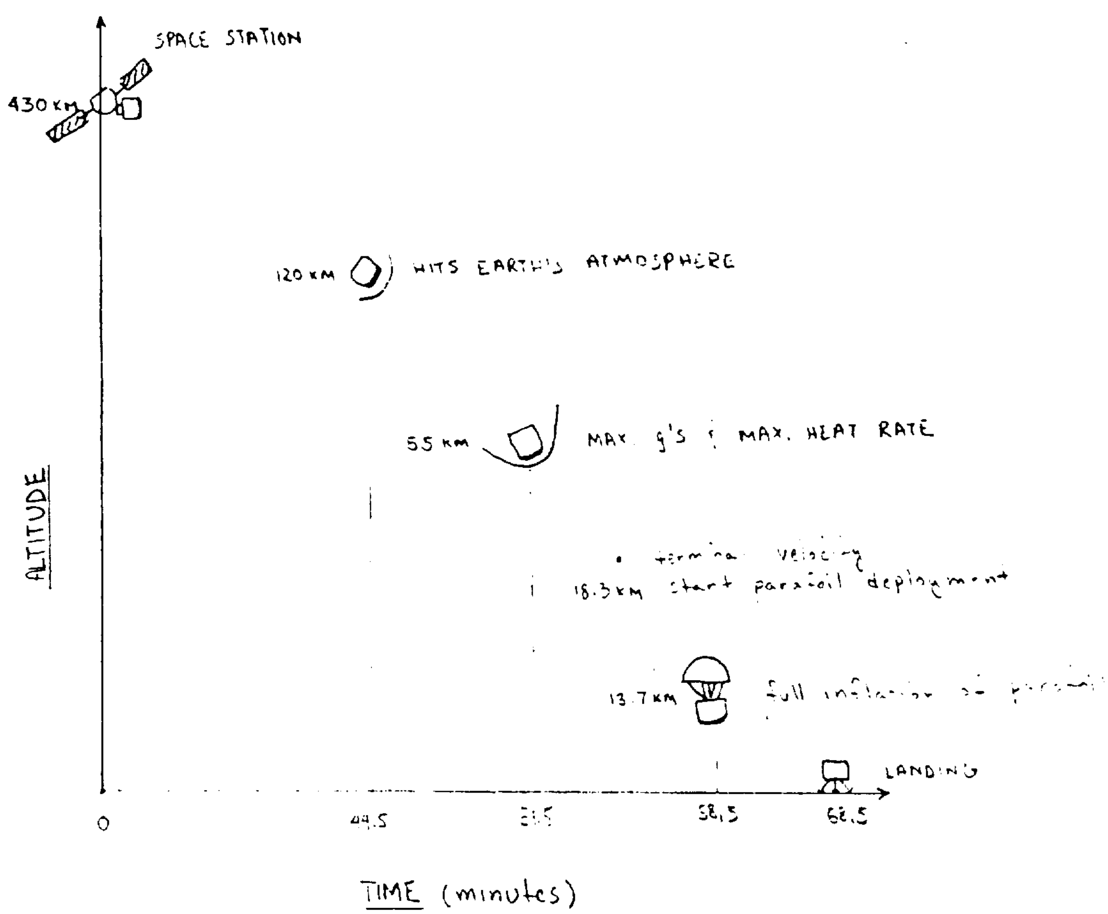
FIGURE VIII, 15 STABILITY



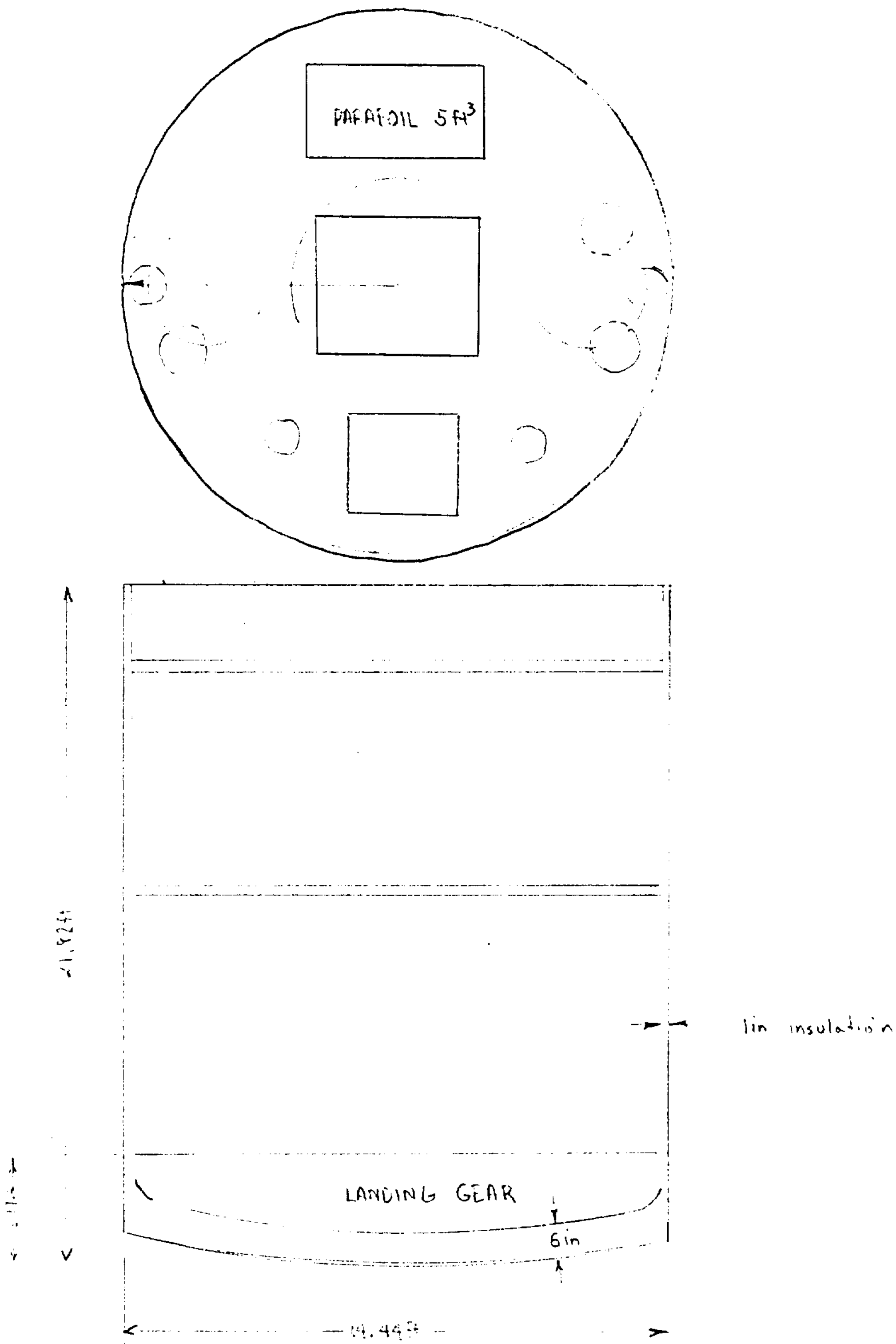
Reentry involves integration with other subsystems. The attitude and control system must position the module for reentry, and the propulsion system will have to supply the delta v required for reentry.

The possible landing site selection will involve mission planning. The life support subsystem will supply the thermal suits and beds to absorb large g forces. Power will be necessary for deployment of parafoil and landing gear. Communication is needed for parafoil control from ground control, and the structural subsystem will have to integrate the heat shielding and landing gear in to the module. Reentry will have to work closely with all the other systems because reentry will be involved with all the other subsystems.

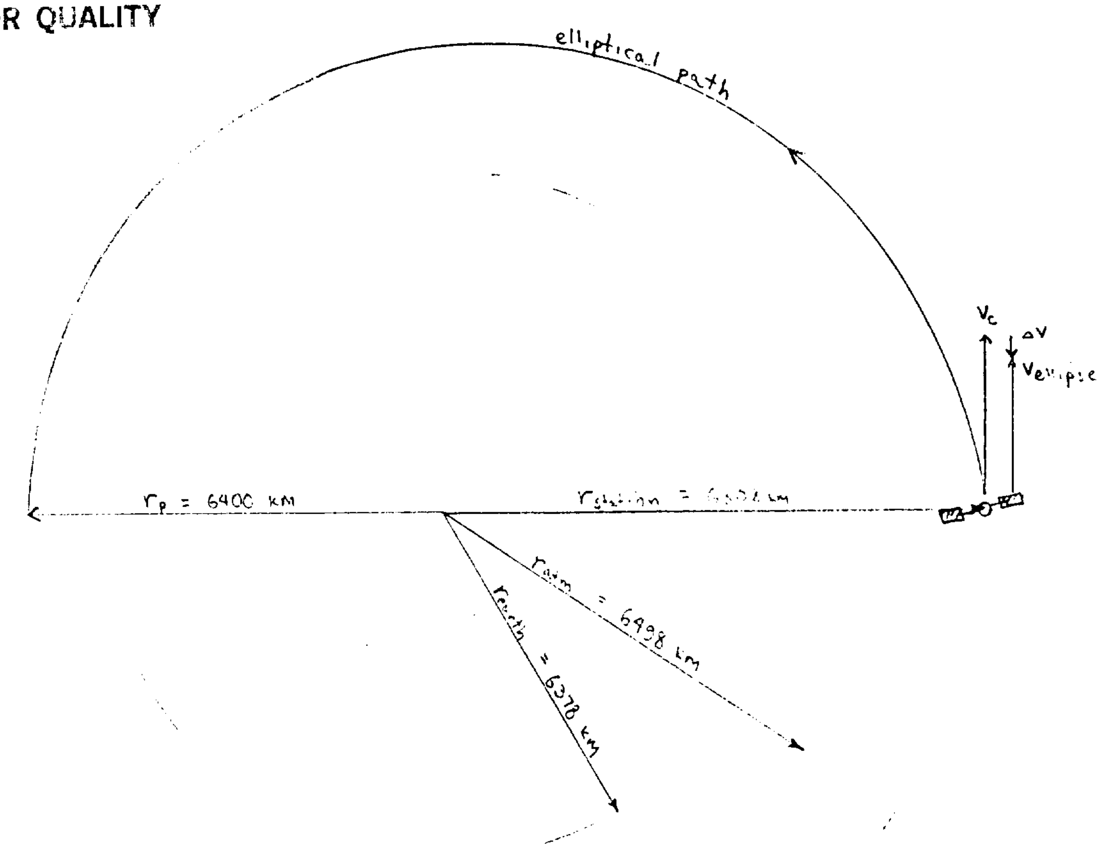
ALTITUDE vs. TIME (FIGURE 8.13)



PARTS OF REENTRY (FIGURE 8.1)



REENTRY TRAJECTORY (FIGURE 8.3)



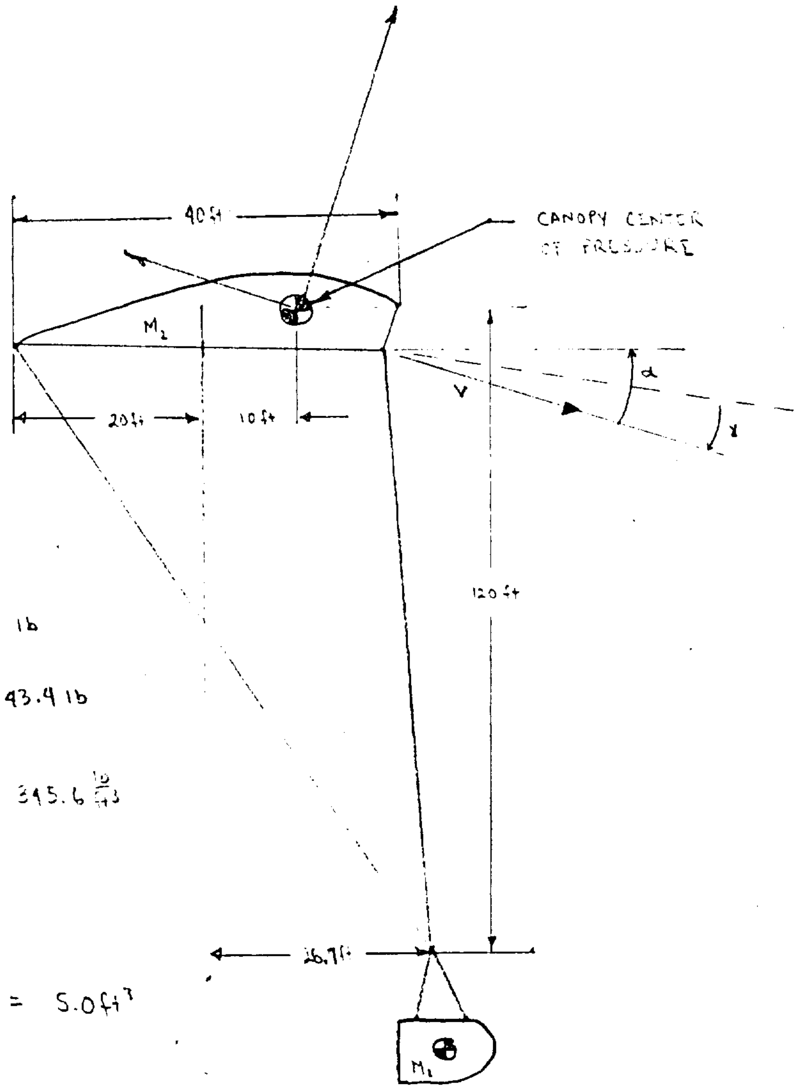
$$V_{c1} = \sqrt{\frac{\mu_\oplus}{r_{station}}} = 7.6517 \ \frac{km}{sec}$$

$$V_{ellipse} = \sqrt{\mu_\oplus \left(\frac{1}{r_{station}} - \frac{1}{a} \right)} \quad ; \quad a = \frac{r_{station} + r_p}{2}$$

$$= 7.5326 \ \frac{km}{sec}$$

$$\Delta V = V_{c1} - V_{ellipse} = .1191 \ \frac{km}{sec}$$

PARAFOIL SIZING (FIGURE 8.11)



m_1 = 40,000 lb

m_2 = 643.5 lb

m_3 = mass of cables and parachute packages = 94 lb

m_4 = mass of pilot parachutes = 143.4 lb

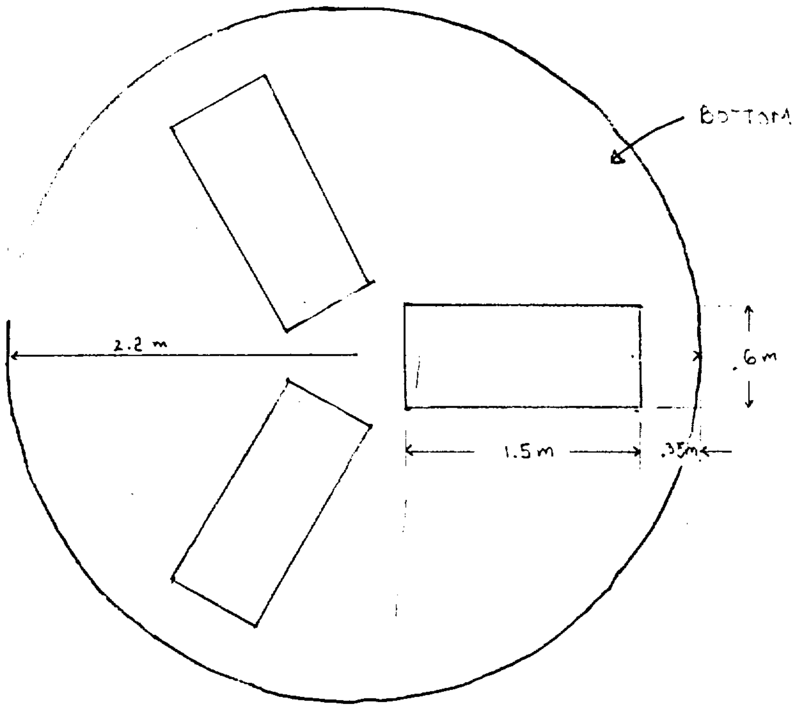
Packing Density $= .2 \frac{lb}{in^3}\left(\frac{12in}{1ft}\right)^3 = 345.6 \frac{lb}{ft^3}$

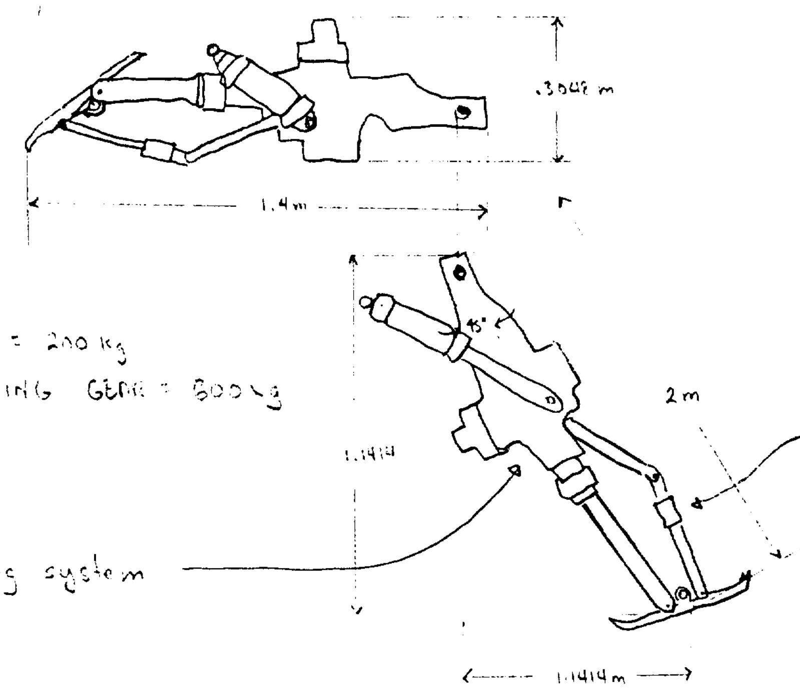
$786.9 \text{ lb} / 345.6 \frac{lb}{ft^3} = 2.2769 \text{ ft}^3$

The total area needed for parachutes and cables $= 5.0 \text{ft}^3$

ARS PHASE II, PIONEER SYSTEMS INC.
Pg. 105

CONCEPTUAL DESIGN OF LANDING GEAR (figure VIII. 14)

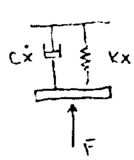




mass of one strut = 200 kg

TOTAL MASS OF LANDING GEAR = 800 kg

shock absorbing system



(not exactly to scale)

IX. MISSION COSTING

Costing methods were based on those incorporated by Rockwell International for costing of the Space Station (AAE 241, Noteset 238.19). The same coefficients and scaling exponents were used, but the other variables were estimated. Although much error is possible by costing in this manner, it does allow for a reasonable estimate of the project's costs.

Because of the costing method used, many rough estimates had to be made, including those for design complexity, DC, production complexity, PC, and percent new design, %ND. Slight variations in any of these numbers tended to cause significant differences in the system's final cost. Although most estimates should be relatively close, some may be incorrect. For example, the life support system aboard FREES is an open-loop system. The Space Station life support system is closed-loop, which is much more complex. Therefore, a very low design complexity was chosen to offset the difference.

A complete breakdown of the subsystems' costs is provided in Table IX.1, with sample calculations provided in Appendix IX-A. It was determined that the cost of design, development, testing, and engineering (DDTE) of the FREES vehicle will be $355.7 million, while the cost of production is only $98.6 million per vehicle [see Table IX.1]. Most importantly, it was estimated that nearly $270 million was saved in design costs by incorporating components previously developed for use on the Space Station, Space Shuttle, and satellites. From Appendix IX-B, the total cost to design and produce eight vehicles was estimated at $1.14 billion. Although this appears to be a very large sum of money, the average cost is only $143.1 million per vehicle. If project funding was to become severely limited for any reason, the fleet could be cut back to five or six vehicles, although a six-vehicle fleet would still cost $947 million.

Finally, cost estimates were made for the entire project, assuming a lifetime of six years, which was the minimum design life of the system [see Appendix IX-C]. The cost of the entire project, excluding the cost of launch services, supplies, and vehicle maintenance, is approximately $6.6 billion. This reduces to an average annual cost, excluding inflationary effects, of $1.1 billion, and an average cost per launch of $137.5 million. Note, however, that the driving factor in this cost is the launch vehicle, which is $110 million alone (Neilon, p.53). Therefore, the cost of this entire system could be drastically reduced with the development of a more powerful or cheaper expendable launch vehicle.

TABLE IX.1 - VEHICLE COSTING

SUBSYSTEM	WEIGHT KG	WEIGHT LBS	DDTE[1,2] A	DDTE[1,2] B	DC[3]	%ND[4]	COST[5]	SAVINGS[6]	PRODUCTION[1] A	PRODUCTION[1] B	PC[7]	COST[8]
STRUCTURES	5048	11,106	1.76	.49	.8	75	121.7	40.6	.42	.44	1.0	30.4
PROPULSION	1647	3624	.10	.88	.75	25	30.5	91.5	.11	.55	.6	7.2
ELEC POWER	1275	2805	.57	.58	.8	50	27.3	27.3	.04	.78	.8	18.8
COMMUNIC	64	140	7.81	.58	1.0	90	148.2	16.5	.05	.92	.8	4.5
G,NAV,CTRL	57	126	4.57	.52	.8	10	5.4	48.8	.86	.49	.8	8.8
RCS	119	262	.10	.88	.6	25	2.4	7.3	.11	.55	.6	1.7
LIFE SUPPORT	892	1962	11.72	.41	.15	25	11.8	35.4	.79	.50	.6	25.2
TH CTRL-ACTIVE	40	88	1.50	.26	1.0	80	3.8	1.0	.18	.55	.5	1.1
TH CTRL-PASSIVE	466	1025	.35	.42	.8	75	4.6	1.5	.05	.39	1.0	.9
TOTALS	9608	21,138					355.7	269.9				98.6
DOCKING ADPT	2400	5280	.45	.49	.4	100	14.4	0	.06	.44	.75	2.3

1. Coefficients and exponents from AAE 241 Noteset 238.19, in 1984 dollars
2. Design, Development, Testing, and Engineering
3. Design Complexity
4. Percent New Design
5. Cost of DDTE in $ millions, 1989 dollars
6. Estimated savings through use of previous designs
7. Production Complexity
8. Cost of Production in $ millions, 1989 dollars

NOTE: The Escalation Index is 1.2 to convert from 1984 to1989 dollars (Koepke, AAE 241)

APPENDIX IX-A

COSTING EQUATIONS

Design, Development, Testing, and Engineering

$$Cost_{DDTE} = A\ [\text{Weight-lbs}]^{B}\ (DC)\ (\%ND)\ (EI)$$

Production

$$Cost_{Prod} = A\ [\text{Weight-lbs}]^{B}\ (PC)\ (EI)$$

Where A=Costing Coefficient

B=Scaling Exponent

DC=Design Complexity

%ND=Percent New Design

PC=Production Complexity

EI=Escalation Index

Example - Structures Subsystem:

$$Cost_{DDTE} = 1.76\ [11{,}106\ \text{lbs}]^{.49}\ (.8)\ (.75)\ (1.2)$$

$$= \$121.7\ \text{Million (1989 Dollars)}$$

$$Cost_{Prod} = .42\ [11{,}106]^{.44}\ (1.0)\ (1.2)$$

$$= \$30.4\ \text{Million (1989 Dollars)}$$

APPENDIX IX-B
TOTAL COST OF RESUPPLY VEHICLES
[$ Millions, 1989 Dollars]

Cost of DDTE*		$ 355.7
Cost of Production		
Cost per Vehicle	$ 98.6	
Vehicles per System	x 2	
Cost per System	197.2	
Systems Required	x 4	
Total Cost of Production (8 vehicles)		788.8
Total Cost of Resupply Vehicle		**1144.5**
Average Cost Per Vehicle		**143.1**

*Design, Development, Testing, and Engineering

ANNUAL OPERATIONAL COSTS
[$ Millions, 1989 Dollars]

Expendable Launch Vehicles		
Cost per Titan IV*	$ 110	
ELV's Required	x 8	
Total Annual Cost of Operation		**$ 880**

* This does not include launch services, supplies, or maintenance.

APPENDIX IX-C

TOTAL PROJECT COST

[Six Years of Operation]

[$ Millions, 1989 Dollars]

Resupply Vehicles (8)*		$ 1,144.5
Docking Adapter on Space Station		16.7
Delta II Launch Vehicle for Adapter Transport		50.0
Expendable Launch Vehicles		
Annual	$ 880	
Years of Operation	x 6	
	5,280	
Initial Spare ELV	+110	
Total (49 Titan IV's)		5,390.0
Total Project Cost (Six Years of Operation)		**6601.2********
Average Annual Cost		**1100.2********
Average Total Resupply Cost (Two Launches)		**275.1********
Average Cost Per Launch		**137.5********

* Includes Design, Development, Test, and Engineering

** Does not include cost of launch services, supplies, or maintenance

References

1. Storrie-Lombardi, Michael and Lisa, October 12, 1988: The Next Day, MacTech Quarterly, Spring 1989.

2. Norman, P.G., The New AP101S General Purpose Computer (GPC) for the Space Shuttle, Institute of Electrical and Electronics Engineers (IEEE) Proceedings, March 1987.

3. K. Tu, J.H. Johnson, W.E. Teasdale, D.N. Cravey, Y.S. Kuo, P.L. Harton, and Y.C. Loh, Space Shuttle Communications and Tracking System, IEEE Proceedings, March 1987.

4. American Astronautical Society, Guidance and Control 1979, Univelt,1979.

5. American Astronautical Society, Guidance and Control 1980, Univelt, 1980.

6. Sutton, George P., Rocket Propulsion Elements, Wiley, 1986.

7. American Astronautical Society, Guidance and Control 1988, Univelt, 1988.

8. Chobotov, V.A., Spacecraft Attitude Dynamics and Control, Kinko's copies, 1989.

9. Van Woerkom, P. Th L.M., Automatic Control in Space 1982, Pergamon Press, 1982.

10. Greensite, Arthur L., Analysis and design of Space Vehicle Flight Control Systems, Spartan Books, 1970.

11. Quasius and McCanless, Star Trackers and System Design, Spartan Books, 1966.

12. NASA, The Space Transportation System User Handbook, NASA, 1970.

13. Satellite Technology and Its Applications, Wiley, 1988.

14. Ashby, Michael F. and Jones, David R.H., Engineering Materials-An Introduction to their Properties and Applications, Permagon Press, 1982.

15. Rault, T., ARIANE Transfer Vehicle (ATV) to Supply Space Station. AIAA Paper #87-1862.

16. Corliss, William R., Spacecraft Power, NASA, 1970.

17. Gehrke, H., Heqn, J., and Dietrich, G., Fuel Cells Systems for Space Applications in Europe, Acta Astronautica, May 1988.

18. Conway, B. A. Orbital Mechanics, Kinko's, Urbana, 1988.

19. Lembeck, Mike Class Notes, Kinko's, Urbana, 1989.

20. Loh, W. H. T. Re-entry and Planetary Entry, Springer Verlag, New York, 1968.

21. National Aeronautics and Space Administration Project Gemini, Washington D. C., 1969.

22. Purser, Paul E. Manned Spacecraft, Fairchild publications Inc., New York, 1964.

23. Roskam, Jan Airplane Design, Roskam Aviation, Lawrence, 1986.

24. SAMPE Space Shuttle Materials, Huntsville, 1971.

25. Koepke, Andy AAE 241 Notes on Threats, Kinko's, Urbana, 1989.

26. Bain, Dan AAE 241 Notes on Thermal Control. Kinko's, Urbana, 1989.

27. Lembeck, Michael AAE 241 INERT, Kinko's, Urbana, 1989.

28. Bolger, Philip Space Rescue and Safety 1975, Univelt, San Diego, 1975.

29. Neilon, John J. Use of Expendable Launch Vehicles, Virginia, 1988.

30. Humphries, W. R. Growth Evolution of the Space Station ECLSS, Huntsville, 1986.

31. Zweben, C. SAMPE Quarterly, volume 7, 1976.