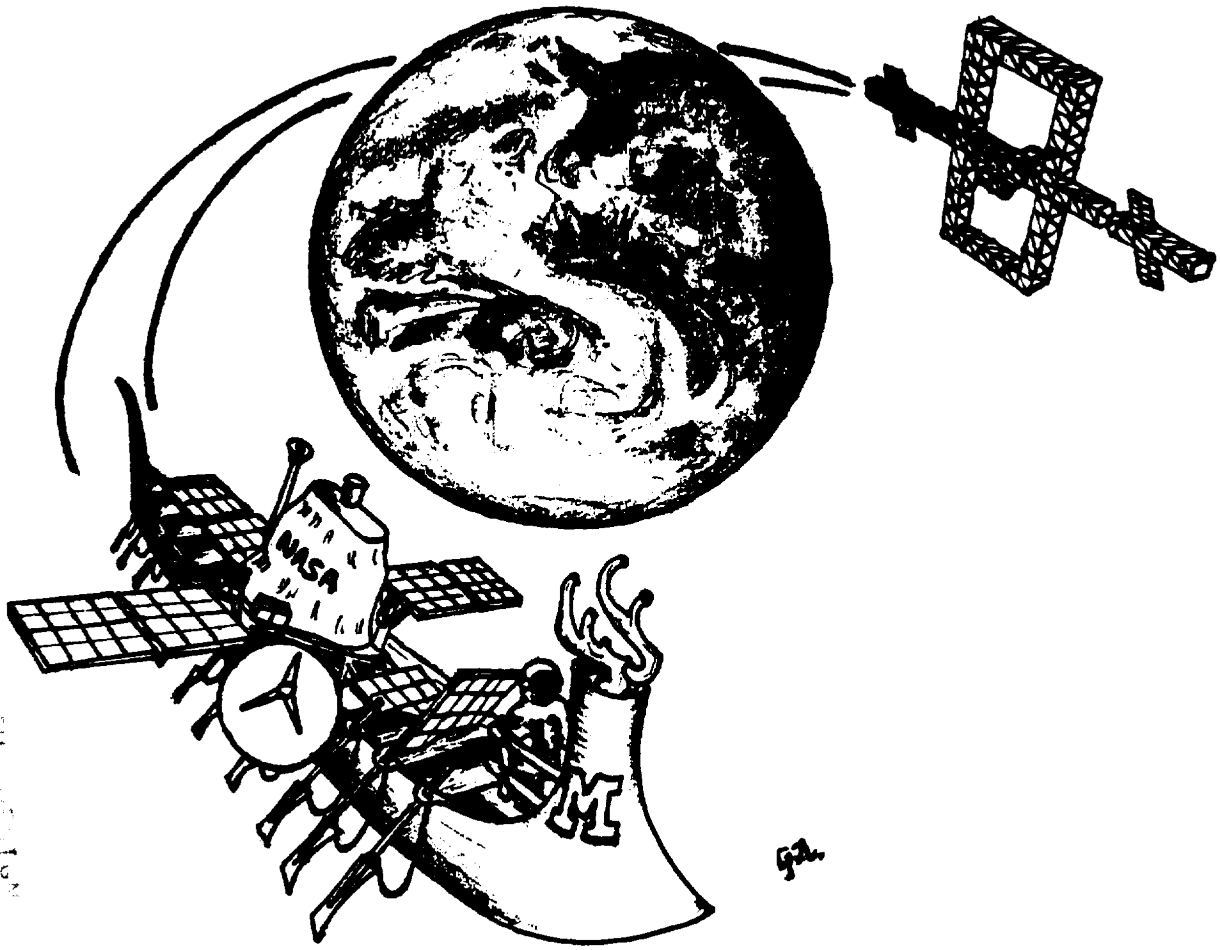
Project Argo



AERO 483
Aerospace System Design
The University of Michigan
NASA /USRA
April 1989
NASW-4435

Project Argo
The Design and Analysis of an All-Propulsive and an Aeroassisted Version of a Manned Space Transportation Vehicle

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Cover design by Grenmarie Agresar

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Foreword

Aerospace Engineering 483, "Aerospace System Design", is one of a number of design courses open to students in Aerospace Engineering at the University of Michigan. Each year a new topic is selected for the design study, which is carried out by the entire class as a team effort. There are no exams or quizzes in this course, but the total output of the effort consists of three parts: a) a formal oral presentation at the end of the semester; b) a scale model of the design, unveiled at this presentation; and c) the final report at hand.

The course was initiated by the late Professor Wilbur C. Nelson in 1965. The 1988-89 design is the thirty-second in the series then begun.

The current design topic is of a manned, space-based Space Transportation Vehicle capable of satisfying a variety of anticipated future mission requirements. Its nominal mission is traveling between the Space Station *Freedom* in Low Earth Orbit and Geosynchronous Orbit to perform tasks such as delivering and retrieving payloads or carrying out satellite refurbishing and repair. Project Argo represents a preliminary sizing and configuration design of such a vehicle. The design considers two alternate versions: in one, the mission to and from the Space Station is carried out using rocket thrust only, while in the second version, the return to the home base takes place via a pass through the atmosphere, where aerodynamic drag replaces one application of rocket thrust, eliminating the need for the corresponding amount of propellant.

The Project Argo team consisted of 43 seniors. As is customary in this course, the students elected a Project Manager and Assistant Project Manager at the beginning of the semester and subsequently organized themselves in several technical groups, one for each of the major subsystems of the design; the work of each group is directed by a Group Leader. The Managers direct and control the team activity and integrate the many inputs from the groups into a single, coherent design. The concept of a system approach to design was carried throughout the design process.

A Final Report Committee, with representatives from each Group, was assigned the major task of integrating the inputs from the Team into this document. An ad-hoc Committee was formed to create the scale models of the two vehicles.

We gratefully acknowledge the support we enjoyed during the three-year tenure in the NASA/USRA University Advanced Design Program. Grants from the National Aeronautics and Space Administration and the Universities Space Research Association in Phase II of the Program have immensely contributed to the continued success of the course. The Grants provided funding for a graduate teaching assistant, for travel, for various administrative costs, for educational innovation, for construction of the scale models, and for reproduction and distribution of this report. Special recognition is due Mr. Steve Hartman and Ms. Elaine Schwartz from NASA Headquarters, Washington, D.C. and John R. Sevier, Director, Advanced Design Program, and Ms. Carol Hopf, Deputy Director, USRA, Houston, Texas.

NASA Lewis Research Center, Cleveland, Ohio, gave support in the form of key lecturers and other technical assistance; Dr. Karl A. Faymon and Ms. Lisa L. Kohout provided guidance and maintained contact with the design team during the year. We are thankful to them for their support and encouragement.

Professor Harm Buning April 17, 1989

Acknowledgements

There were several people who provided the Project Argo design team with vital input to whom we would like to express our sincere gratitude. From the NASA Lewis Research Center in Cleveland, Ohio, we thank Dr. Karl Faymon and Lisa Kohout for their continued support and guest lecture on power systems as well as Joe Hemminger and Al Pavli for their guest lecture on propulsion systems. We would also like to thank our guest lecturer and Adjunct Professor at the University of Michigan Jack R. Lousma for sharing his experiences as a former astronaut. Thanks also to Don Saxton and Norman Brown of the NASA Marshall Space Flight Center in Huntsville, Alabama and Dr. D. Stuart Nuchtwey of the NASA Johnson Space Center in Houston, Texas.

List of Commonly Used Acronyms

STV	Space Transportation Vehicle
ASTV	Aeroassisted Space Transportation Vehicle
CSTV	Chemical Space Transportation Vehicle (all-propulsive)
GEO	Geosynchronous Earth Orbit, where satellites are located
LEO	Low Earth Orbit, where Space Station *Freedom* will be located
OMV	Orbital Maneuvering Vehicle, a workhorse vehicle based at LEO
EVA	Extravehicular Activity, astronaut activity outside the ship
MMU	Manned Maneuvering Unit, an astronaut's "jet pack" to propel himself in space
RMS	Remote Manipulator System, the "Canadian arm" used on the shuttle to manuever objects in space
LOX	Liquid oxygen
LH2	Liquid hydrogen
SSF	Space Station *Freedom*
TPS	Thermal Protection System

Chapter 1

Introduction

1.1 **Project Introduction**

1.2 **Team Organization**

Class Photo

1.1 Project Introduction

1.1.1 Manager's Introduction

It has long been apparent that space is mankind's final frontier. Man travelling to the moon became reality with the Apollo program and the quest to conquer space was begun. The United States has been a major pioneer in this ongoing quest. The Space Shuttle program has provided a reusable means for man to reach Low Earth Orbit (LEO). The next step for the U.S. is to provide a permanent manned presence in space with the Space Station *Freedom*, scheduled to be completed by the late 1990's. After establishing a permanent presence in LEO, the next logical step would be to develop a mechanism to reach Geosynchronous Earth Orbit (GEO) reliably and repeatedly to deploy, repair, and even retrieve satellites and other payloads. Project Argo is a possible solution to address this need.

Project Argo is the design of a manned Space Transportation Vehicle (STV) that would transport payloads between LEO (altitude lying between 278 to 500 km above the Earth) and GEO (altitude is approximately 35,800 km above the Earth) and would be refueled and refurbished at the Space Station *Freedom*. *Argo* would be man's first space-based manned vehicle and would provide a crucial link to geosynchronous orbit where the vast majority of satellites are located. The vehicle could be built and launched shortly after the space station and give invaluable space experience while serving as a workhorse to deliver and repair satellites. Eventually, if a manned space station is established in GEO, then *Argo* could serve as the transport between the Space Station *Freedom* and a "Geostation." If necessary, modifications could be made to allow the vehicle to reach the moon or possibly Mars. The benefits that *Argo* would provide would be an instrumental part of mankind's continuing conquest of space.

Argo is named after the ship of the Greek adventurer Jason and his crew, the Argonauts. Jason needed to retrieve the golden fleece, which was guarded by a dragon, in order to receive his share of the kingdom. Their voyage was full of incidents, but they successfully completed their mission. Our ship, like Jason's, will face many obstacles such as the Van Allen radiation belts, solar flares, and extreme temperatures during reentry through Earth's atmosphere. Yet our *Argo* is also designed to successfully complete its mission as did Jason's.

Argo's nominal mission consists of transporting 10,000 kilograms to GEO and 5,000 kilograms back to LEO. However, the payload capacity can reach maximums of 21,000 kilograms up to GEO or 10,000 kilograms down provided that no payload is carried in the other direction. These limits far exceed any current satellite masses and allow for larger satellites in the future or multiple satellites per mission. The mission duration will be a maximum of seven days and will allow for the targeting of up to three satellites per mission. This time limit is long enough to accomplish our mission, yet short enough to conserve mass in areas such as life support and power. The crew will consist of two astronauts (or "Argonauts") who will have the capability of extravehicular activity (EVA). This capability is important in that it allows the repair of satellites in GEO, the security of successful deployment, and the addition to the experience of man in space. During the vehicle's flight, the crew will be able to correct minor problems without aborting the mission.

Project Argo is unique in that it actually consists of the design and comparison of two different concepts to accomplish the same mission. The first is an all-propulsive vehicle which uses chemical propulsion for all of its major maneuvers between LEO and GEO. The second is a vehicle that uses aeroassisted braking during its return from GEO to LEO by passing through the upper portions of the atmosphere. During this maneuver, the drag on the vehicle slows it down without the use of propellant. With the proper approach and control, the correct amount of speed is reduced and the vehicle arrives at LEO as if all the braking had been done by propulsive means. This procedure allows for enormous fuel savings, but produces many difficult design considerations such as protecting the vehicle and crew from the high heating rates and decelerations while still achieving the required reduction in speed. Instead of arbitrarily choosing one method over the other, we have designed two configurations capable of the same mission. As a result, we can compare and contrast the two concepts. Please refer to Figures 2.1 and 2.2 on pages 2-2 and 2-3 for illustrations and specifications of the two vehicles.

1.1.2 Mission Justification

The United States is moving into a new era of habitation and utilization of space. With the Space Station *Freedom* scheduled for deployment by the end of this century, our nation is ready to begin a permanent manned presence in space. The Space Station will provide us with many opportunities for new scientific research and technological advancements. The *Argo* would serve as a bridge between the Space Station and GEO. The main purpose of developing an STV would then be to complete missions beyond the range of the Space Shuttle.

At present, we see nominal missions of satellite delivery to GEO, satellite repair there, and the retrieval of satellites that need more extensive repair so they may be serviced at the Space Station. These missions will enable us to more efficiently maintain our satellites, which are important in our everyday life. Furthermore, if a satellite is no longer serviceable, the aeroassisted version of the STV could carry the dead satellite back down and release it into the Earth's atmosphere prior to reentry, causing the satellite to burn up. This process will open up more space in the already-crowded GEO without adding to the space debris problem.

In addition, *Argo* is better suited for delivering payloads than the current upper stage boosters. The boosters can fail, and their payloads would be lost. These expendable boosters are not only expensive, but also contribute to the increasing amount of space debris orbiting the Earth. The STV, on the other hand, would not lose the payload if the mission were aborted. Instead, the payload could be delivered on a future mission. Furthermore, after delivering the payload, the STV can go on to repair and retrieve additional satellites, all in the same mission, thus making it more efficient than a booster.

An important advantage of Project Argo is its flexibility to perform other missions. Using its Remote Manipulator System (RMS), *Argo* could be used to clean up some of the identifiable space debris. It may also be used as a rescue vehicle to transport a habitat module away from the Space Station in an emergency. A lunar and perhaps even a Mars mission are possibilities for the all-propulsive vehicle. A lunar base may be a future goal, and the STV would already be developed.

In conclusion, the world relies heavily on communications and other satellite services everyday. Project Argo can work to keep those services in use and hence improve our quality of life on Earth. And as man moves to a new frontier, *Argo* will be there to improve the quality of life in space, wherever it may be.

1.1.3 Mission Scenario

In order to get a better understanding of Project Argo and its uses, picture this sample mission. First, the heavy-lift launch vehicle brings the necessary fuel to LEO. Next, the Orbital Maneuvering Vehicle (OMV) brings the fuel to the Space Station Freedom to refuel *Argo*. Then the Space Shuttle *Discovery* is launched with a "new and improved" modular satellite that *Argo* is to take to GEO to replace an older, outdated satellite. After the shuttle docks at the station, the satellite is transferred from the shuttle to *Argo* using the station's remote manipulator system (RMS) arm. Next, *Argo* departs from the Space Station *Freedom* and heads for GEO on a direct elliptical path that targets the position of the satellite to be replaced. Six hours later, *Argo* is in GEO in perfect position to perform the first portion of its mission. One of the Argonauts enters the airlock, suits up, and leaves the vehicle to deploy the new satellite and at the same time retrieve the old one. Next, *Argo* performs an orbit-walking maneuver to rendezvous with another satellite that merely needs a module replaced. The next day, an Argonaut again performs an EVA to fix the defective satellite and ensure it is working properly. Finally, after a full mission's work, the Argonauts head for home. They again take a direct elliptical path back to LEO (possibly using the benefits of aeroassisted braking depending on the vehicle) and perform a phasing maneuver to synchronize themselves with the Space Station. Lastly, *Argo* docks with the Space Station *Freedom* in its own specially-designed hangar connected to the habitats. With the completion of the successful mission, *Argo* is refurbished and awaits its next departure which is within a month's time.

1.1.4 Comparison of the Two Concepts

An important aspect of Project Argo is the study of the differences between the all-propulsive and aeroassisted versions. Of course, the major advantage of the Aeroassisted STV (ASTV) is the fuel that it saves. We calculated that the aeroassist technology can save from 16% to 34% fuel, depending upon how much payload mass is moved up and down. Hence, our estimates have shown that the operational costs of the ASTV are significantly lower than those of the Chemical STV (CSTV). Thus, this savings easily makes up for the higher development cost of the ASTV. However, there is more to consider than just cost when comparing the two designs. Below, Table 1.1 summarizes the advantages of the ASTV and the CSTV.

ASTV

<u>Advantages</u>
•Fuel savings => i.e.,
 lower operational cost
 less fuel tank volume & mass,
 lower cost to transfer fuel than with CSTV
•Less engine burn => longer engine life than with CSTV

CSTV

<u>Advantages</u>
•Larger, less-constrained payload area than the ASTV
•Is easier to configure for other missions than the ASTV
•Lower g-loading than the ASTV
•Lower heating values than the ASTV
•Lower stresses on vehicle than ASTV
•Don't have additional mass of the aerobrake
•Technology is known

Table 1.1 Advantages of the ASTV and CSTV

There is indeed a trade-off between the two vehicles that is nearly impossible to quantify. Deciding which is superior depends on the goals of the current mission as well as long-range goals. If a vehicle is desired to carry payloads to GEO in the most fuel-efficient manner over the next ten years, then the ASTV may be the best choice. However, if the long-range goal of the vehicle is to shuttle payloads of varying size and shape between LEO and a lunar base, than perhaps the CSTV will be superior. Therefore, there are two solutions to the problem.

1.2 Team Organization

Aerospace Engineering 483 is the senior space system design class. This year there are 43 students enrolled in the class to tackle Project Argo--the design of a space-based, manned space transportation vehicle. The class structure consists of the Project Manager, Assistant Project Manager, and eight design groups. The managers are responsible for delegating tasks, coordinating group efforts, scheduling the agendas for the class meetings, and insuring that tasks are completed in a timely fashion. They also oversee the final report and final presentation. Each group has a group leader who has similar responsibilities at a group level. The team structure is shown in Figure 1.1 followed by a group photo and team roster.

The team meets three hours, twice a week. While some meetings include a guest lecturer, most meetings are dedicated to updating and integrating each group's contributions to the overall design. What follows is a brief description of the main duties of each group.

The Spacecraft Configuration and Integration group develops the baseline configuration of each version of the vehicle. This group must interact heavily with all the other groups to come up with the size, mass, and overall picture of both an all-propulsive and an aeroassisted version of *Argo*.

The Mission Analysis group must plan the nominal mission. The main task that falls into this category is designing all of *Argo*'s maneuvers between LEO and GEO as well as any "fine tuning" maneuvers required to position the vehicle in the correct location along the orbit. Mission Analysis must also calculate the required amounts of velocity change and time to complete the nominal mission.

The Atmospheric Flight group designs the aerobrake and thermal protection system of the aeroassisted version of *Argo*. The group must study the effects of lift, flight path, heating rates, and decelerations on various aerobrake configurations and determine the best candidate for the needs of *Argo*'s mission.

The Propulsion group determines the main engines and propellant requirements. Their charge is to select a method of propulsion, the number and type of engine to be used, as well as the propellant storage configuration. Propulsion is also responsible for designing the attitude control system to keep *Argo* stable and allow for subtle maneuvers.

The Power and Communications group designs the power system. Their main task is to come up with a suitable means of generating and distributing power. Other duties of this group include developing a radiation mechanism for dissipating excess heat as well as designing the communications, the avionics, and the data management systems.

The Life Support and Human Factors group deals with all issues concerning the comfort and safety of the crew. These issues include cabin atmosphere, cabin design, food and water supply, radiation protection, and maximum decelerations incurred.

The Logistics and Support group is responsible for maintaining *Argo* and preparing it for the next mission. The group decides on a method of docking as well as refueling and refurbishing the vehicle. As a result, the group must define how *Argo* will impact the Space Station *Freedom* and what additional provisions should be made.

The System Analysis group analyzes both versions of *Argo*. The group defines and weighs the advantages and disadvantages of both the all-propulsive and the aeroassisted vehicles. It also determines the overall cost of the vehicle as well as critically defines and justifies *Argo*'s mission.

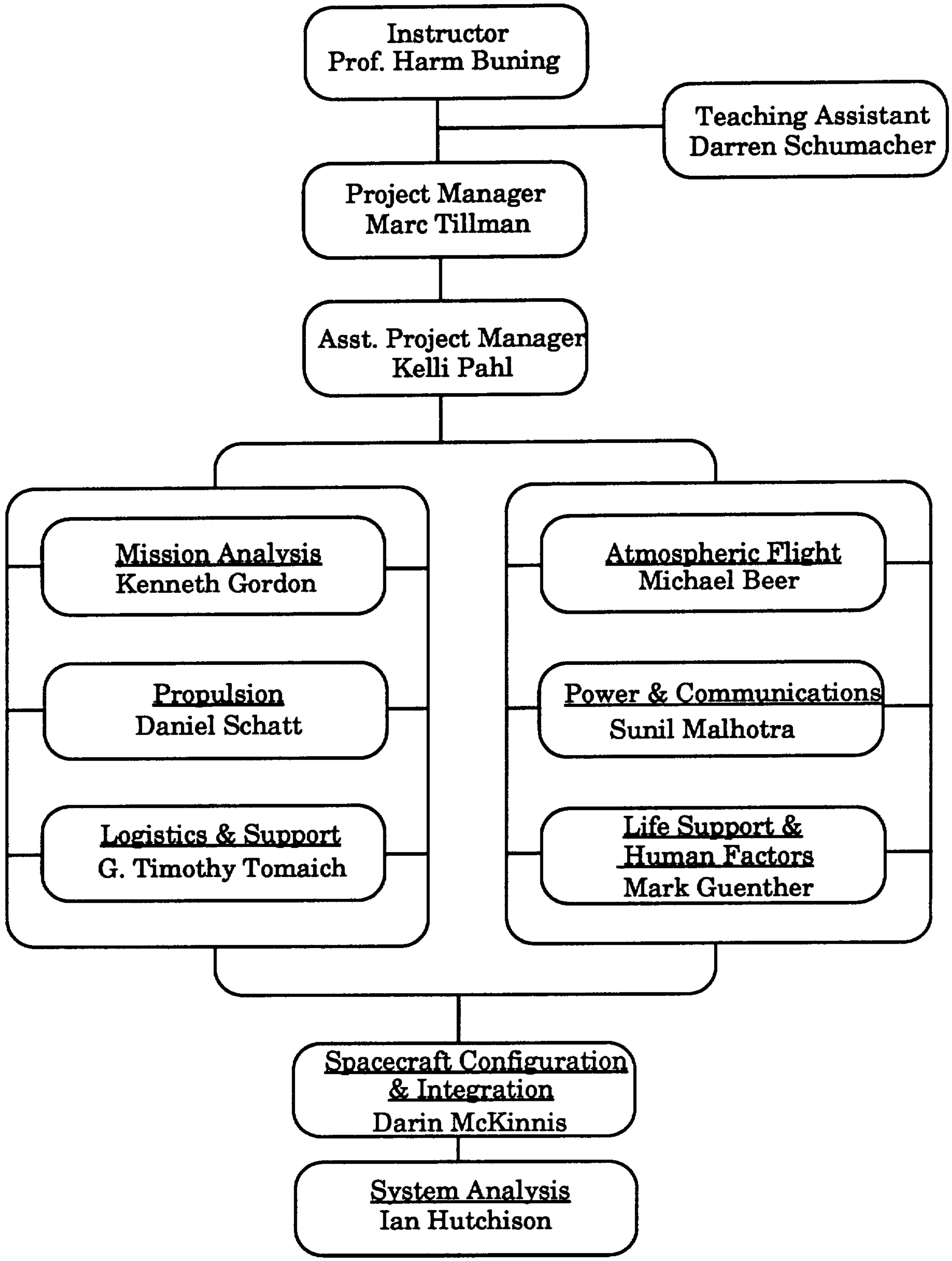

Figure 1.1 Team Structure

Aerospace Engineering 483
Project Argo
Team Roster

Instructor: Prof. Harm Buning
Teaching Assistant: Darren Schumacher
Project Manager: Marc Tillman
Assistant Project Manager: Kelli Pahl

Spacecraft Configuration & Integration
Dick Barkman
Alan Folz
Mark Gawronski
Darin McKinnis*
Patrick Murdock

Mission Analysis
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Brian Boluyt
Conrad Chu
Kenneth Gordon*
Jeff Longcore
Douglas Seifert
Eric Wegryn

Atmospheric Flight
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Michael Mileski
Gerald Padnos
Timothy Smith

Propulsion
Rodger Dotson, Jr.
Walter Fournier
Jon Quigley
Elliot Sala, Jr.
Daniel Schatt*
John Waidelich, Jr.

Power & Communications
Stephen Gebes
Daniel Herr
Michael Kocsis
Sunil Malhotra*
Douglas Prince

Life Support & Human Factors
Gerardo Artache
Robert Dillman II
Robert Gagnon
Mark Guenther*
Howard Wang

Logistics & Support
Grenmarie Agresar
Chip Beebe
Henry Caffrey
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* denotes Group Leader

Ad Hoc Committee for the Final Report

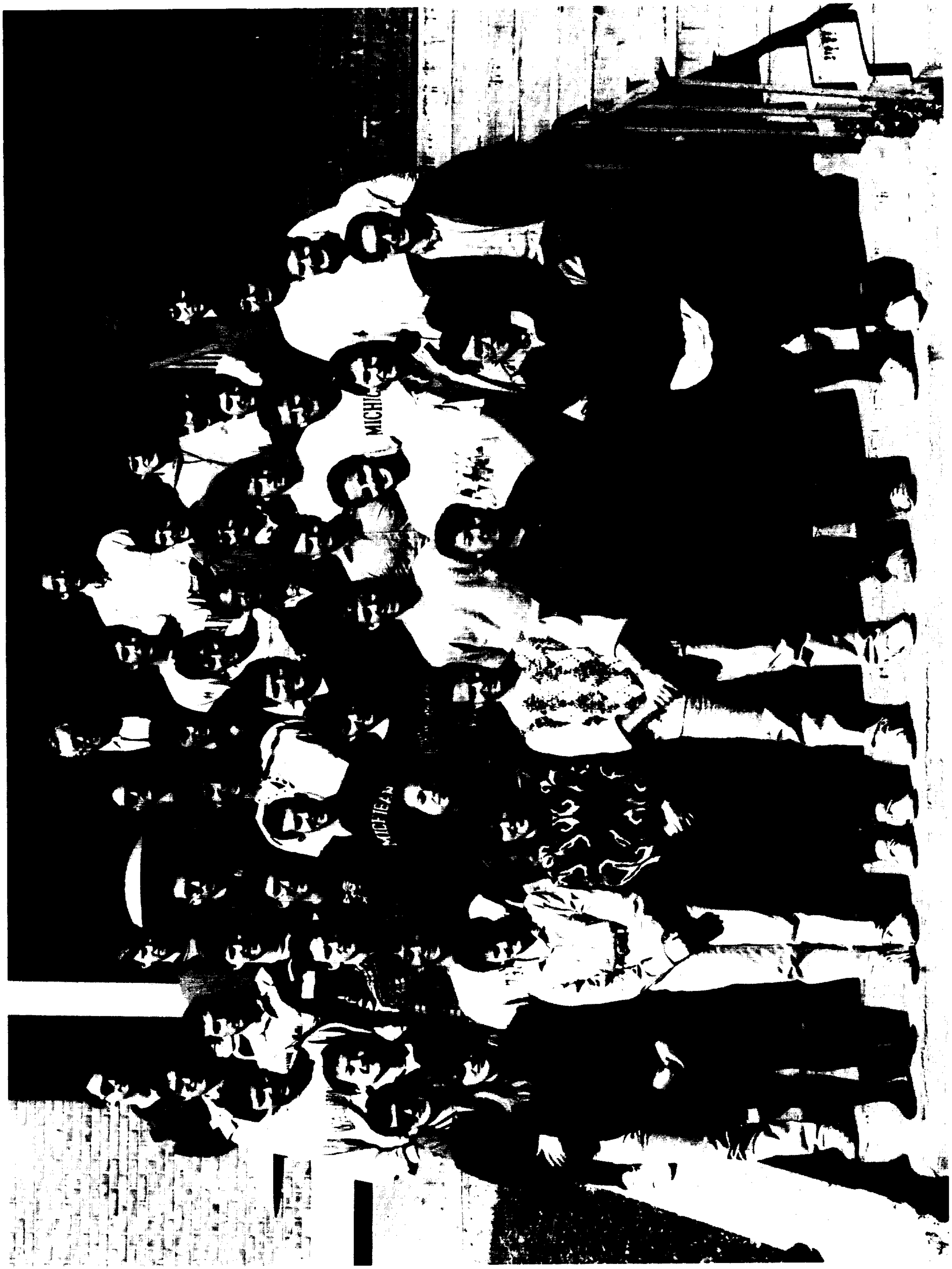
Chairman: Howard Wang

Douglas Seifert
John Waidelich
Michael Mileski
Daniel Herr
Matthew Wilks
Glenn Law
Alan Folz

Ad Hoc Committee for the Models

Chairman: Darin McKinnis

Grenmarie Agresar
Chip Beebe
Conrad Chu
Mark Gawronski
Ian Hutchison
Jeff Longcore
Patrick Murdock
Gerald Padnos
Jon Quigley
Eric Wegryn
Matthew Wilks



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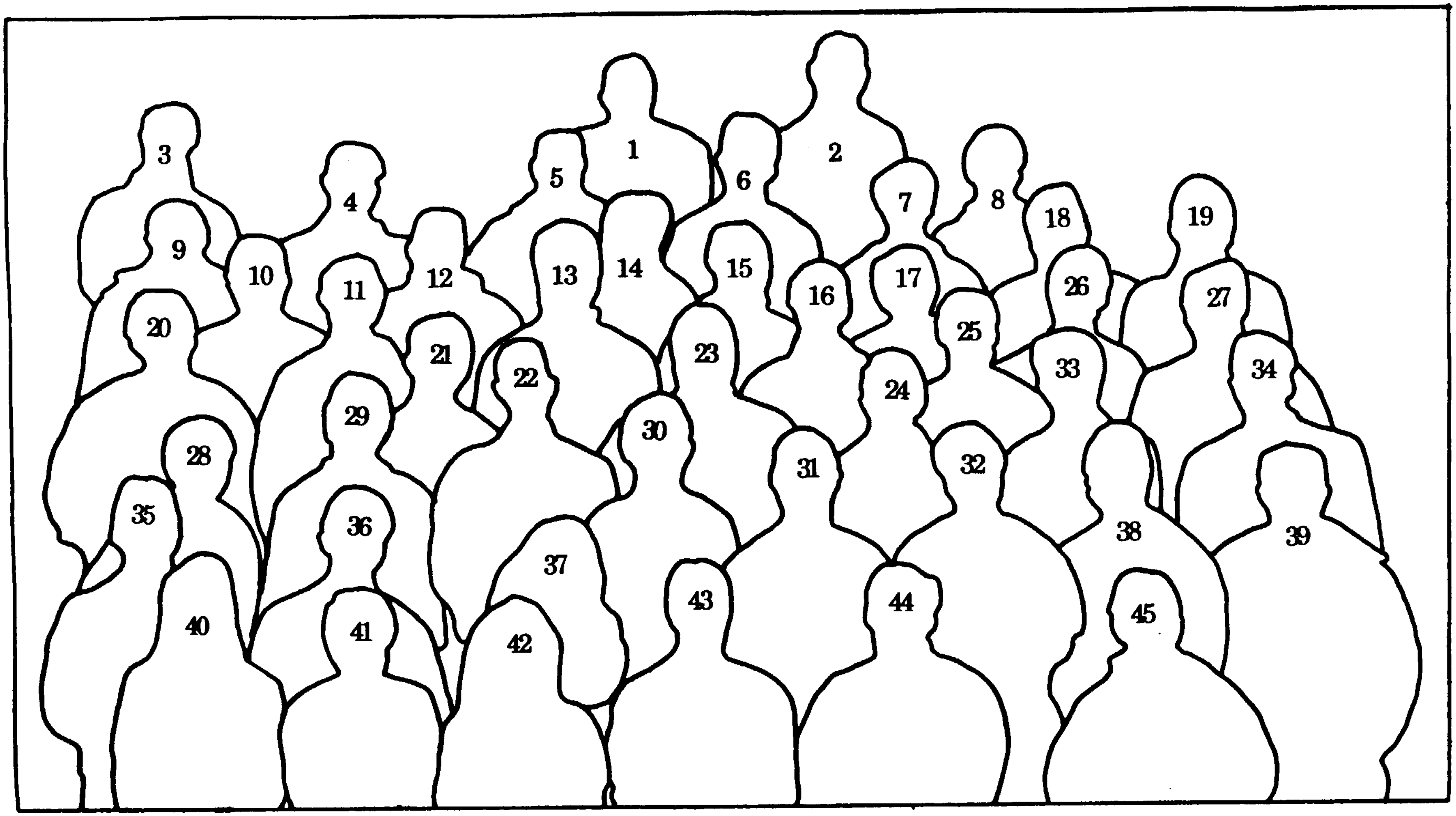

AERO 483 University of Michigan Aerospace System Design, Winter 1989

1. Prof. Harm Buning
2. Marc Tillman
3. Jon Quigley
4. Elliot Sala
5. Walter Fournier
6. Kenneth Gordon
7. Steve Gebes
8. Derek Paige
9. John Waidelich
10. Tim Tomaich
11. Matt Wilks
12. Jerry Artache
13. Ian Hutchison
14. Rodger Dotson
15. Mark Guenther
16. Sunil Malhotra
17. Alan Folz
18. Pat Murdock
19. Mike Mileski
20. Darren Schumacher
21. Glenn Law
22. Bob Dillman
23. Henry Caffrey
24. Mike Kocsis
25. Doug Prince
26. Dan Herr
27. Tim Smith
28. Dan Schatt
29. Dick Barkman
30. Jeff Longcore
31. Chip Beebe
32. Darin McKinnis
33. Doug Seifert
34. Brian Boluyt
35. Eric Wegryn
36. Howard Wang
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38. Mark Gawronski
39. Gerry Padnos
40. Suzanne O'Donnell
41. Jenny Allen
42. Kelli Pahl
43. Rob Gagnon
44. Conrad Chu
45. Mike Beer

Chapter 2

Spacecraft Configuration and Integration

2.0 Summary

The Spacecraft Configuration and Integration Group is responsible for integrating the various subsystems into a cohesive, efficient vehicle and for developing the vehicle configuration. Both the CSTV and the ASTV were designed with a premium placed on vehicle flexibility. Figures 2.1 and 2.2 summarize the main features of the CSTV and ASTV respectively.

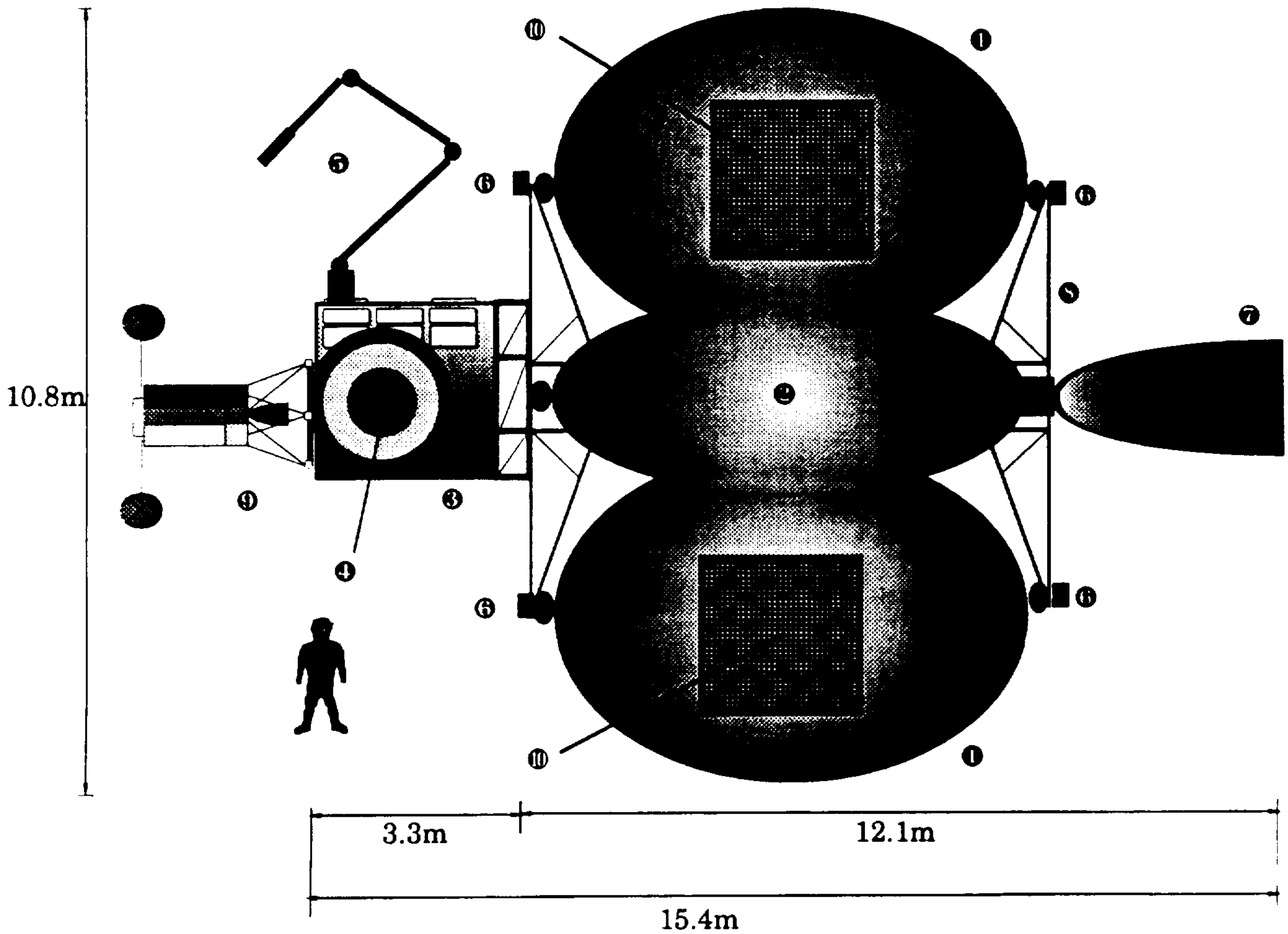
CSTV

The main feature of the CSTV is its modularity. This configuration contains three modules or components: the propulsion module, the command module, and the payload and servicing module. These modules are independent of each other so that, for example, the command module can support the crew even when disconnected from the other modules. This modular approach greatly increases the flexibility of the vehicle and its potential for future growth.

The propulsion module (PM) includes the Rocketdyne RS-44 main engines, the propellant tanks of liquid oxygen (LOX) and liquid hydrogen (LH2), as well as the support structure for these components.

Forward of the propulsion module is the command module (CM) which will house a crew of two for a nominal mission of up to 7 days, or 11 days in an emergency situation. The CM contains the auxiliary tanks which supply LH2 and LOX for the power generation system and the oxygen for the crew atmosphere. The CM also includes a specially designed airlock which also serves the vehicle as the berthing interface to the Space Station and as a radiation storm shelter in the event of solar flare.

The payload and servicing module (PSM) is located forward of the CM to accommodate the greatest variety of potential payloads. In this location there is virtually no restriction in shape or size of payloads. The PSM will be selected from a variety of modules stored and maintained at the space station. Pallets, such as those used aboard the Space Shuttle, and trusswork supports are two such possibilities. However, any user can use an existing PSM or, if necessary, design his own PSM to suit his particular needs. Although a satellite designer will need to design to meet the vehicle constraints imposed by the launch vehicle, he will not need to design to meet any additional constraints for the STV. This greatly increases the payload handling capabilities of our vehicle and eases the work of payload designers.



SPECIFICATIONS

Crew: 2
Mission Duration: 7 days
Dry Mass: 6341 kg
Gross Mass: 86,566 kg

Payload up: 10,000 kg
Payload down: 5000 kg

Engine: Rocketdyne RS-44 (2)
Thrust: 67,000 N each
Fuel: LH2/LOX

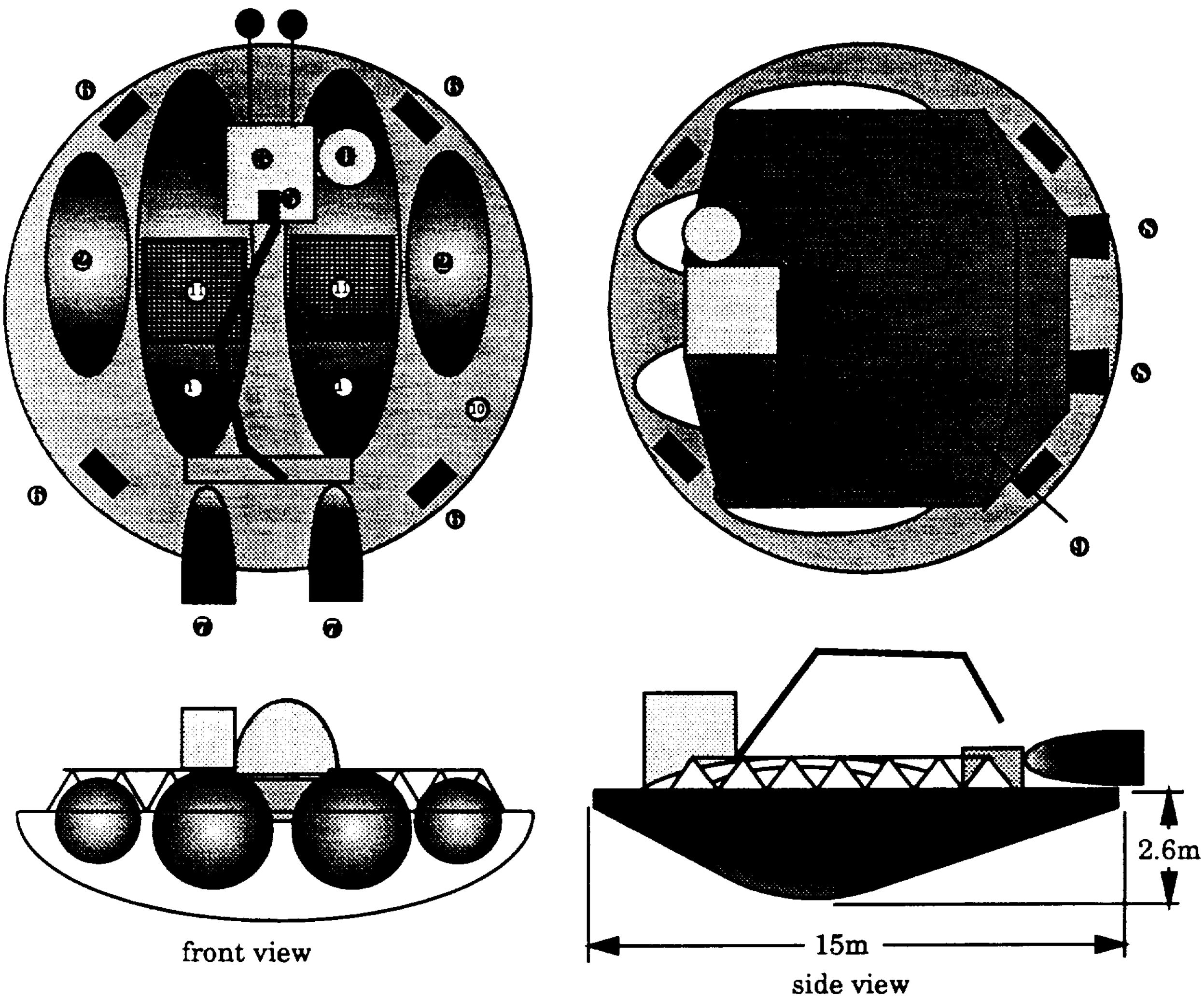
LH2 Tank: 4.5 x 8 m Elliptical
LOX Tank: 3 x 7 m Elliptical

Propulsion Mod.: 12.1 x 9 x 10.8m
Crew Mod.: L = 3.3m, D = 2.5 m

INDEX

1. Liquid Hydrogen Tanks (2).
2. Liquid Oxygen Tanks (2).
3. Command Module.
4. Airlock/Storm Shelter.
5. Remote Manipulator Arm.
6. RMS unit (4).
7. RS-44 Engines (2).
8. Thrust Structure/Tank Supp.
9. Payload mounted on pallet.
10. Radiators.

Figure 2.1: CSTV Design



SPECIFICATIONS

Crew: 2
Mission Duration: 7 days
Dry Mass: 7841 kg
Gross Mass: 66,321 kg

Payload up: 10,000 kg
Payload down: 5000 kg

Engine: Rocketdyne RS-44 (2)
Thrust: 67,000 N each
Fuel: LH2/LOX

LH2 Tank: 3.4 x 11 m Elliptical
LOX Tank: 2.5 x 7 m Elliptical

Aerobrake: 15m dia., 2.6m deep
Crew Module: 3 x 2.6 x 2.6 m

INDEX

1. Liquid Hyrdogen (LH2) tank.
2. Liquid Oxygen (LOX) tank.
3. Crew Cabin.
4. Airlock/Storm Shelter.
5. Remote Manipulator System (RMS).
6. Reaction Control System (RCS) (4).
7. Rocketdyne RS-44 Engines (2).
8. RS-44 Engines. Retracted position.
9. Payload Plate.
10. Aerobrake.
11. Radiators.

Figure 2.2: ASTV Design

ASTV

The ASTV must provide thermal protection for itself and its payloads when passing through Earth's atmosphere. A spherical raked-cone aerobrake using advanced Space Shuttle tiles provides this protection for the ASTV. This is the same system being employed by the Aeroassisted Flight Experiment (AFE) which is to be tested in 1991. Seated within the aerobrake are the main propulsion tanks of LH2 and LOX. These tanks are positioned as deep as possible in the aerobrake to increase stability of the vehicle and to provide the largest possible volume for payloads. In this position the tanks are also protected from space debris and micrometeoroids by their proximity to the hard and encompassing aerobrake. A payload platform of composite trusswork is located above the tanks to protect them and to provide secure handling of the payloads.

The command module (CM) sits atop the tanks to provide the crew with visibility of the payload platform and to permit easy berthing with the Space Station. The CM is very similar in design to the CSTV CM but takes into account the g-loading effects of aerobraking on the crew.

The ASTV uses Rocketdyne RS-44 engines, the same as those used on the CSTV, but with an important added feature. The engine nozzles on the ASTV are capable of retraction and extension, allowing the nozzles to extend beyond the lip of the aerobrake during all phases of flight except the aerobraking maneuver. The nozzles are retracted during aerobraking to protect them from the extreme temperatures. This feature allows the engines to be located as shown on Figure 2.2 where they do not adversely affect payload capability. Furthermore, the engines fire parallel to the major axes of the propellant tanks, minimizing the need for engine gimballing and increasing vehicle stability.

Overview

The previous discussion provides a summary of the main features the CSTV and ASTV configurations. The interested reader will find in the following discussion the evolutionary process by which the final configuration was selected and developed. The details of each design are then provided. The CSTV is presented in Section 2.2 and the ASTV follows in Section 2.3.

2.1 Design Rationale

In configuring these two vehicles, the most important consideration is the role or mission the vehicle is to fulfill. The role of the STV is to provide services and support to various spacecraft. In configuring our vehicles we have taken this role into account. Rather than designing a highly integrated vehicle we chose to design vehicles which possess the most flexibility possible and provide the greatest array of services to their users. Thus, while seeking to make our vehicle as fuel efficient and cost effective as possible, we maintained this goal of vehicle flexibility as our highest priority.

Servicing the wide variety of current spacecraft and providing for the needs of future spacecraft demands that the STV be flexible and versatile. It should not place unnecessary restrictions on its payloads and it should meet or exceed the predicted requirements of future users. For example, we have sought to maximize the payload capability of the STV so that very large volumes can be accommodated without difficulty.

In the future the STV may be asked to complete missions for which it is not nominally designed. The payloads transported will be of greater or less mass. The STV may transport payloads to locations other than GEO, such as polar orbit, the moon, or beyond. The STV may be utilized to carry personnel other than its crew to these various locations or be used as the building block for some larger space vehicle. The STV will have many uses, will serve many users, and will probably be used in some manner that cannot be anticipated. Hence, the servicing role of the the STV requires that it be as flexible to variations from the nominal mission as possible. Thus, in configuring these vehicles we have strived to create the most flexible design possible.

2.2 Chemical Space Transportation Vehicle

2.2.1 Candidates

The all-propulsive design (designated CSTV for Chemical Space Transportation Vehicle) is driven by mass efficiency, versatility, and growth. Except for logistic and center of mass considerations, the geometry of the vehicle is free to follow an efficient, functional shape. The following is a list of the designs we used as a basis from which we could determine the configuration that would best accomplish our goals.

Front End Loading

Front end loading is attractive because it places the fewest restrictions on the geometry of the payload. The candidates shown in Figures 2.3 and 2.4 vary chiefly in the extent of their modularity. Figure 2.3 is a three piece design including a propulsion unit, command module, and payload. The more elaborate design in Figure 2.4 adds a separate avionics ring and an Apollo-style emergency re-entry vehicle. Segmenting the components in this manner allows greater modification to the basic structure, but at the expense of additional mass. The designs show two of the tank configurations that the group examined. The vehicle of Figure 2.3 uses a central truss to support two-point tanks, while Figure 2.4 encloses cylindrical tanks in a longer outer structure.

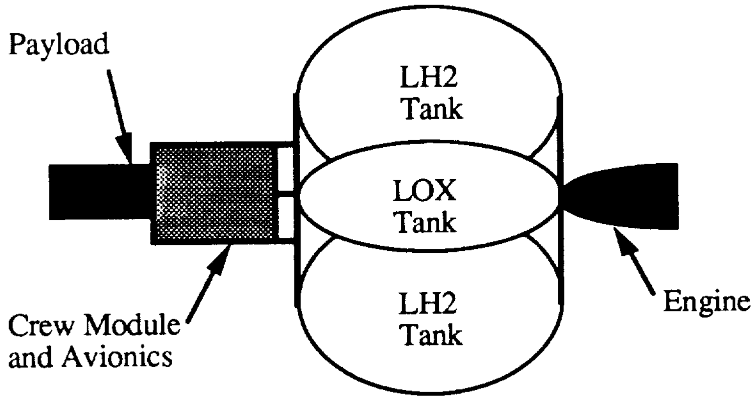


Figure 2.3

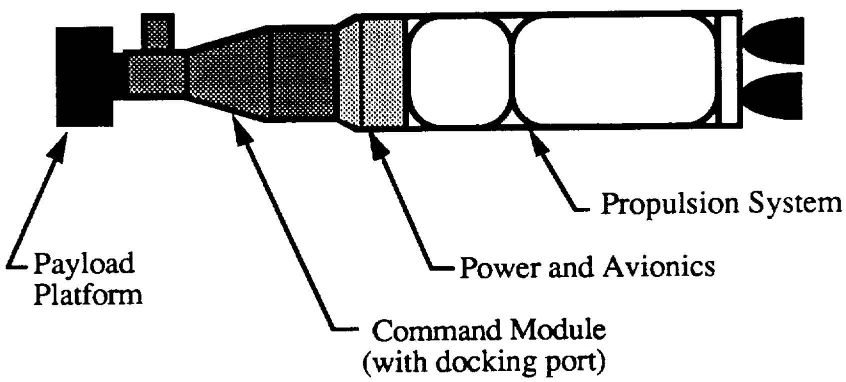


Figure 2.4

Middle Loading

The configuration shown in Figure 2.5 is the same as that in Figure 2.3, except that the payload is carried, released, and retrieved in a simple cargo bay made of trusses. This is similar to the layout used on the shuttle, with the command module forward of the payload. The tank design of Figure 2.4 could also be used, but modularity would suffer because of the long umbilicals that would run from the propulsion unit to the CM. Use of a mid-loading payload bay would add fixed weight, because it must be designed to carry the heaviest, bulkiest mission planned in addition to transmitting thrust loads to the CM.

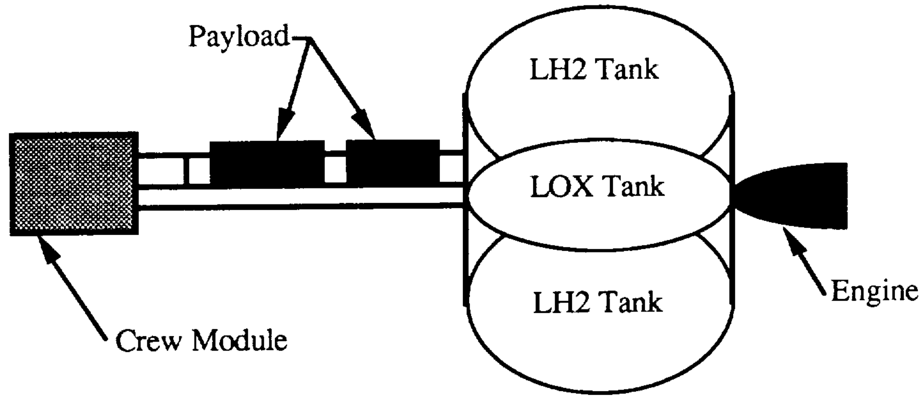


Figure 2.5

Ring Design

This interesting alternative, shown in Figure 2.6, is a large ring in which the payload is in the center, surrounded by the craft. This creates a sort of hangar, where the payload can be worked on. It has four engines and a CM located in the top half of the ring. The ring design would restrict the size of the payload that can be transported, and would require much more mass and integrated design than the other configurations.

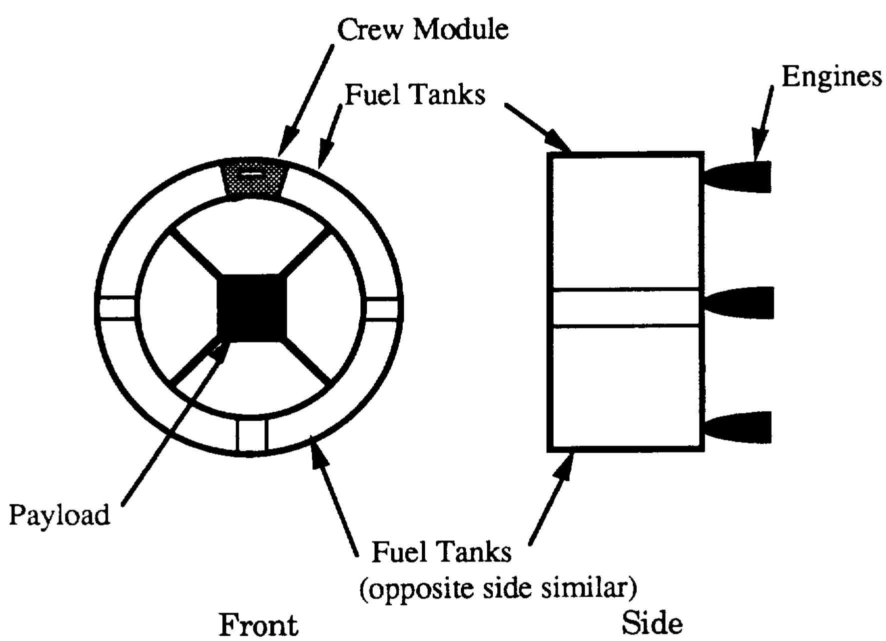


Figure 2.6

2.2.2 Design Selected

The CSTV was chosen to be a modular vehicle with a front-mounted payload and elliptical tanks, as shown in Figure 2.3. We determined that this design has best fulfilled our goals of mass efficiency, versatility, and growth.

The main propellant is stored in four low-pressure, elliptical tanks. Elliptical tanks are used at the request of the Propulsion Group and the rationale for this decision is given in Chapter 5, section 5.2. Integrated tanks would use more mass to enclose the the same volume. Tank integration also uses valuable space around the crew cabin which is needed for airlock placement and EVA activities. For redundancy, we will have two liquid hydrogen and two liquid oxygen fuel tanks in the configuration shown in Figure 2.1. These will supply the fuel to the two Rocketdyne RS-44 rocket engines. The fuel tanks and the engines will be attached and supported by a single, central frame. [ref. 2.1] The frame is lightweight yet sturdy enough to bear the loads of the engine thrust. Reaction control thrusters are also located in this section of the STV with their fuel included in the main tanks. They are attached to the PM structure at the ends of the liquid hydrogen tanks, giving the maximum moment arm for maneuvers.

The interface between the PM and CM will contain the connections for main engine and thruster control. The reactants for the fuel cell power system, as well as a life support tank of nitrogen, will be located here. Also, the interface can release the CM from the PM in case of emergency and for maintenance purposes.

The CM will be cylindrical in shape to reduce the amount of surface area needed to be covered by radiation shielding. The CM structure will consist of ring frame made of graphite epoxy. This will be covered by aluminum which will serve as a pressure vessel as well as radiation shielding. The CM will support two crew members for up to 11 days. Details of its interior design and layout are given in Chapter 7. Externally mounted equipment includes the RMS, the airlock, antennae, and MMU. Most of the remaining surface will be covered by radiators for thermal control. A combination airlock/docking port/storm shelter tube will be located on the side of the CM. The docking port will face away from the module for docking to the Space Station. For protection from solar flares, a heavily shielded storm shelter is included in the docking tube. Its position here does not take away space from the crew, and the combination of all three components saves overall mass.

Adding to the CSTV's modularity, the command module will be able to separate from the propulsion unit. This allows for mating to a second propulsion unit, adding a second fuel tank package (see Figure 2.10c), and emergency operations.

Airlock

The airlock on *Argo* has three functions: to provide for berthing to the station, to allow EVAs, and to shelter the crew during solar flares. The berthing operation requires a standard berthing ring as shown in Figure 2.7. [ref. 2.2] The ring weighs 138 kg and is 2.25m wide at the base. Since this is wider than the airlock, it requires a flange to be placed on the end of the airlock cylinder. EVAs are performed through the forward-facing hatch, which also provides a second exit. The storm shelter is a thick-walled portion of the cylinder just large enough to hold the crew, emergency provisions, and fundamental ship controls. To close off the storm shelter, a sectional wall is used. The wall sections and auxiliary control modules are stored on the hatch faces. Lightweight, fold-down plastic seats are provided if the crew wishes to use them. For further operation of the storm shelter, see Chapter 7.

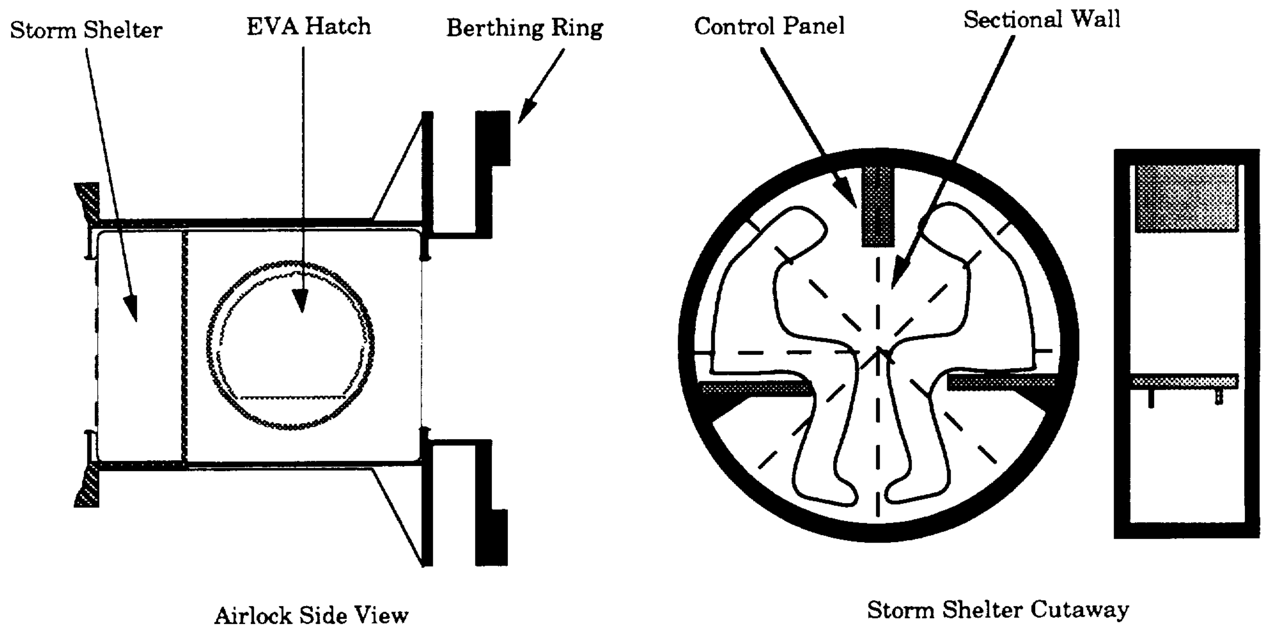
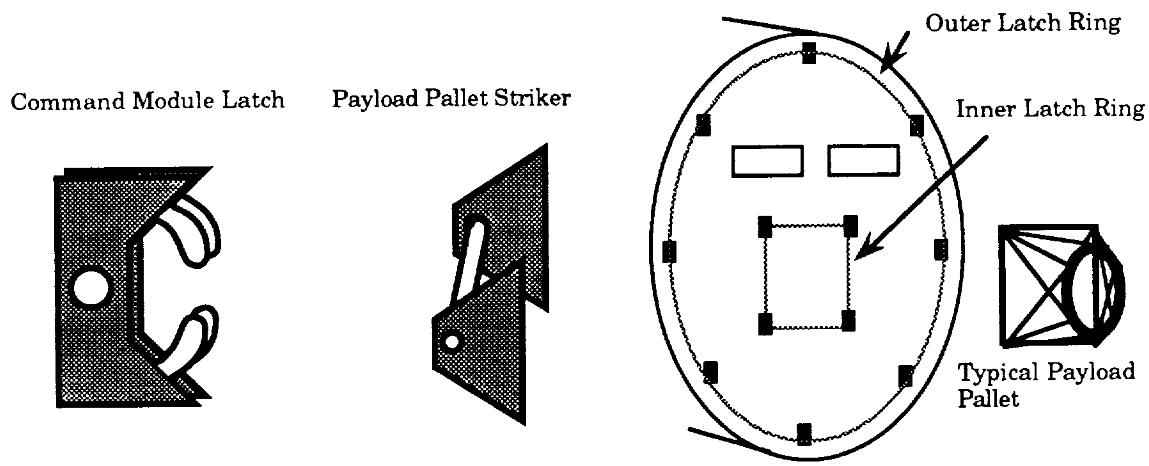


Figure 2.7

Payload and External Operations

The payload will be carried on the front of the command module and held in place with customized pallets. The pallet will act as an interface between the mounting hardware on the command module and the satellite. The base configuration includes a simple, one-satellite pallet. Mounting hardware on the command module consists of a set a latches which catch striker bars located on the payload pallet, similar to the way in which the berthing ring works. This operation is illustrated in Figure 2.8. These latches can be operated by solenoids or by mechanical release. Because the size of the CSTV's cargo is variable, a second set of latches is provided for small payload pallets. For more than one payload, extended pallets can be added as discussed in Section 2.4.



Latch-Striker Mechanism

Latch Placement

Figure 2.8

For handling the payload, a Remote Manipulator System and a Manned Maneuvering Unit will be used. This combination has been proven in past shuttle missions. The first step in deploying a satellite is to connect the RMS to the satellite. The satellite is then released it from its pallet. Release can be accomplished using mechanical latches or pyrotechnic bolts. The satellite is placed in orbit either by the RMS or the MMU. For retrieval, the MMU will rendezvous with the satellite and bring it in close enough for the RMS to grapple it. Then the arm will put the satellite in place on the pallet where the astronaut will anchor it.

2.2.3 Mass Analysis

In looking at the masses of the CSTV, we see that the propulsion component is the most massive. Although the fuel tanks alone are light structures, the fuel they contain is enormously heavy. This will tend to keep the center of mass towards the rear of the CSTV. This is favorable for stability since the thrust vector will pass through the center of mass. A slight error in alignment will not introduce moments to turn the CSTV.

As the fuel tanks empty, the center of mass will move rearward toward the engines as the fuel is pushed back due to the thrust of the engines. Then, as more fuel is burned and the PM is becoming lighter, the center of mass will move in the direction of the crew module. After all impulse burns when the CSTV is nearing the Space Station, the fuel tanks will contain only reserves and residuals. This will make the propulsion component much less massive, and the center of mass will be closer to the crew module. The following calculations concern the axial center of mass only. Due to certain components and variable payloads, the center of mass may be slightly off the centerline in the lateral direction. This can be adjusted for by gimballing the engines.

In calculating the center of mass, we took each major component of the vehicle as a simple geometric shape. [ref. 2.2] Then, knowing the mass of each component, we measured the distance of the component's center of mass from a reference point (we used the front of the crew module). Then we multiplied this distance by the component mass and did this for each component. We then summed up those figures and divided it by the total mass. This gave the distance from the reference point to the overall center of mass. The formula is $\Sigma XiMi / \Sigma Mi$, where Xi is the distance from the reference point of each component center of mass and Mi is the mass of the component. The figures we calculated are given below:

Beginning of mission (fuel tanks full):	6.3 m
With 500 kg payload	6.2 m
End of mission (reserves and residuals left):	4.7 m
With 500 kg payload	4.3 m
Tanks half full (assume fuel in rear half of tanks):	7.1 m
With 500 kg payload	6.9 m

All distances are measured rearward from the front edge of the command module.

2.2.4 Possible Modifications for Future Missions

The CSTV's modularity allows easy expansion for future missions while adding a minimum of extra weight to the base configuration. Multiple deployment and retrieval missions simply require a larger payload pallet. Manned expansion modules are treated essentially as payloads and are docked on the payload mounts. Heavy-lift and autonomous configurations would require extra hardware, but installation should be straight-forward.

Multiple Payload Deployment and Retrieval

Figure 2.9 shows payload pallets designed to carry two or more satellites. This is the extent of the modifications required to the base configuration.

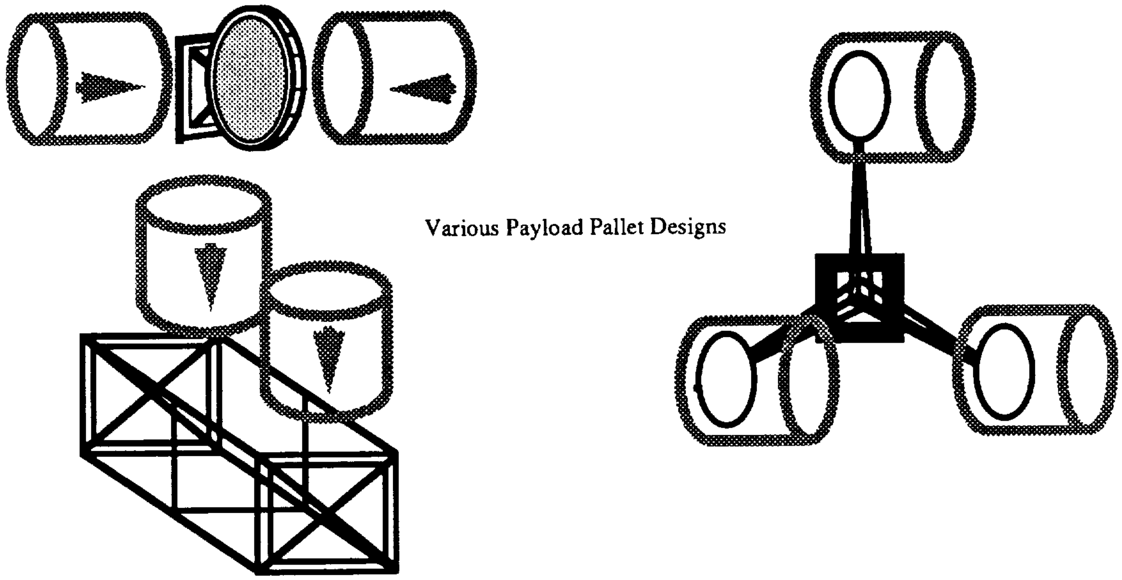


Figure 2.9

Manned Modules

Scientific or space construction missions (Figure 2.10a) may need additional pressurized workspace. Replacing the payload with a manned module would require structural change only in the airlock. To provide a connection between the modules, a tunnel must be run from the EVA hatch to the secondary module. The EVA hatch would need to be fitted with a sealing mechanism for this purpose.

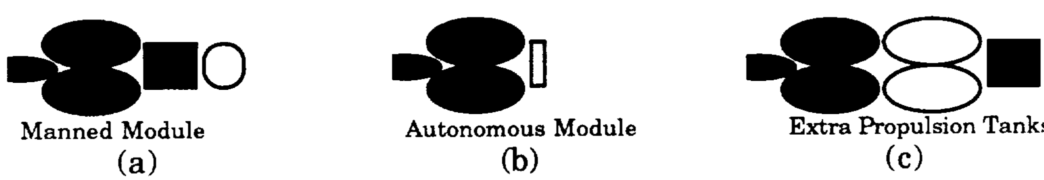


Figure 2.10

Autonomous Missions

Simple or hazardous missions might be accomplished without a crew. Replacing the command module with a small avionics package (Figure 2.10b) would reduce the mass of the vehicle considerably, resulting in larger payload capabilities or fuel savings.

Heavy-Lift Missions

Heavier payloads could be carried with the addition of more fuel. The modular design facilitates placing an extra set of tanks between the propulsion unit and the command module (Figure 2.10c). Modification to the propulsion unit entails running a set of LOX and LH2 feed lines from the engines, through the central truss and to the umbilical plate.

2.3 Aeroassisted Space Transportation Vehicle

Two primary requirements exist in the design of an ASTV: first, any structure or payload must remain within the impingement cone created by the aerobrake; second, the overall center of mass of the vehicle must remain fore of the metacenter to maintain aerodynamic stability and control. Any design in which *Argo* would fail to meet these restrictions cannot be considered.

When *Argo* enters the atmosphere for the aerobraking segment of the mission, the flow behind the aerobrake (or shield) will constrain the size and shape of the payload being returned to LEO. The vacuum formed behind the ASTV will be conical in shape, and the parameter which characterizes this impingement cone is the "base turning angle," the angle between the freestream flow vector and the line connecting the edge of the shield with the reattachment of the flow. This angle is approximately fifteen degrees for the shield design used in Project Argo, the spherical raked-cone. [ref. 2.3] *Argo* will fly at an angle of attack of five degrees to achieve lift for control purposes, and a margin for error is desired, so all components of the vehicle and its payload down are required to fit in an impingement cone defined by a clearance angle of twenty-five degrees. This clearance angle is measured from the set of lines perpendicular to the base of the brake and extending from its edge. These angles are indicated in Figure 2.11.

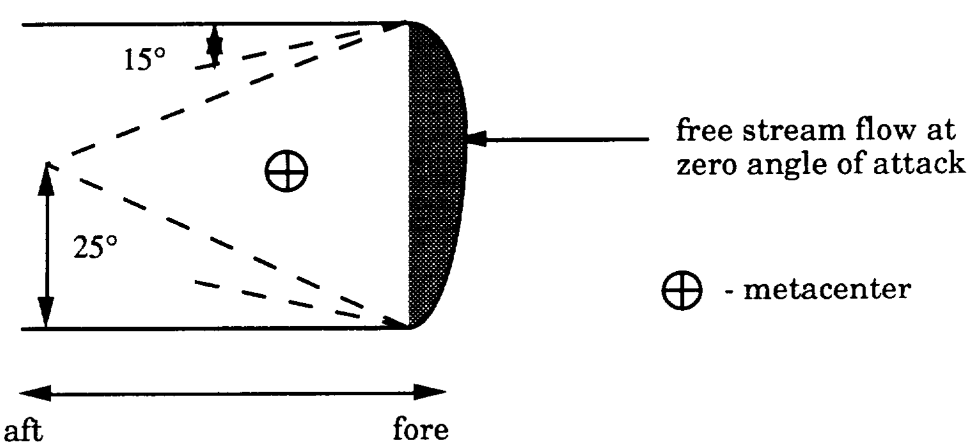


Figure 2.11: Impingement Cone

In order to maintain aerodynamic stability and control, the overall center of mass of the vehicle must remain fore of the metacenter. ("Metacenter" is a nautical term indicating the intersection of the flotation force vector with the plane of symmetry.) The metacenter for the spherical raked-cone design has been calculated to lie aft of the stagnation point by a distance equal to 60% of the diameter of the base of the brake. [ref. 2.3] Because of this restriction, it is critical that massive components of the vehicle be packed as close to the face of the brake as possible.

2.3.1 Configurations Considered

Four general configurations were considered for *Argo*. Each of these has characteristic strengths and weaknesses and meets the two primary restrictions outlined above.

Thrust Through Aerobrake

The prime advantage of this design, shown in Figure 2.12, is the fact that any payload can be situated such that the line of thrust passes through the center of mass (c.m.), regardless of the amount of payload. Adding payload will move the c.m. away from the stagnation point, but the c.m. of the payload can be placed on the thrust line.

Having the engines buried inside the shield is beneficial because it moves the c.m. closer to the stagnation point, but it requires that the payload be placed farther behind the shield. Whether this results in a net advantage depends on the mission. Aft movement of the payload slightly decreases the usable volume within the impingement cone and introduces the remote possibility that a payload may exceed the capacity of this design but not that of another design with the same brake size.

The engines must be retracted behind the shield for the atmospheric segment of the mission. This requires that the shield have a door in it, and that it be capable of opening and closing in flight with 100% success and no leakage. The machinery required to do this would add significant mass, the capability of firing the main engines while in the atmosphere is lost, and *any* failure of the door mechanism or seals would be catastrophic.

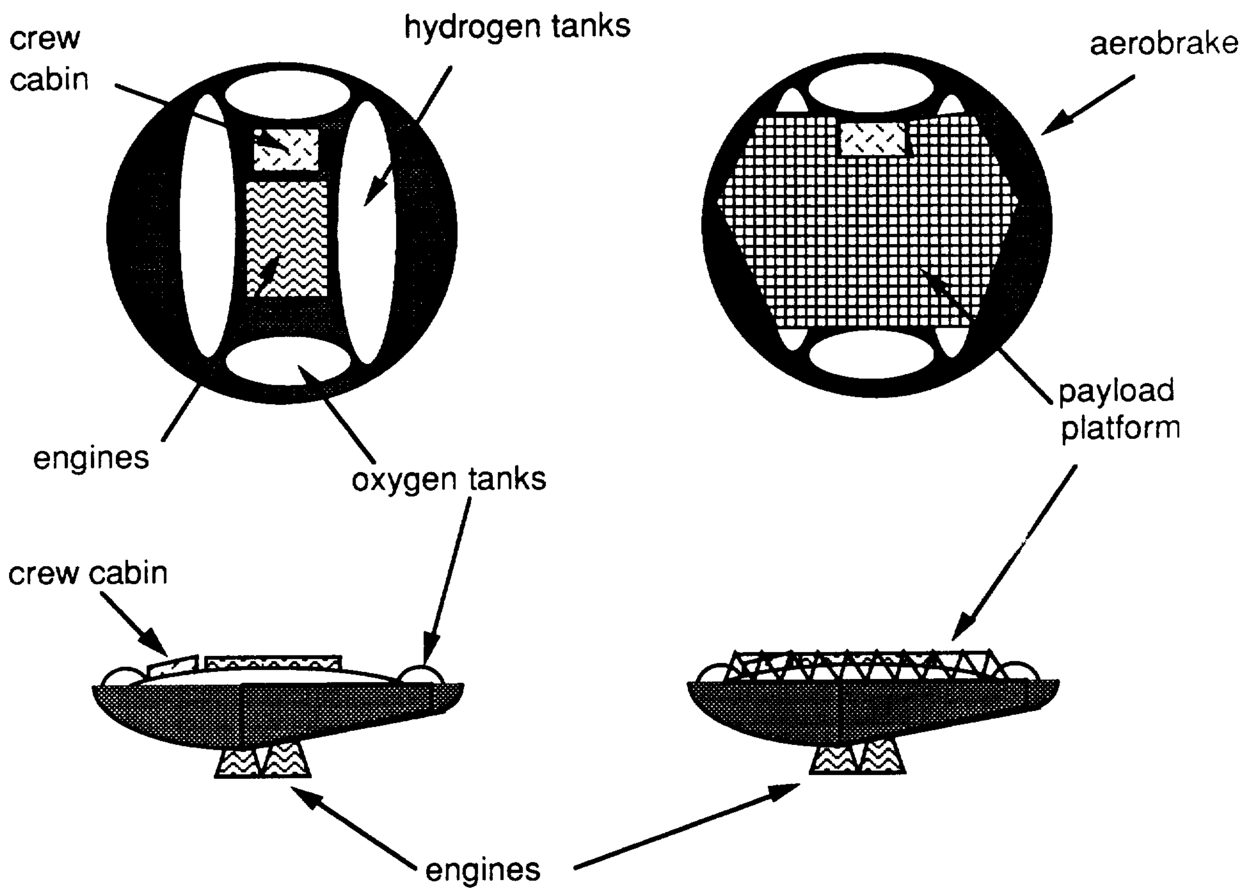


Figure 2.12: Thrust through aerobrake

Thrust Away From Aerobrake

This configuration, shown in Figure 2.13, has the engines oriented so that they fire behind the brake, exactly opposite the firing direction of the last configuration. One option is to place the engines near the aerobrake, but this causes the plume to render unusable much of the volume contained in the impingement cone. The other possibility is to place the engines near the apex of the impingement cone. This would require a fairly massive support structure to transmit the thrust forces to the rest of the vehicle. The center of mass would be moved considerably aft of the stagnation point, and overall mass would increase. This support structure would also place unnecessary constraints on the payload shape and volume, make loading and unloading payloads inconvenient, and make docking difficult if not impossible.

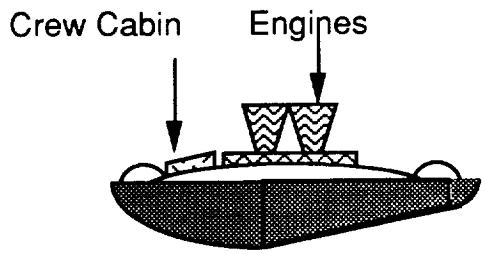


Figure 2.13: Thrust away from aerobrake

Payload Bay

In this design, shown in Figure 2.14, all of *Argo*'s largest components are packed as efficiently as possible around the circumference of the shield, leaving an opening in the center to be used as the payload bay. Moving the tanks away from the center of the cone moves them aft slightly, due to the curvature of the aerobrake. This shift of the c.m. is counteracted, however, by placing the payload closer to the shield than it is in other designs. The cargo could extend out of the payload bay, but it must be anchored within the bay, as only the bay is designed to withstand the stress of supporting the payload in atmospheric flight.

The payload bay configuration has a mass savings due to the elimination of the large payload platform, though some platform is required between the brake and the payload bay. The relatively small payload bay may require that the aerobrake diameter be increased to accommodate reasonable payloads, resulting in an overall increase in mass.

For any size aerobrake, a large portion of the volume inside the impingement cone is wasted in this configuration, due to the strictly defined payload bay. This is considered a major disadvantage.

2.3.2 Configuration Selected

The fourth design considered has been selected as the final design, because it maximizes payload capacity by minimizing restrictions. It is pictured in detail in Figure 2.1 and also shown in Figure 2.15. The crew cabin is placed between and above the hydrogen tanks, but this is the only vehicle component interfering with the payload's unobstructed use of the volume within the impingement cone. A payload platform covers the entire usable payload area.

There is a mass sacrifice in covering the entire usable area of the shield's base with a payload platform, but we have concluded that it is necessary in order to make full use of *Argo*'s capabilities. The mission profile is defined as being capable of transporting up to three satellites to

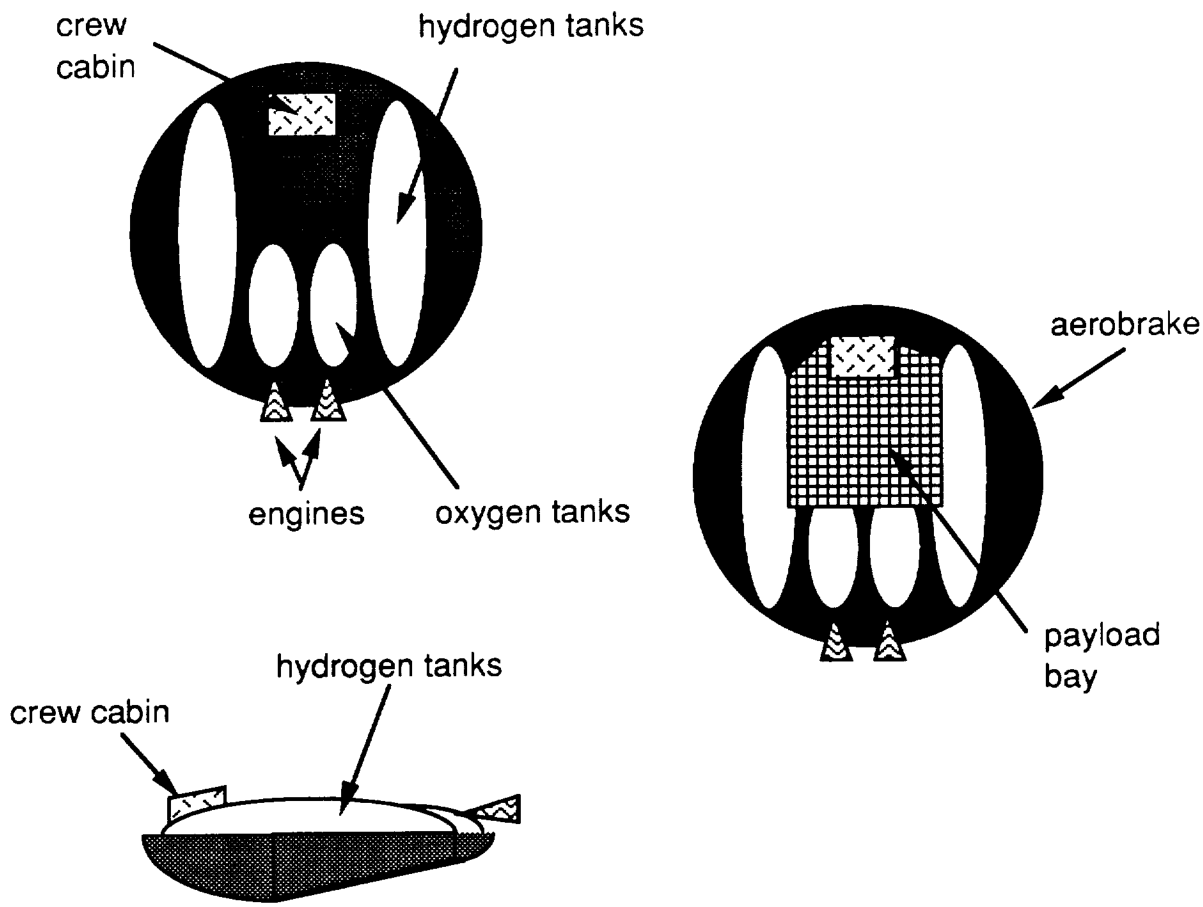


Figure 2.14: Payload Bay

GEO. Any area designated as a payload bay must be reinforced to support the payload, and this area must be maximized in order to accommodate the widest range of payload possibilities. The platform mass, estimated at 300 kilograms, is considered a part of the payload, not the vehicle itself.

The platform will be a mesh structure with supporting trusswork. The mesh allows heat from the radiators beneath to diffuse rapidly. The platform will be constructed of a lightweight, high strength composite of low thermal conductivity. It is designed such that portions can be removed in order to gain access to the tanks beneath. To save mass on missions which do not utilize the entire platform area, missions can be completed with platform sections removed.

The fact that the c.m. of the payload is moved aft by placing the cargo on a platform on top of the tanks does not introduce a problem to the metacenter constraint. Having complete flexibility in the manner in which the payload can be oriented, it is impossible for any realistic 10,000 kg payload to shift the c.m. aft of the metacenter. Thus, aerodynamic stability and control can be maintained for any payload down. Another critical problem is introduced, however, due to the relatively wide variation in the vehicle c.m. due to varying payload and fuel load. This will be discussed in detail in Section 2.3.3.

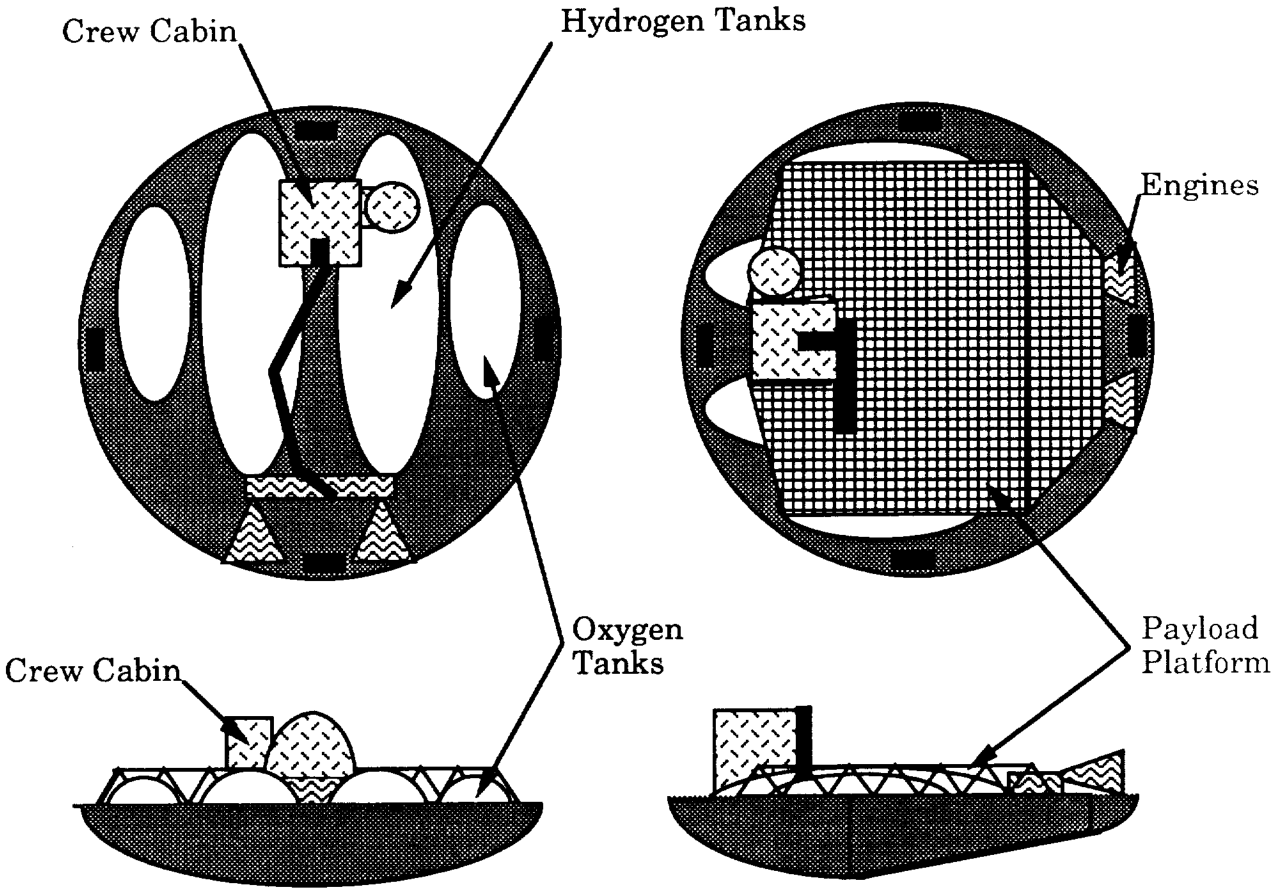


Figure 2.15: Design Selected

The position of the crew cabin partially above the payload platform has its own advantages. The crew is provided with a window through which they can directly observe the payload, extravehicular activity (EVA), and the remote manipulator system (RMS) arm. An astronaut who exits the EVA hatch arrives directly on the payload platform, and convenient handholds are on the exposed sides of the crew cabin. The platform itself is equipped with footholds and provides a uniform surface upon which the astronauts can maneuver. The raised crew cabin provides a convenient location for the communications antennae, minimizing the length required for the wires connecting the antennae to other communications equipment. The antennae are placed on swinging booms, so that they can extend over the lip of the shield to enable communications at all times in the space flight segment of the mission. The fact that the crew cabin is placed as close to the edge of the shield as possible allows the length (and mass) of the boom to be minimized. The RMS arm is provided a convenient location to be anchored: the aft, inside edge of the cabin. This allows easier use of the RMS by the astronauts, as they can observe the arm directly, and its shoulder is at approximately the same location and orientation as the astronaut's own shoulder. The position of the RMS also decreases the overall length (and mass) required of the arm: the arm need be only ten meters long to reach any point on the platform. This length could be shortened further by placing the arm shoulder at the center of the platform, but this would introduce unnecessary restrictions to the payload. With the shoulder on top of the cabin, the arm does not restrict payload, and it can be folded down and secured to the platform.

The engines can be retracted to lie within the impingement cone during the atmospheric segment of flight, and they can still fire, at a decreased efficiency, in the retracted position. The engines have gimballing capability to accommodate limited shifts in the center of mass of *Argo*. (See Section 2.3.3.)

Two radiators are placed on the outside of the hydrogen tanks, facing opposite each other. This will cause the heat to be expelled towards the payload, but the energy will not be sufficient to cause damage.

The RCS thrusters are at four points, symmetrically located at the edge of the brake. They lie outside the payload platform, and are not in a position to interfere with payload. This is covered in detail in Chapter 5.

The airlock, identical to that described in Section 2.2.2, is positioned beside the crew cabin, allowing for easy berthing and access to the payload.

2.3.3 Mass Analysis

For control of a space vehicle, the line of thrust must always pass through the center of mass of the vehicle. The RS-44 is capable of gimballing six degrees in any direction. This will easily accommodate lateral variations in the c.m. of the ASTV, as the vehicle is nearly symmetric and payloads can be loaded such that the c.m. moves very little in the lateral direction. As the engines are fired, the fuel will collect near the engines, shifting the c.m. toward the engines. While this will cause a sudden shift in the c.m., the effect will be slight and the engines can be quickly reoriented to compensate. The most drastic c.m. shifts will result from the addition of large payloads, especially when the tanks are low on fuel. The graphs in Figure 2.16 illustrate how vehicle c.m. is affected by payload. The payload c.m. location is defined as its distance aft of the payload platform. The vehicle c.m. location is measured aftward from the stagnation point on the fore face of the aerobrake.

With fuel tanks full and no payload, the c.m. is at its most fore point, 2.4 meters aft of the stagnation point and 9 meters from the engine-side edge of the brake. The engines are situated such that their line of thrust passes through this point when they are fully gimballed. (No efficiency is lost by gimballing the engines.) The engines can now gimbal 12 degrees from this position, allowing the c.m. to move aft by a distance equal to $9\sin 12°$, or 1.9 meters. This corresponds to 4.3 meters being the farthest the c.m. can lie aft of the stagnation point. From the graphs, it is clear that mission capability has been limited by the gimballing capacity of the engines. This result is inevitable when using a vehicle configuration in which the engines do not fire approximately normal to the base of the aerobrake. (Reasons for the elimination of these designs are found in Section 2.3.1.)

This problem can be replied to in two ways. First, we can request that Rocketdyne modify the RS-44 to increase its gimballing capability. If the RS-44 could gimbal approximately fifteen degrees, this would allow the ASTV to accommodate any payload that can fit within the impingement cone. If this modification is not possible, we can examine Figure 2.16 to point out that very few missions have actually been eliminated. Taking payload up to GEO, *Argo* can handle any except the most extreme shape of a 10,000 kg payload. Very few payloads would need to be situated such that its own c.m. must be seven meters aft of the platform. Most payloads could be laid down, lowering the c.m. greatly. For example, a ten meter long, five meter diameter cylindrical payload weighing 10,000 kg could be easily managed if it were lying down. Looking at the payload down, *Argo* has no problem with any 1000 kg payload. Even a 10,000 kg payload can be transported if it can be situated in a way that locates its c.m. within 0.8 meters of the platform.

2.3.4 Possible Modifications for Future Missions

For simple, hazardous, or extended missions, the crew module can be replaced by an unmanned command module for autonomous missions. This is the only major modification which can be made to our basic configuration. Fuel tank volume is limited by aerobrake diameter, so we have chosen to integrate the remainder of the vehicle. This reduces possibilities for future expansion.

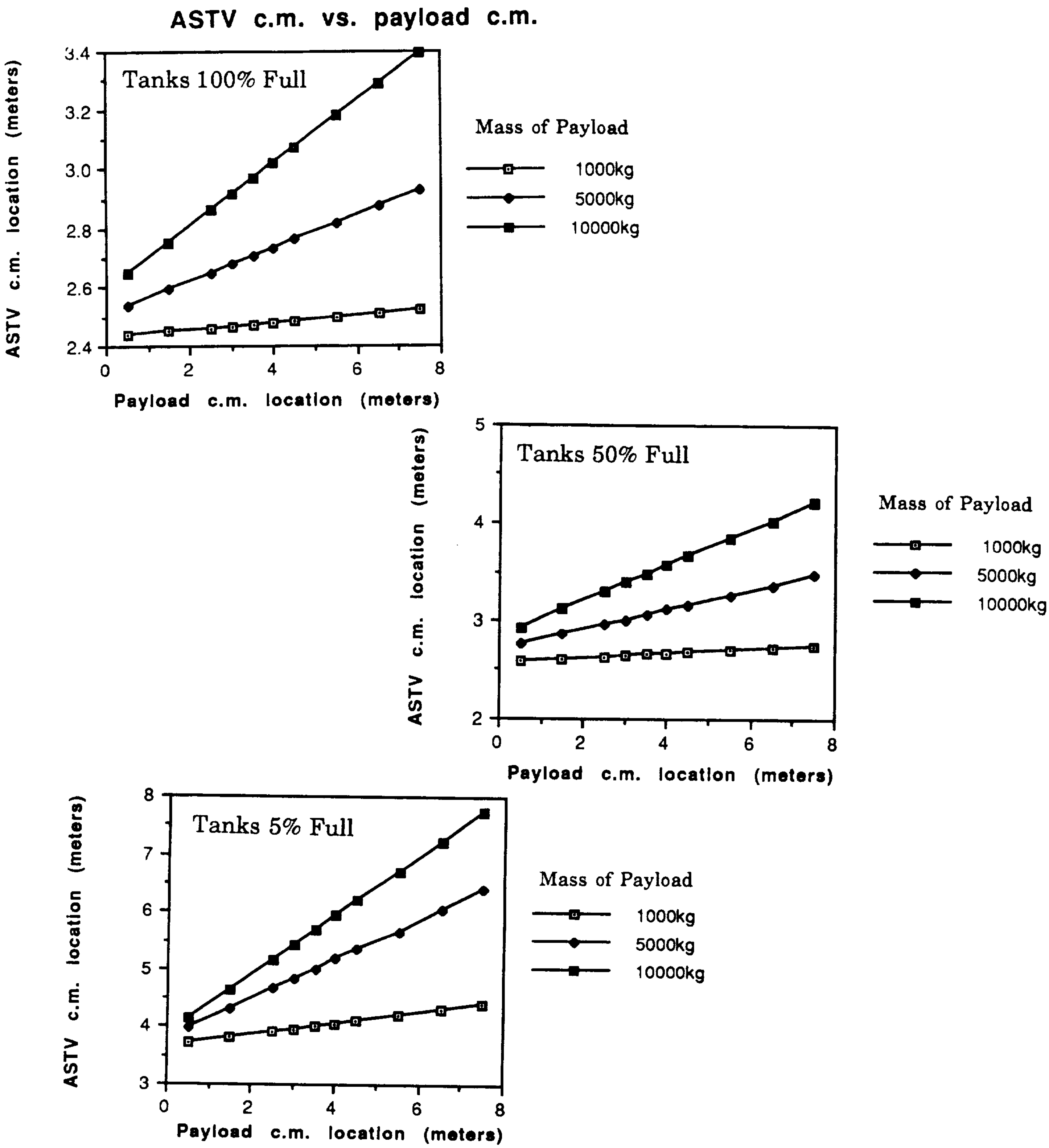


Figure 2.16: Mass Analysis

2.4 Mass Breakdown

All masses are in kilograms. Propellant masses are for the nominal mission of 10,000 kg LEO to GEO, 5000 kg GEO to LEO.

	CSTV	ASTV
GROSS MASS	**86566**	**66321**
DRY MASS	**6341**	**7841**
Main Propellants	**77925**	**56280**
LOX	60115	42590
LH2	10019	7098
Residuals and Reserves	7791	6592
RCS Propellants	**2300**	**2300**
LOX	1540	1540
LH2	515	515
Residuals and Reserves	245	245
Propulsion Hardware	**1555**	**1325**
H2 Tanks (2)	586	402
Engines (2)	370	370
O2 Tanks (2)	308	262
Plumbing	100	100
Valves	16	16
RCS System	175	175
Structures	**1180**	**1180**
Equipment Mounting	245	245
Basic Body	180	180
Micrometeroid Shield	169	169
Crew module Interior	100	100
Crew module Mounting	100	100
Auxiliary Tank	100	100
Thrust Structure	100	100
Umbilical Panels	50	50
Payload Mounting	29	29
Launch Scar	18	18
Contingency	89	89
Power Generation	**373**	**373**
Fuel Cells (3)	285	285
Reserve LOX	62	62
Reserve LO2	8	8
Distribution	18	18
Thermal Control	**130**	**130**
Radiator	100	100
Internal Cooling	30	30
Communications	**55**	**55**
Antennae (2)	40	40
Transciever	15	15
Data Management	**76**	**76**
NIU (6)	30	30

Workstations	30	30
Storage Disks	10	10
Processors (6)	6	6
Avionics/Guidance	**54**	**54**
IMU (3)	45	45
Star Trackers (3)	9	9
Aerobrake	-	**1630**
Tiles	-	650
Structure/Stringers	-	400
G Polymide Skin	-	300
Honeycomb Base	-	100
Silicon Adhesive	-	100
RCG Coating	-	70
Strain Pad	-	10
Atmosphere Control	**196**	**196**
Nitrogen Gas and Tanks	78	78
Emergency Pressurization	40	40
LiOH Cartridges	26	26
Oxygen	22	22
Heat Exchange/Water Sep.	10	10
Atmosphere	10	10
Fans	6	6
Pressure Regulators	3	3
Odor Control	1	1
Crew Systems	**363**	**363**
Crew	160	160
Commode	75	75
Seating	40	40
Water Pump Packs (3)	30	30
Food and Water	28	28
Galley	10	10
Interior Lights	10	10
Health Maintenance	5	5
Water Plumbing/Storage	3	3
Hand Wash	2	2
External Operations	**1235**	**1235**
EVA Suits (2)	364	364
RMS	350	350
Airlock	291	291
MMU	160	160
MMU Servicer	60	60
Tools	10	10
Radiation Shielding	**1125**	**1225**
Crew Compartment	550	650
Storm Shelter	575	575

2.5 References

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2.3 Park, Chul, "A Survey of Aerobraking Orbital Transfer Vehicle Design Concepts," NASA Ames Research Center.

Chapter 3

Mission Analysis

3.0 Summary

The function of the Mission Analysis group is to determine the orbital trajectories and maneuvers which best satisfy the nominal all-propulsive and aeroassisted missions. Accomplishing this task consists of investigating the orbital mechanics of the various possible transfers and maneuvers.

This chapter is different from all other chapters in that in most cases, the results are given in the form of equations, rather than hard, concrete numbers. Our results must be portrayed in this fashion because they change mission by mission. We felt that in a more general form, our results would be much more useful to potential mission planners.

In order to select the maneuvers for the nominal mission, we considered the worst case. For example, we calculated maximum ΔV's and maximum transfer times for each possibility. Then we compared each possibility with the others and selected the best one. After this analysis, it was possible to find a maximum ΔV required for both the all-propulsive and aeroassisted STV. Figure 3.1 summarizes the maximum ΔV which must be provided by the main engines of the STV. Note that the aeroassisted requirement is substantially lower than the all-propulsive requirement. The difference arises because a major portion of the insertion into LEO is made with the aerobraking maneuver which makes no use of the main engines. The values in Figure 3.1 were calculated assuming that the Space Station was at a 278 km altitude, the lower limit of LEO, and that Geosynchronous orbit (GEO) was at a 35,789 km altitude, neglecting the oblateness of the Earth.

	ΔV Description	All-Propulsive	Aeroassisted
ΔV_1	Initiation of LEO to GEO Transfer	2470 m/s	2470 m/s
ΔV_2	Insertion into GEO	2019 m/s	2019 m/s
---	Operations in GEO	602 m/s	602 m/s
ΔV_3	Initiation of GEO to LEO Transfer	1835 m/s	1868 m/s
ΔV_4	Insertion into LEO, Phasing	2438 m/s	364 m/s
---	Rendezvous	31 m/s	31 m/s
TOTAL		9395 m/s	7354 m/s

Figure 3.1: Maximum Propulsive ΔV Requirements of Nominal Mission

In Section 3.2, we describe missions beyond the nominal mission which the STV design is capable of. We also discuss mission enhancements using the Orbital Maneuvering Vehicle.

In Section 3.3, we describe the Three Impulse Transfer and give the reasoning behind discarding it as an option in the nominal mission.

In Section 3.4, we describe the sources of error which effect the orbital maneuvers we describe. We also describe how we incorporated them in our calculations, if at all.

Finally, in Section 3.5, we describe a sample mission flight plan.

3.1 The Nominal Mission

The nominal mission calls for the ability of the STV to carry a payload from SSF (in LEO inclined 28.5° with respect to the equator) into GEO, maneuver to other positions in GEO for repair, retrieval and/or placement, and return a payload from GEO to Space Station *Freedom*. The nominal mission is divided into the following phases:

1) Separation maneuver from SSF to departure point
2) Transfer from LEO to GEO
3) Operations in GEO
4) Transfer from GEO to LEO
5) Aerobrake maneuver (ASTV only)
6) Phasing, Rendezvous, and Docking with SSF

In the following sections, we will describe the maneuvers necessary for each phase of the nominal mission.

3.1.1 Separation from SSF

According to Space Station *Freedom* specifications, any orbital transfer burns must take place at least 18 km from the Space Station [ref. 2]. Therefore, the STV must somehow leave the vicinity of the Space Station before initiating any orbital transfer maneuver. To do this, the STV will undock from SSF (undocking is described in Chapter 8) and then apply a small "low-z" ΔV of about 0.1 m/s. The "low-z" burn is a short burst of the attitude thrusters vectored in such a way that there is a very small component of thrust in the direction of the Space Station. In this way, most of the exhaust gasses will be vectored away from the Space Station.

The STV will maintain this separation rate for about 15 minutes. Then, the STV will initiate a 1.1 m/s burn which will allow it to "orbit walk" (see Section 3.1.4) in one LEO revolution to the required 18 km distance. Once at this point, the STV may initiate the transfer maneuver.

3.1.2 LEO to GEO Transfer

The best transfer method for getting from LEO to GEO is the Direct Elliptical Transfer (DET). The DET is a generalized form of the Hohmann Transfer (described in Section 3.1.4). Both follow elliptical trajectories between LEO and GEO. However, while the Hohmann transfer covers exactly 180° of travel, the DET can vary the angle traveled, allowing a greater flexibility in flight time.

In the Direct Elliptical Transfer, ΔV_1 will place the STV in an elliptical orbit which intersects GEO at the hinge line. This ΔV is applied just before or just after the hinge line, depending on the point in GEO which is targeted. Once at GEO, the STV must apply ΔV_2 (See Figure 3.2). This ΔV will:

1) Circularize the STV from the elliptical transfer orbit to GEO
2) Give it 28.5° of plane change (reason for arrival at hinge line)
3) Correct the flight path angle.

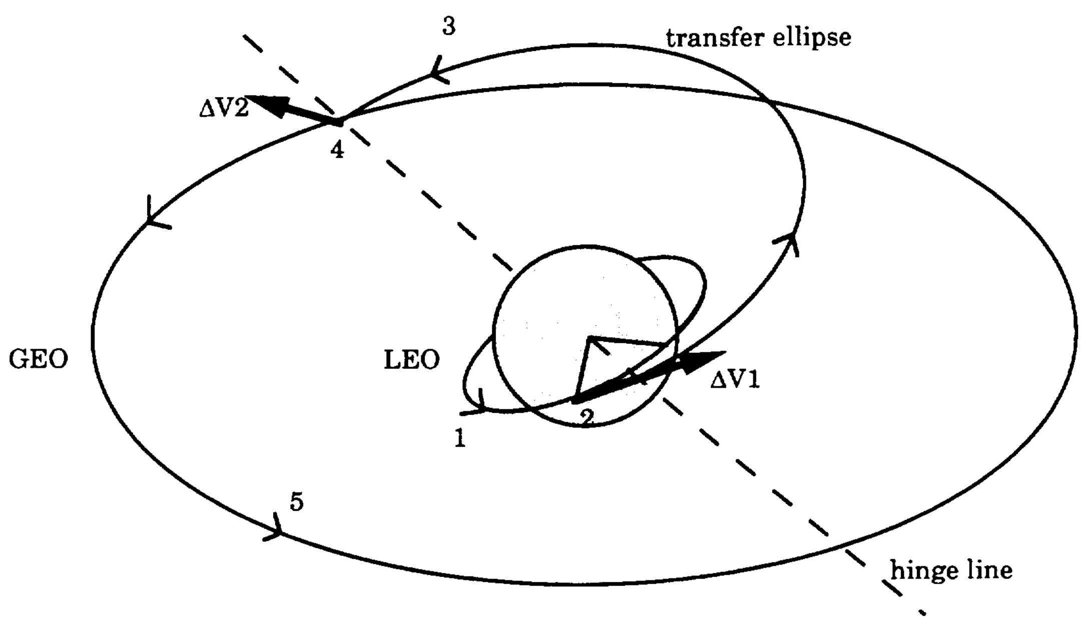


Figure 3.2: The Direct Elliptical Transfer

The orbital mechanics for this maneuver are complex compared to other transfer methods. However, it can be shown that the ΔV's are given by

$$\Delta V_1 = \sqrt{\frac{\mu}{R_{leo}}} \left\{ [1+e]^{1/2} - 1 \right\} \tag{3.1}$$

$$\Delta V_2 = \sqrt{\frac{\mu}{R_{leo}}} \sqrt{\frac{2-e}{n} - \frac{2}{n}(1-e)^{1/2} \cos\gamma \sin\alpha} \tag{3.2}$$

where γ is defined as the flight path angle of the STV as it crosses the hinge line at GEO, $\alpha = 28.5°$ is the amount of plane change, and e is the eccentricity of the transfer orbit.

The DET transfer time from LEO to GEO can be calculated using

$$t = \frac{\tau}{2\pi}(\eta - e \sin \eta) \tag{3.3}$$

where η is the eccentric anomaly and τ is the period of the transfer ellipse as given in Section 3.1.4. The eccentric anomaly is given by

$$\eta = 2 \tan^{-1}\left\{ \sqrt{\frac{1-e}{1+e}} \tan\frac{\theta}{2} \right\} \tag{3.4}$$

In Equation 3.4, θ is the angular sweep of the transfer and is a function of the eccentricity, e. Because all other factors are constant depending on the eccentricity, the total transfer time depends only on the eccentricity.

The solution of the DET for a specific target is an iterative process dependant on the selection of an arbitrary independent variable. We chose e as the independent variable because the total ΔV is of the form

$$\Delta V = A + Be^{1/2} + Ce^{1/4}$$

a relatively simple relationship. Other choices of an independent variable led to less predictable forms. The correct choice of the eccentricity depends on the position of the target in GEO. Determining e for a specific target trajectory is an iterative process which must consider both the time of flight and the angle swept by the trajectory from LEO to GEO. The analysis of this iterative process is beyond the scope of this report, but it involves computerized iterations of the formulae given in this section.

The maximum ΔV required for the DET represents sufficient flexibility in transfer time to allow two flight windows per day at a ΔV increase of only 5.3% over the Hohmann Transfer. The maximum ΔV requirement was determined by defining an angular region at GEO relative to the hinge line through which the target must traverse. This region was defined so that the target would take more time to traverse it than the orbital period of the STV at LEO. The STV could therefore be at a predesigned starting point in LEO while the target was somewhere in the GEO angular region. The maximum ΔV gives the STV sufficient flexibility to target *any* point within the GEO angular region.

There are two angular regions at GEO, each relative to a different half of the hinge line. An orbiting target in GEO will pass through each region once a day, thus allowing two targeting opportunities by the STV each day. These two flight windows can be guaranteed for a total ΔV of 4489 m/sec (one way). The maximum transfer time required is 7.21 hours.

Because this transfer method involves only two ΔV's, only two engine start-ups are needed per one-way trip.

3.1.3 Operations in GEO

The next phase of the nominal mission consists of operations in GEO. In GEO, we must have the capability to target more than one location for delivery, repair or service of satellites and spacecraft. The STV will traverse the distance between these points using a maneuver known as *orbit walking*. Orbit walking consists of performing either a posigrade or retrograde ΔV while in a circular orbit like GEO. This ΔV changes the vehicle's orbit from circular to slightly elliptical. A posigrade burn will raise the apogee of the orbit slightly while a retrograde burn will lower the perigee of the orbit slightly.

Changing to a slightly elliptical orbit also gives a corresponding change in orbital period. A posigrade ΔV gives a longer period; a retrograde burn a shorter one. For example, suppose a retrograde ΔV is given to the STV. The perigee will be lowered and the new elliptical orbit will have shorter period. The STV will make one revolution around Earth and return to its apogee at GEO. The vehicle arrives ahead of the point in GEO where it began the maneuver. In this manner, the STV will continue "walking in steps" ahead in GEO until a posigrade ΔV of magnitude equal to the original retrograde ΔV is applied at GEO altitude. The angular range, θ, covered in the walk is found using the formula [ref. 1]

$$\theta = \frac{6\, n\, \pi\, \Delta V}{\omega\, R_{geo}} \qquad (3.5)$$

In Equation 3.5, n is the number of orbit walking "steps", ΔV is the amount of applied velocity change, R_{geo} is the radius of GEO, and ω is the angular rate of GEO. Each "step" takes about 24 hours.

For the nominal mission, we required the capability of targeting three positions in GEO. The first position is targeted using the Direct Elliptical Transfer from LEO. The second and third positions will be reached using orbit walking. In the nominal mission definition, it is required that the STV be able to orbit walk 180° in three days. Referring to Figure 3.3, we see that such a maneuver requires 171 m/s to initiate. The total ΔV for the maneuver is twice this amount, or 343 m/s. Figure 3.3 also shows that if the target is close enough, it may be reached in less than three days with the same amount of total ΔV.

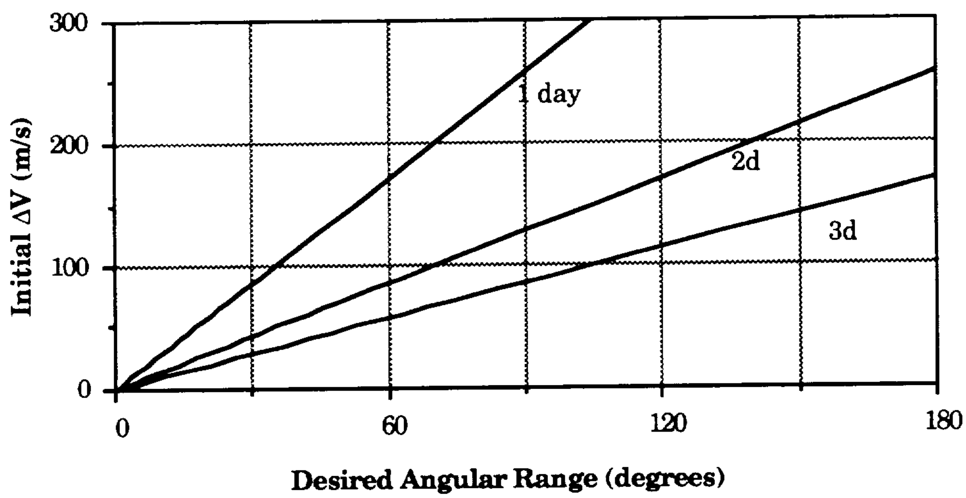


Figure 3.3: Necessary ΔV For Initiating GEO Orbit Walk

3.1.4 Maneuvering From GEO to LEO

After completing mission operations in GEO, the STV will return to a point 15 km below the Space Station. The STV won't return directly to SSF altitude because it must complete phasing and rendezvous. These aspects are described in Sections 3.1.4 and 3.1.5.

The transfer method for this phase will be a Hohmann Transfer which consists of two engine burns. The first burn, ΔV_3, will be applied at the hinge line in GEO and will place the STV in a minimum energy transfer ellipse with apogee equal to GEO radius and perigee equal to the radius of the space station minus 15 km. It will traverse 180°, bringing the STV to the other end of the hinge line. Here, the second burn, ΔV_4, will circularize the STV (Refer to Figure 3.4). In practice, ΔV_4 will not be applied all at once because of phasing considerations.

In the Hohmann Transfer, the plane change may be effected in a combination plane change - trajectory change or *dog-leg* maneuver. The transfer may consist of one dog-leg maneuver at GEO incorporating all the plane change, one dog-leg maneuver at LEO incorporating all the plane

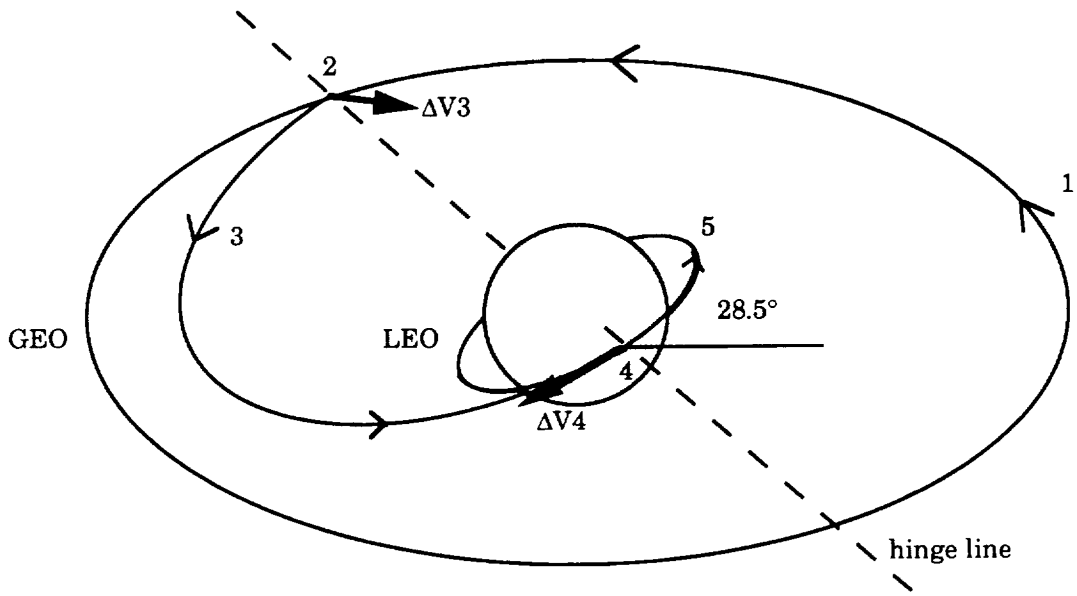


Figure 3.4: The Hohmann Transfer

change, or it may consist of two dog-leg maneuvers which divide the total plane change. We found that the total ΔV required for the maneuver is minimized if the plane change is done in two steps. We defined the amount of plane change done at LEO as α and the amount of plane change at GEO as β. Figure 3.5 shows the total ΔV required for the transfer versus the amount of plane change at LEO.

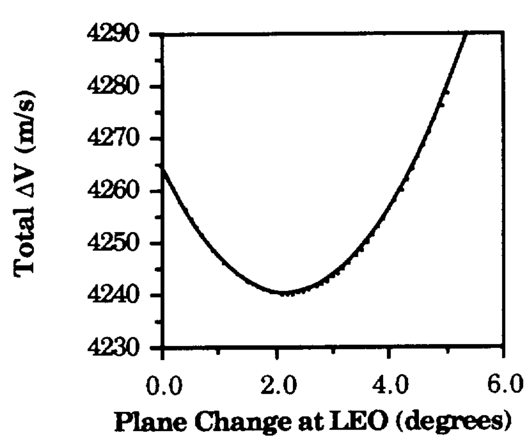


Figure 3.5: Hohmann Transfer With Some Plane Change at LEO

Figure 3.5 indicates that if α=2.2° and β=26.3°, then there is a 25 m/s savings in the total ΔV required to complete the transfer than if all the plane change were made at GEO. It is also evident from the curve that it is prohibitively expensive in ΔV to make all of the plane change at LEO.

The ΔV required for the first burn is given by [ref. 1]

$$\Delta V_3 = \sqrt{\frac{\mu}{R_p}} \sqrt{\frac{2n}{n+1} + 1 - 2\sqrt{\frac{2n}{n+1}} \cos\alpha} \tag{3.6}$$

The second ΔV is given by [ref. 1]

$$\Delta V_4 = \sqrt{\frac{\mu}{R_p}} \sqrt{\frac{2}{n(n+1)} + \frac{1}{n} - 2\sqrt{\frac{2}{n^2(n+1)}} \cos\beta} \tag{3.7}$$

In Equations 3.6 and 3.7, n is the ratio of the radius of GEO and the perigee radius, R_p:

$$n = \frac{R_{geo}}{R_p} \tag{3.8}$$

The transfer time required for the Hohmann Transfer is one-half its orbital period. The period of an elliptical orbit depends only on the semi-major axis, a, which is defined as the average of the perigee and apogee. The period is given by

$$\tau = \frac{2\pi}{\sqrt{\mu}} a^{3/2} \tag{3.9}$$

where μ is the gravitational constant of Earth. From this equation we determined that the total transfer time is 5.27 hours.

All-Propulsive Hohmann Transfer

In the all-propulsive case, the STV will use the optimum plane change angles as given by Figure 3.5. They are α=2.2° and β=26.3°. If we assume that on the return trip the STV comes down to 278 km altitude from GEO at 35787 km altitude, then the total ΔV for the transfer is 4240 m/s.

Aeroassisted Hohmann Transfer

In the aeroassisted case, the aerobrake cannot provide any plane change. Therefore, to compute the required ΔV for a transfer from 35787 km to 278 km altitude, we must use α=0.0° and β=28.5°. The total ΔV for the aeroassisted Hohmann Transfer is 4265 m/s.

3.1.5 Phasing

Upon completion of GEO operations, the STV must return from GEO and rendezvous with the Space Station. For rendezvous, a target point typically around 15 km below and 15 km behind the Space Station is selected. From this point the Terminal Phase Initiation can occur. The Terminal Phase culminates in docking of the STV with the Space Station.

For return to LEO, a Hohmann Transfer is used, minimizing the ΔV required. An engine burn at the hinge line in GEO, ΔV_3, reduces the spacecraft's velocity enough to provide a perigee in LEO. However, this will not in general result in a rendezvous opportunity, as the STV will be arriving in LEO while the target point is at a random position in its circular orbit. So instead of performing the circularization burn, ΔV_4, upon reaching perigee, the STV is inserted into a phasing orbit which will allow it to arrive back at perigee at the same time as the target point.

When the STV reaches the perigee of the Hohmann Transfer, the target point will be at a known angular separation θ from the STV. For rendezvous, it is desired for the CSTV to make one circuit of the phasing orbit (one or more circuits for the ASTV) and return to perigee in the same time interval for the target point to complete its current orbit (that is, cover $360° - \theta$) plus one additional orbit. A phasing orbit is chosen with perigee R_p equal to the target point's radius, and an apogee R_a sufficient to give the desired orbital period. Figure 3.6 shows the phasing and rendezvous method.

All-Propulsive Phasing

Upon reaching LEO, the STV performs a retrograde burn (known as $\Delta V_{4'}$) which will provide the desired amount of plane change $\alpha=2.2°$ and reduce its apogee from GEO radius to the desired apogee, R_a, which will never be greater than 8562 km above the Earth's surface. $\Delta V_{4'}$ is given by

$$\Delta V_{4'} = \sqrt{\frac{\mu}{R_p}} \sqrt{4 - \frac{4 + 2n_1 + 2n_2}{(n_1 + 1)(n_2 + 1)} - 2\sqrt{4 - \frac{4 + 4n_1 + 4n_2}{(n_1 + 1)(n_2 + 1)}} \cos\alpha} \tag{3.10}$$

In Equation 3.10, $n_1 = R_{geo}/R_p$ and $n_2 = R_a/R_p$. The STV will now complete one orbit while the target completes one and a fraction, bringing both to the same point at perigee. Here, the STV performs another retrograde burn (known as $\Delta V_{4''}$) to circularize its orbit at this altitude. It then waits for the proper conditions for Terminal Phase Initiation. $\Delta V_{4''}$ is given by

$$\Delta V_{4''} = \sqrt{\frac{\mu}{R_p}} \left[1 - \sqrt{2 - \frac{2R_p}{R_p + R_a}} \right] \tag{3.11}$$

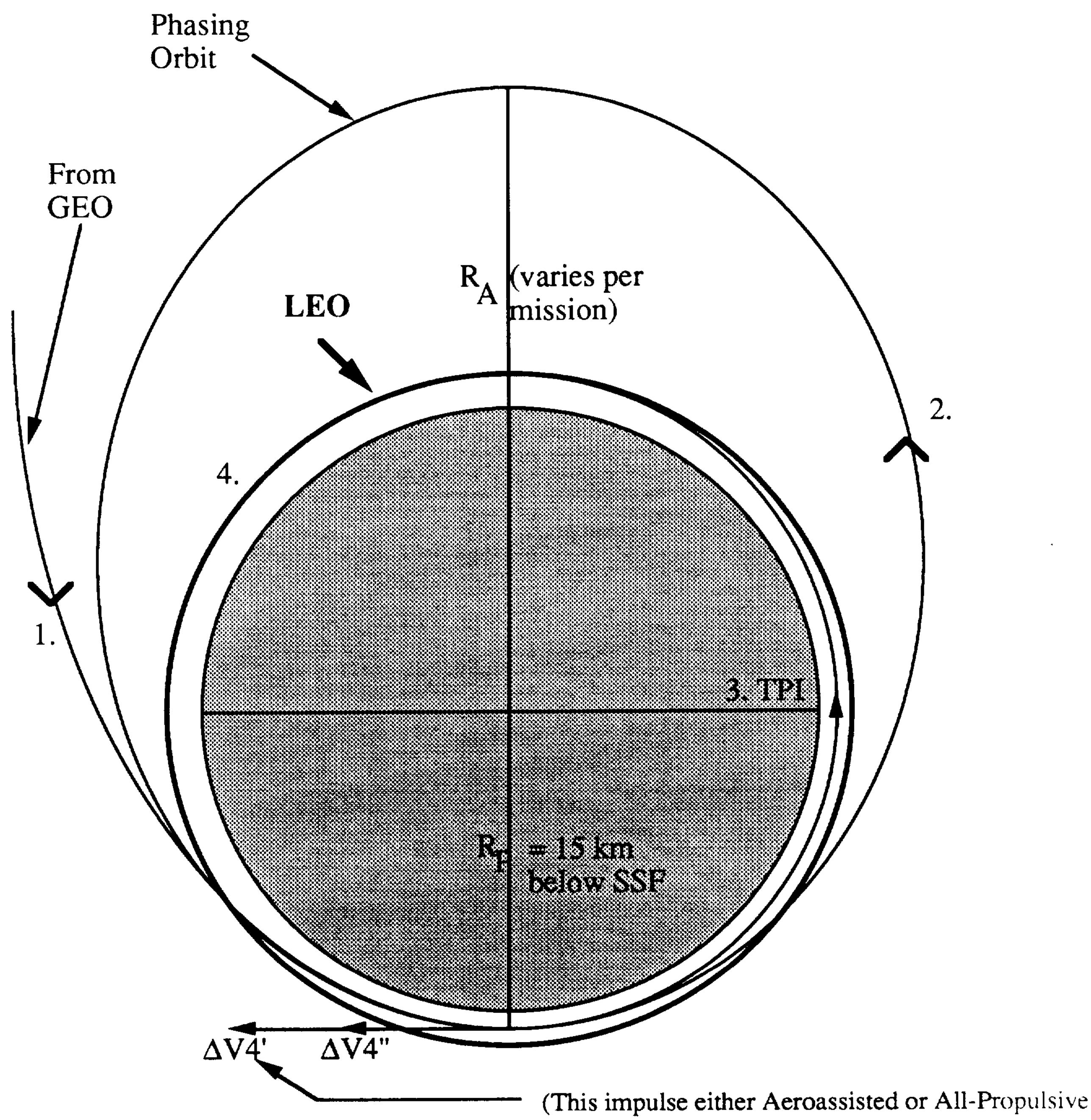


Figure 3.6: Phasing Maneuver

The sum of the ΔV's applied during these two burns is the same as the ΔV_4 for the Hohmann transfer, so no extra fuel is required for this rendezvous method.

Aeroassisted Phasing

In the case of the aeroassisted vehicle $\Delta V_{4'}$ is provided by Earth's atmosphere. The new ASTV orbit has a Theoretical Vacuum Perigee, R_{po}, which is located within the atmosphere. If the orbit is not modified, the ASTV will re-enter the atmosphere. A small correction, $\Delta V_{4''}$, applied at first apogee (never more than 1422 km above Earth's surface), must be completed to raise the perigee out of the atmosphere to the desired value R_p, 15 km below SSF. $\Delta V_{4''}$ is given by

$$\Delta V_{4''} = \sqrt{\frac{\mu}{R_p}}\left[\sqrt{2 - \frac{2R_a}{R_a + R_p}} - \sqrt{2 - \frac{2R_a}{R_a + R_{po}}}\right] \tag{3.12}$$

The ASTV then makes n circuits of this new orbit until rendezvous with the target point. The greater the number of revolutions, the less ΔV and therefore fuel is needed when the circularization burn is completed. However, increasing n also increases the phasing time, so a trade-off exists.

After completing n circuits of the phasing orbit, $\Delta V_{4'''}$ is applied to circularize the orbit. It is given by

$$\Delta V_{4'''} = \sqrt{\frac{\mu}{R_p}}\left[1 - \sqrt{2 - \frac{2R_p}{R_p + R_a}}\right] \tag{3.11}$$

It should be noted that both $\Delta V_{4''}$ and $\Delta V_{4'''}$ are made propulsively, increasing the fuel cost. If phasing with SSF were not necessary, these impulses could have been made in the aerobrake maneuver. Thus, the aerobrake is not used to its full potential.

3.1.6 Rendezvous

Terminal Phase Insertion

From the target point below and behind the Space Station, rendezvous with the Space Station can be achieved in a predetermined time T with two applied ΔV's. A point in the orbit of the Space Station will be selected for rendezvous based upon lighting and other considerations. From this information the desired time until rendezvous T will be selected. The desired angular range, ωT, can then be found. The angular rate ω is the angular rate of a space craft in LEO.

Once the angular range has been determined, the two required ΔV's can be calculated. The initial ΔV changes the approach rate of the STV relative to SSF such that, after a time t=T has elapsed, the position of the

STV will coincide with that of the Space Station. It is computed by Equation 3.14:

$$\overrightarrow{\Delta V_i} = \left[\dot{x}_d(0) - \dot{x}(0)\right] e_x + \left[\dot{y}_d(0) - \dot{y}(0)\right] e_y \tag{3.14}$$

In Equation 3.14, x-dot and y-dot are the relative approach rates as a function of time of the STV in the horizontal and vertical directions respectively. The subscript d denotes the desired relative approach rates which will ensure rendezvous with SSF. These are given by [ref. 1]

$$\dot{x}_d(0) = \frac{14\, y(0)\, [1 - \cos \omega T] - [\, 6\, y(0)\, \omega T - x(0)\,] \sin \omega T}{T\, [\, 3 \sin \omega T - \frac{8}{\omega T}(1 - \cos \omega T)\,]} \tag{3.15}$$

$$\dot{y}_d(0) = \frac{-y(0)\, [\, 3\omega T \cos \omega T - 4 \sin \omega T] - 2\, x(0)\, [1 - \cos \omega T]}{T\, [\, 3 \sin \omega T - \frac{8}{\omega T}(1 - \cos \omega T)\,]} \tag{3.16}$$

x(0) and y(0) give the horizontal and vertical displacement of the STV relative to the Space Station at the application of the initial ΔV.

The braking ΔV is given by

$$\overrightarrow{\Delta V_b} = [\, - \dot{x}(T)\,]\, e_x + [\, - \dot{y}(T)\,]\, e_y \tag{3.17}$$

where x-dot (T) and y-dot (T) are given by:

$$\dot{x}(T) = [\, -3\, \dot{x}_d(0) - 6\omega\, y(0)\,] + [\, -2\dot{y}_d(0)] \sin \omega T + [\, 4\, \dot{x}_d(0) + 6\omega\, y(0)\,] \cos \omega T \tag{3.18}$$

$$\dot{y}(T) = [\, 2\dot{x}_d(0) + 3\omega\, y(0)] \sin \omega T + \dot{y}_d(0) \cos \omega T \tag{3.19}$$

Selection of the angular range can keep the ΔV's on the close order of 45 m/s for rendezvous times between 10 and 83 minutes. These rendezvous times correspond to angular ranges of 40° and 330°.

To prevent rocket exhaust impingement, the braking burn cannot be performed in the immediate vicinity of the Space Station. However, it can be close enough so that only a few extra m/s will be necessary for station keeping and final approach for docking. The braking burn will occur about 300 m ahead of the Space Station.

Close-in Proximity Operations

Rendezvous maneuvers will be considered completed when the STV is approximately 300 m from the Space Station. At this point, Proximity Operations begin. A small ΔV will be applied with a "low-z" burn which will establish a closing rate to bring the STV to the Space Station in a short period of time. For example, a ΔV of 1 m/s will close the 300 m gap in about 5 minutes. The STV will reduce its closing rate to zero by applying a reverse "low-z" burn in the opposite direction. ΔV's applied very close to the Space Station will be limited to a fraction of a meter per second so as not to disturb the environment around the Space Station.

3.2 Other Possible Missions with Nominal Mission Budget

In addition to the nominal mission described in the preceding section, the STV will be able to accomplish many other missions. In all of these missions, the ΔV required is below the maximum ΔV of the nominal mission.

3.2.1 Polar Orbits at GEO and LEO Altitudes

In addition to being able to target satellites in equatorial GEO, the STV will be able to service satellites in polar GEO orbits. Polar orbits are inclined 90° to the equatorial plane of Earth. Because the STV is in an orbit inclined at 28.5°, it must accomplish 61.5° of plane change to get to a polar orbit. This plane change could be accomplished by a dog-leg maneuver, just as with the Hohmann Transfer. However, since the amount of required plane change is so much greater, the required ΔV is much greater. The ΔV can be reduced if the vehicle utilizes a Three Impulse Transfer with a very high intermediate altitude (see section 3.3.1). There is a ΔV savings if the plane change is made when the vehicle's velocity is lowest.

We assumed a three day time limit on the Three Impulse Transfer and wrote a computer program which calculated the amount of ΔV necessary to place the STV into the polar plane at GEO altitude. With this mission profile we were able to achieve placement into polar GEO orbit with a minimum ΔV of 4688 m/s (one-way trip). This is roughly 450 m/s over a standard Hohmann Transfer to GEO.

Placement into lower altitude polar orbits, however, is more costly. To place our vehicle into polar LEO (278 km altitude) on the same three day flight profile, it would cost us a ΔV of 6427 m/s for a one way trip. This ΔV is so large that we would not be able to transport as large a payload, if any, given the size of the propellant tanks.

3.2.2 Orbits in Planes Other than 28.5°, Equatorial, or Polar

Because there are many satellites which are not in LEO, equatorial GEO, or polar orbits, it is desirable to determine what maximum inclination the STV could reach in a transfer from LEO to circular orbits of varying radius. We completed a simple analysis of these possible orbits by assuming a maximum allowable one-way ΔV of 4265 m/s. This is the ΔV it takes for a LEO to GEO transfer with 28.5° of plane change conducted entirely at GEO. In the analysis, we assumed that the entire plane change would be conducted at the higher altitude.

For each altitude between LEO and GEO, there is a maximum possible amount of plane change which may be made without exceeding 4265 m/s. Figure 3.7 shows the relationship between the radius of the circular orbit and the maximum plane change.

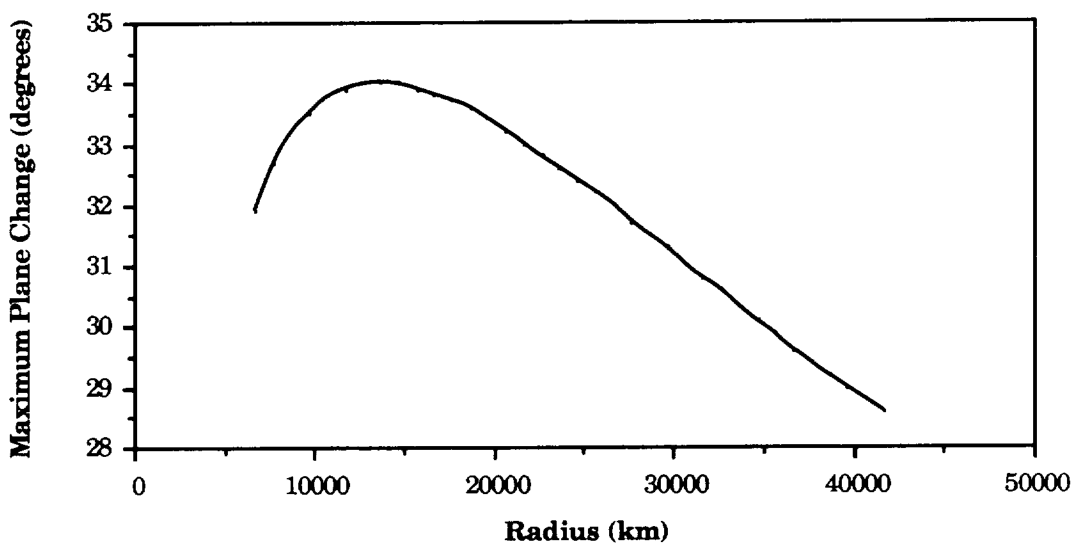


Figure 3.7: Maximum Plane Change Allowed With ΔV = 4265 m/s

Note that Figure 3.7 begins with positive slope. In this region, the ΔV cost of boosting and circularizing are low with respect to the ΔV cost of the plane change. The curve peaks at approximately 13,500 km. At this altitude, a maximum plane change of 34 degrees is possible. Beyond this point, the slope of the curve becomes negative. This is because the ΔV cost of boosting and circularizing now are greater than the cost of making the plane change.

Figure 3.7 shows that many circular orbits are possible with no increase in the one-way ΔV requirement. This is also true of non-circular orbits, but the number of them is so large that it is beyond the scope of this analysis to investigate them.

3.2.3 Lunar Missions

Given the ΔV capabilities of the STV, missions to lunar orbit are feasible. Analyzing the trajectory and subsequent ΔV cost to accomplish a lunar mission is complex since it is a two-body problem in three dimensions. Additionally, the angle of inclination of the moon's orbit varies between 28°35' and 18°19' with time so that the required ΔV's also vary with time.

We performed a rough approximation of the ΔV required by modelling the flight path as a Hohmann Transfer to lunar altitude. We assumed the STV would release its payload before it reached GEO altitude. The payload would then continue its trajectory until it reached Lunar altitude. This, of course, assumes a payload with reasonable propulsive capabilities so that once it reached the moon, it could inject itself into lunar orbit. After releasing the payload, the STV would continue on its orbit and circularize at GEO.

This maneuver would require 7283 m/s, compared to 4464 m/s for a nominal DET to GEO. Although this ΔV is higher than the nominal ΔV, careful mission planning could still allow the STV to perform this maneuver if the mass of the payload were small enough.

This type of mission may be useful for many reasons. For example, the STV could deliver construction parts and supplies to help build a lunar colony. Or, it could deliver scientific satellites for lunar study.

3.2.4 The Orbital Maneuvering Vehicle

Future missions could be enhanced greatly if the Orbital Maneuvering Vehicle (OMV) were used in conjunction with the STV. The OMV will be part of NASA's future space infrastructure, and it is proposed to be the "work horse" for various orbital needs. The OMV is a less massive and more simple space craft than the STV, giving it much greater maneuverability and efficiency for in-orbit operations.

Although the OMV is not capable of transferring to GEO under its own power, the STV could carry it there where it could be used to enhance subsequent STV missions. For example, the OMV could be used as a ferry while the STV was occupied in GEO operations such as satellite repair. The STV would serve as a base and the much less massive OMV would travel about GEO, replacing repaired satellites and gathering new ones to bring back to the STV. The OMV could also be used to assemble a network of satellites brought to GEO by the STV as it returned to the Space Station.

A possible mission scenario is the modification of an existing network of satellites. Instead of replacing the network with a brand new one, the OMV would collect the dated satellites of the existing network and group them in one position in GEO. Then the STV would travel to that position and astronauts would complete the necessary modifications to the satellites. Then the STV would return to LEO while the OMV replaces the network satellites in their proper positions.

Using the OMV would present a substantial savings in fuel. Because the OMV is so much less massive than the STV, it uses less fuel for orbital operations. It also makes the space infrastructure more efficient, an important long term goal of NASA.

3.3 Selection of Transfer Methods

For the nominal mission, we use the Direct Elliptical Transfer and the Hohmann Transfer. Another common transfer method is the Three Impulse Transfer. In this section, we will describe the Three Impulse Transfer briefly and give some insight as to why it was not used in the nominal mission.

3.3.1 The Three Impulse Transfer

The Three Impulse Transfer consists of two Hohmann Transfers made back-to-back (Refer to Figure 3.8). The first impulse or main engine burn is done on the hinge line and places the STV in the first Hohmann

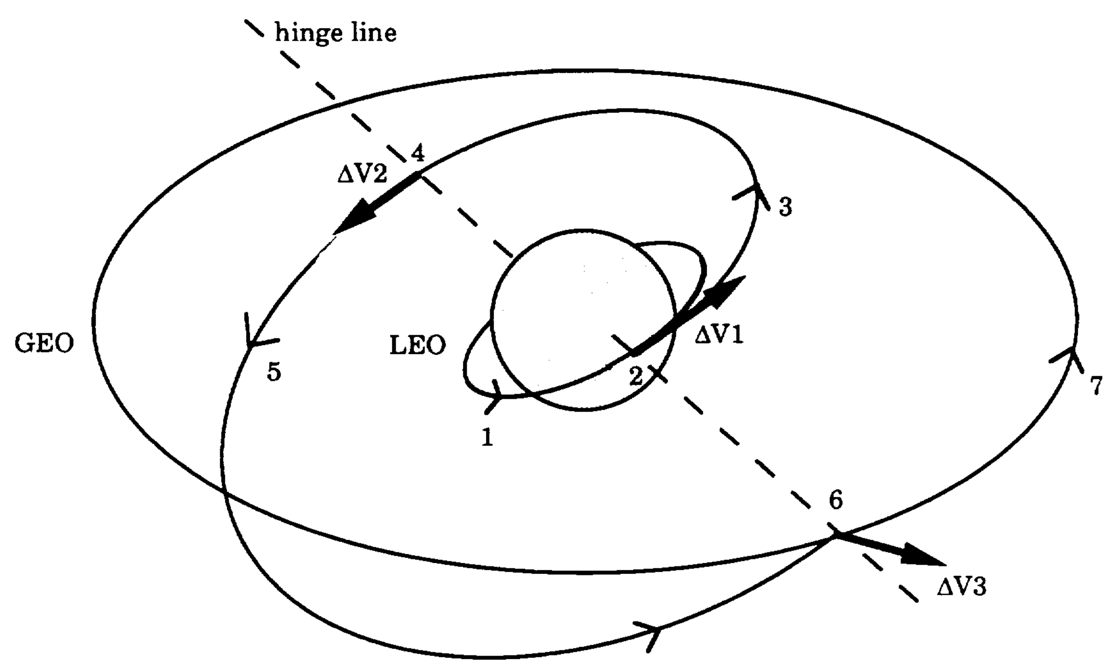


Figure 3.8: Three Impulse Transfer

ellipse which sends it to an intermediate altitude. Once there, the second impulse is immediately applied. This impulse places the STV in the second Hohmann ellipse which brings it up to GEO altitude. Here, the third impulse is applied which circularizes the STV. The entire maneuver covers 360° of travel.

It is desirable in some cases to have the intermediate altitude be greater than GEO altitude. This is due to the 28.5° plane change requirement. The higher the altitude at which the plane change is effected, the lower the corresponding ΔV cost.

The Three Impulse Transfer gives the ability to target arbitrary points in GEO. This ability is derived from the fact that varying the intermediate altitude varies the transfer time. Once the target is selected, it can then be determined how long it will take for it to reach the hinge line. Then, careful selection of the intermediate altitude will give a transfer time such that the STV reaches the hinge line at the same time as the target. The total transfer time is the sum of the half-periods of the two transfer ellipses. We wrote a computer program which numerically solved for the intermediate radius using the following equation:

$$\frac{1}{2}\left\{\frac{2\pi}{\sqrt{\mu}}\left(\frac{R_{leo}+R_I}{2}\right)^{3/2}\right\}+\frac{1}{2}\left\{\frac{2\pi}{\sqrt{\mu}}\left(\frac{R_I+R_{geo}}{2}\right)^{3/2}\right\}=t_r \tag{3.20}$$

In Equation 3.20, the left hand side is the sum of the half-periods of the two transfer ellipses. The right hand side, t_r, is the time it takes for the target to reach the hinge line and depends on the original position of the target.

Once the radius of the intermediate orbit is known, the necessary ΔV's can be determined. Just as with the Hohmann Transfer, different amounts of plane change may be done at each of the three altitudes to optimize the maneuver. We defined α as the amount of plane change done at LEO, β as the amount of plane change done at the intermediate altitude, and γ as the amount of plane change done at GEO. The sum of these three angles is always 28.5°, thus giving the required total plane change. The three ΔV's of the burn are computed with the following equations:

$$\Delta V_1=\sqrt{\frac{\mu}{R_{leo}}}\sqrt{3-\frac{2R_{leo}}{R_{leo}+R_I}-2\sqrt{2-\frac{2R_{leo}}{R_{leo}+R_I}}\cos\alpha} \tag{3.21}$$

$$\Delta V_2=\sqrt{\frac{\mu}{R_I}}\sqrt{4-\frac{4R_I^2}{(R_{leo}+R_I)(R_I+R_{geo})}-2\sqrt{\left(2-\frac{2R_I}{R_{leo}+R_I}\right)\left(2-\frac{2R_I}{R_I+R_{geo}}\right)}\cos\beta} \tag{3.22}$$

$$\Delta V_3 = \sqrt{\frac{\mu}{R_{geo}}}\sqrt{3 - \frac{2R_{geo}}{R_I + R_{geo}} - 2\sqrt{2 - \frac{2R_{geo}}{R_I + R_{geo}}}\cos\gamma} \qquad (3.23)$$

The total ΔV for the maneuver is the sum of these three.

3.3.2 Transfer Method Criteria

The STV's nominal mission requirement is the ability to transfer a payload from LEO to GEO and back. Because the Space Station's orbit is inclined 28.5° with respect to equatorial GEO, the transfer must also incorporate a 28.5° orbital plane change. With this in mind, we investigated the following three orbital transfer methods:

1) The Hohmann Transfer
2) The Three Impulse Transfer
3) The Direct Elliptical Transfer

Each of these transfer methods satisfies the basic mission requirement.

The nominal mission consists of two orbital transfers. First, the transfer from SSF in LEO to a specific point in GEO, and second, the return from GEO to LEO. We had to choose the proper transfer method from among the three we investigated to satisfy each of the parts of the nominal mission. In making these decisions, we followed these criteria:

1) The transfer method for the LEO to GEO transfer must have the ability to target a specific position in GEO.
2) Both transfer methods must have a low ΔV to save fuel expenses.
3) Each transfer should take as little time as possible to allow maximum time for operations in GEO.
4) Both transfers must involve as few engine ignitions as possible to extend the operational life of the engines.

Figure 3.9 on p. 3-19 summarizes the results of our investigation for each transfer method. Note that the values given are for a one-way transfer.

In Figure 3.9, all the values hold for both a LEO to GEO transfer and a GEO to LEO transfer.

3.3.3 Transfer From LEO to GEO

As described in Section 3.1.2, we selected the Direct Elliptical Transfer for this phase of the mission. This transfer method most effectively satisfies all the basic requirements. It combines all of the necessary criteria: a low maximum ΔV of 4489 m/s, small maximum transfer time of 7.21 hours, only two engine start-ups, and targeting ability.

Method	ΔV Requirement	No. of Ignitions	Transfer Time	Targeting
Hohmann Transfer	4240 m/s	2	5.27 hrs	No
Three Impulse Transfer	4240 m/s - 4925 m/s	3	5.27 hrs - 23.9 hrs	Yes
Direct Elliptical Transfer	4265 m/s - 4489 m/s	2	4.10 hrs - 7.21 hrs	Yes

Figure 3.9: Characteristics of The Three Transfer Methods

We eliminated the Hohmann Transfer because it does not allow targeting of arbitrary points in GEO. This ability is necessary for the phase one transfer.

We did not use the Three Impulse Transfer for two reasons. First, the maximum transfer time is 23.9 hours. Having such high transfer times would reduce the amount of time allowed for GEO operations, thus decreasing the effectiveness of the STV. Second, the transfer involves three main engine start-ups. Since the main engines allow a limited number of start-ups before they must be replaced, the Three Impulse Transfer will lead to decreased service life of the engines.

The Three Impulse Transfer can save up to 25 m/s of ΔV over the Direct Elliptical Transfer in some cases. However, this savings does not outweigh the added cost of more frequent engine replacement and/or overhaul.

3.3.4 Transfer from GEO to LEO

For this phase of the mission, we selected the Hohmann Transfer. It gives the lowest possible ΔV, 4240 m/s, a low transfer time of 5.27 hours, and only two engine start-ups. The other two methods have higher ΔV's and transfer times because of the ability to target arbitrary points, but targeting ability is not necessary for the second phase.

3.4 Sources of Error

Several sources of error will hinder mission planners from having the STV follow a desired trajectory exactly. The previous calculations were done analytically, so any errors arising from them would be due to initial assumptions. For example, in our calculations, Earth was assumed to be perfectly spherical and homogeneous. Atmospheric effects were also neglected outside of the aerobraking maneuver.

During the aerobraking maneuver, it is possible to obtain the desired ΔV. However, to obtain the desired degree of accuracy, there is an uncertainty in the angular distance traversed in the atmosphere. This angular error may be as high as 40°. To correct for this error, the phasing orbit can be modified at first perigee by applying a small ΔV correction.

The position sensing devices on the STV will be capable of determining its position to within 20 m and its velocity to within 0.08 m/s at LEO. At GEO, the STV will be able to determine its position to within 71 m and its velocity to within 0.61 m/s. These errors will be almost negligible, and may be ignored except during close station keeping and docking. For a more detailed description, please refer to Chapter 6 of this report.

More important are the errors bound to be induced during large engine burns. Imperfections in the thrust, direction, and duration of engine burns will need to be corrected utilizing mid-course corrections. These will consist of small bursts, probably from small maneuvering thrusters, inducing small ΔV corrections. Several mid-course corrections may be necessary during the transfer trajectories to and from GEO.

For close-in maneuvering and docking, maneuvering thrusters will be used quite frequently to continually adjust position and velocity of the STV. Thus, maneuvering thrusters will be essential for the STV to correct its position and velocity.

3.5 Sample Mission Flight Plan

The following flight plan describes an STV nominal mission for both the all-propulsive and aeroassisted designs. It will first place a 2200 kg communications satellite directly over Greenwich, England. It will then orbit walk to 87° West Longitude to perform repair work on the AT&T - GTE COMSTAR D3 communications satellite (launched June 29, 1978). After the repair, the STV will orbit walk to 95° West Longitude. The COMSTAR D4 (launched February 21, 1981) satellite has replaced the COMSTAR D1 (launched May 13, 1976) satellite at this position, so the COMSTAR D1 satellite is now unnecessary. The STV will retrieve the COMSTAR D1 and return it to Space Station *Freedom* [ref. 3].

Sample Mission

Event Description	ΔV (m/s)	Event Time	Total Time After Event	kg Fuel Prop/Aero
Preflight				
Fueling	-	28 hr	- 4h 35m	-
Boarding	-	10 min	- 4h 25 m	-
Final Check-out	-	4 hr	- 25 m	-
Attitude Update	-	15 min	- 10m	-
Navigation Update	-	10 min	- 0h 0m	-
Flight Segment				
Mission Begins	-	-	0d 0h 0m	-
Undock from Space Station	-	10m	0h 10m	-
Separation maneuver to departure point	1.20	1h 45m	1h 55m	12.7/10.2
Mass Determination	-	5m	2h	-
Navigation, Attitude Update	-	15m	2h 15m	-
Direct Elliptical Transfer Initiation	2440.42	-	2h 15m	17010/13732
Engine Cut-off	-	10m 14s	2h 25m	-
Mass Determination	-	(5m)	-	-
Navigation, Attitude Update	-	(15m)	-	-
DET Completion Burn	1871.78	-	8h 9m	13046/10533
Engine Cut-off	-	7m 14s	8h 16m	-
Mass Determination	-	5m	8h 21m	-
Attitude for Ignition	-	5m	8h 26m	-
GEO Rendezvous [TPI]	86.91	1hr 59m	10h 25m	371.5/299.9
Check-out of Satellite	-	2h	12h 25m	-
Place Satellite over Greenwich	-	45m	13h 10m	-
Mass Determination	-	5m	13h 15m	-
Attitude for Ignition	-	5m	13h 20m	-
Orbit walk to new position	247.68	47h 52m	2d 13h 12m	905/709
Radar Target Acquisition	-	5m	2d 13h 17m	-

Event Description	ΔV (m/s)	Event Time	Total Time After Event	kg Fuel Prop/Aero
Attitude for Ignition, TPI	86.91	2h 4m	2d 15h 21m	317/249
Repair COMSTAR D3	-	33 h	4d 0h 21m	-
Attitude for Ignition	-	5m	4d 0h 26m	-
Orbit Walk	45.55	23h 55m	5d 0h 21m	158/124
Radar Target Acquisition	-	5m	5d 0h 26m	-
Attitude for Ignition, TPI	86.91	2h 4m	5d 2h 30m	303/237
Retrieve COMSTAR D1	-	1h	5d 3h 30m	-
Mass Determination	-	5m	5d 3h 35m	-
Navigation, Attitude Update	-	10m	5d 3h 45m	-
Hold at GEO until Hinge Line	-	1h 34m	5d 5h 19m	-

NOTE: At this point, the maneuvers are different for the two types of vehicles. The flight plan for the All-Propulsive solution is shown first:

All-Propulsive ***(from Hinge Line at GEO, ready to return to LEO)***

Event Description	ΔV (m/s)	Event Time	Total Time After Event	Fuel Req. (kg)
Hohmann Transfer Burn	1835	-	5d 5h 19m	4337
Engine Cut-off	-	2m 37s	5d 5h 21m	-
Mass Determination	-	(5m)	-	-
Navigation, Attitude Update	-	(10m)	-	-
Phasing Maneuver Entry	2335	-	5d 10h 33m	5556
Engine Cut-off	-	3m 20s	5d 10h 36m	-
Mass Determination	-	(5m)	-	-
Navigation, Attitude Update	-	(10m)	-	-
Circularization below SSF	103	-	5d 12h 9m	246

Event Description	ΔV (m/s)	Event Time	Total Time After Event	Fuel Req. (kg)
Mass Determination	-	(5m)	-	-
Attitude for Ignition	-	(5m)	-	-
Wait at LEO for favorable rendezvous conditions	-	37m	5d 12h 56m	-
TPI	31	49m	5d 13h 45m	45.5
Navigation, Attitude Update	-	10m	5d 13h 55m	-
Proximity Approach	1.2	6m	5d 14h 1m	1.76
Contact with SSF Space Arm	-	20m	5d 14h 21m	-

Aero-Assisted Vehicle ***(from Hinge Line at GEO, ready to return to LEO)***

Event Description	ΔV (m/s)	Event Time	Total Time After Event	Fuel Req. (kg)
Hohmann Transfer Burn	1856	-	5d 5h 19m	4380
Engine Cut-off	-	14m 32s	5d 5h 34m	-
Mass Determination	-	(5m)	-	-
Navigation, Attitude Update	-	(10m)	-	
* Aerobraking Maneuver *	(2355)	15m 12s	5d 10h 45m	-
Navigation, Attitude Update	-	(10m)	-	-
Perigee Lift Burn	33	-	5d 11h 33m	63.5
Attitude Update	-	(5m)	-	-
Apogee Adjustment: Angle error	36	-	5d 12h 21m	69.1
Navigation, Attitude Update	-	(10m)	-	-
Circularization below SSF	169	-	5d 21h 49m	313.2
Mass Determination	-	5m	5d 21h 54m	-
Navigation, Attitude Update	-	10m	5d 22h 4m	-
Wait at LEO for favorable rendezvous conditions	-	26m	5d 22h 30m	-
TPI	31	53m	5d 23h 23m	55.4

Event Description	ΔV (m/s)	Event Time	Total Time After Event	Fuel Req. (kg)
Navigation, Attitude Update	-	10m	5d 23h 33m	-
Proximity Approach	1.2	6m	5d 23h 39m	1.96
Contact with SSF Space Arm	-	20m	5d 23h 59m	-

Post-Flight

Event Description	ΔV (m/s)	Event Time	Total Time Prop/Aero
Contact with SSF Space Arm	-	-	5d 14h 21m / 5d 23h 59m
STV Berthed into hanger	-	10m	5d 14h 31m / 6d 0h 09m
Verify STV-SSF interface	-	05m	5d 14h 36m / 6d 0h 14m
Shut down procedure	-	30m	5d 15h 06m / 6d 0h 44m
Close hanger	-	15m	5d 15h 21m / 6d 0h 59m
Crew exits vehicle	-	10m	5d 15h 31m / 6d 0h 09m

Turnaround: 26 - 77 hours, depending on next mission requirements

Totals: All-Propulsive: 5 days, 15 hours, 31 minutes to complete mission
42309 kg fuel required

Aeroassisted: 6 days, 0 hours, 9 minutes to complete mission
30777 kg fuel required

3.6 References

3.1 Buning, Harm, *Mission Analysis and Orbital Operations*, Ann Arbor, MI,Michigan Book & Supply, 1984

3.2 *Engineering and Configurations of Space Stations and Platforms*, Noyes Publications, 1985. Library of Congress Call Number: TL 797 .E541

3.3 *1983 Satellite Directory*, Bethesda, MD, Phillips Publishing, Inc., 1983

Chapter 4

Atmospheric Flight

4.0 Summary

Project Argo requires a retrograde velocity change (ΔV) near Earth upon return to LEO from GEO. This ΔV changes the orbit from a highly elliptical orbit connecting GEO and LEO to a nearly circular phasing orbit near LEO. The CSTV uses its engines to produce this ΔV, which uses fuel. In contrast, the ASTV enters the upper atmosphere, and uses aerodynamic drag to produce the required ΔV. After the decircularizing burn at GEO, the ASTV will follow an elliptical orbit until it enters the atmosphere. At this point it will be moving at speeds of approximately 10.1 km/s, and aerodynamic heating will be severe.

An aerobrake is a shield attached to the ASTV. The aerobrake has two functions: first, to protect the ASTV from the aerodynamic heating, and second, to vary the aerodynamic characteristics of the vehicle, such as lift and drag. These aerodynamic characteristics are not important for CSTV's, which never enter the atmosphere.

To shield the body of the ASTV from the hostile environment encountered during the atmospheric flight, the aerobrake must include a thermal protection system (TPS). The material of the TPS must be non-ablative to be reusable and therefore cost efficient. In Project Argo the TPS consists of the latest generation heat resistant ceramic tiles, Fibrous Refractory-Composite Insulation (FRCI-40). Also, because the aerodynamic forces on the aerobrake are large, the TPS must have a supporting structure to hold it together. The structure of the aerobrake for Project Argo consists of a strong, low-density, heat resistant material made of graphite polymide.

There are several different aerobrake designs, and for our evaluation it is convenient to classify them by the amount of lift each design can produce. We have grouped these designs into three classes: low-, mid-, and high-lift aerobrakes. The amount of lift also controls the amount of drag for a specific design: high-lift designs have low drag because they are streamlined, and low-lift designs are blunt and have high drag.

A lifting aerobrake design is used in Project Argo to allow directional control during atmospheric entry. The direction of the lift force is controlled by varying the bank angle of the spacecraft: thus rolling the spacecraft changes the trajectory. This control is vital to ASTV's, since the atmospheric density is not predictable. The lift is used to account for off-nominal atmospheric density or small errors in entry angle.

Analysis of the three types of aerobrakes showed that for Project Argo, only the low-lift designs had heating rates low enough for FRCI-40. Of the possible low-lift aerobrake configurations, Project Argo employs a spherical raked-cone design. This shape is a cone with a circular base of 15 meters, with the tip of the cone blunted off in such a way that the stagnation point is off-center. An angle of attack is needed to produce lift. This raked shape is necessary to give aerodynamic stability at an angle of attack with respect to the direction of travel.

4.1 Trajectory Analysis

4.1.1 Introduction

Designing an Aeroassisted Space Transfer Vehicle requires a detailed study of the trajectory of the spacecraft through the atmosphere. Several questions need to be addressed:

- Can the ASTV get the required ΔV from aerodynamic drag alone?
- Will the TPS withstand the aerodynamic heating?
- Can the crew withstand the decelerations?

Because the drag, heating, and decelerations depend on the velocity of the ASTV and the local atmospheric density, these questions can be answered if the trajectory of the spacecraft is known.

4.1.2 Computer Model

Analytical solutions for the trajectory are difficult to find, because the equations of motion of the ASTV through the atmosphere are non-linear. This is in contrast to orbital trajectories above the atmosphere, in space, where analytical solutions are usually quite accurate. To attack this problem, then, we developed a simplified model of the trajectory, and integrated the resulting equations of motion numerically. These simplifying assumptions are:

- Planar motion, in the equatorial plane
- Exponentially varying atmospheric density
- Constant aerodynamic coefficients: C_L, C_D, and m

The method of integration used is the Runge-Kutta two-step method. The program starts with initial conditions of velocity, altitude, and flight path angle, and steps forward by a time step small enough to achieve the desired accuracy. The program continues until the ASTV crashes or reaches apogee at its new orbit, after passing through the atmosphere. A solution file is generated, storing velocity, altitude, and flight path angle at each point along the trajectory. This data is used to calculate all desired information. The lift and drag forces vary with altitude; since they are proportional to density and velocity, the forces are greater at lower altitudes, where the atmospheric density is higher. To find the correct trajectory, we must come into the atmosphere just low enough so the drag forces give the required ΔV to arrive at LEO.

There are two subtleties involved here; first, when we start the program we do not know the initial conditions at GEO that will result in the desired trajectory, and second, only a very small portion of the trajectory requires numerical solution. The portion of the trajectory above the atmosphere can be calculated in one step, because the solution here is just an ellipse. We have used 250 km as the beginning point for our numerical integration of the trajectory, because the atmospheric effects are negligible above this point. The trajectory is calculated in two steps: the elliptical portion from GEO to 250 km, and the numerically solved portion from 250 km to either the surface of the earth or the apogee of the new orbit. Figure 4.1 and 4.2 show a sample trajectory, with a large elliptical orbit and a non-elliptical trajectory within the atmosphere.

With fixed aerodynamic coefficients and no mid-course corrections, the initial velocity at GEO determines a unique trajectory. The trajectory is very sensitive to this velocity; changes of 1 m/s can result in apogees hundreds of kilometers apart. It is easier to use the flight path angle at the top of the atmosphere (250 km), or the "entry angle", as the parameter that determines the orbit. Using this parameter, it is obvious that a steeper entry angle will result in a trajectory that moves deeper into the atmosphere. This will result in higher drag forces and a lower apogee. A more shallow entry angle will result in a trajectory that remains in the upper parts of the atmosphere, which causes smaller drag forces and a higher apogee.

To find the entry angle that will bring the ASTV to GEO, we begin by choosing any entry angle, then find the resulting trajectory. If the ASTV crashes, or does not reach LEO, a more shallow entry angle is used. If the ASTV comes out above LEO, a steeper entry angle is used. This process is repeated until the proper entry angle is found, and the heating rates and decelerations from this run are used.

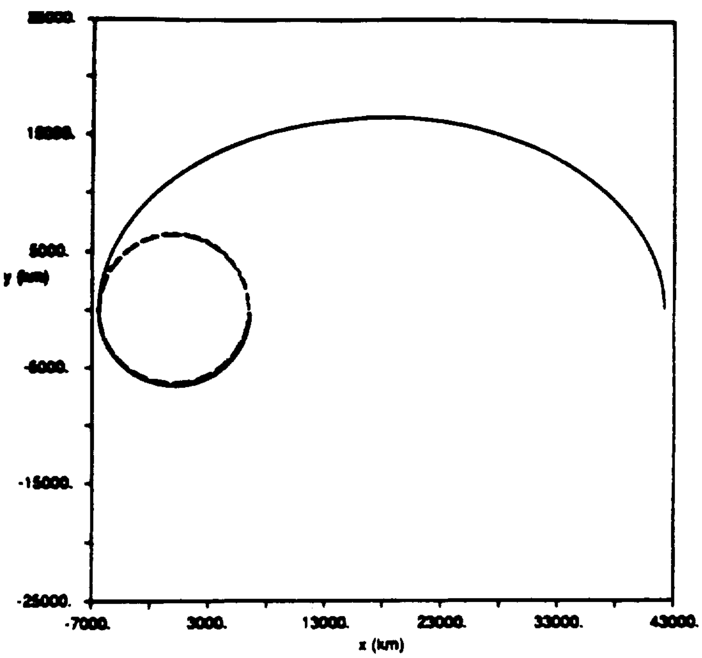


Figure 4.1. Typical ASTV trajectory from GEO to LEO

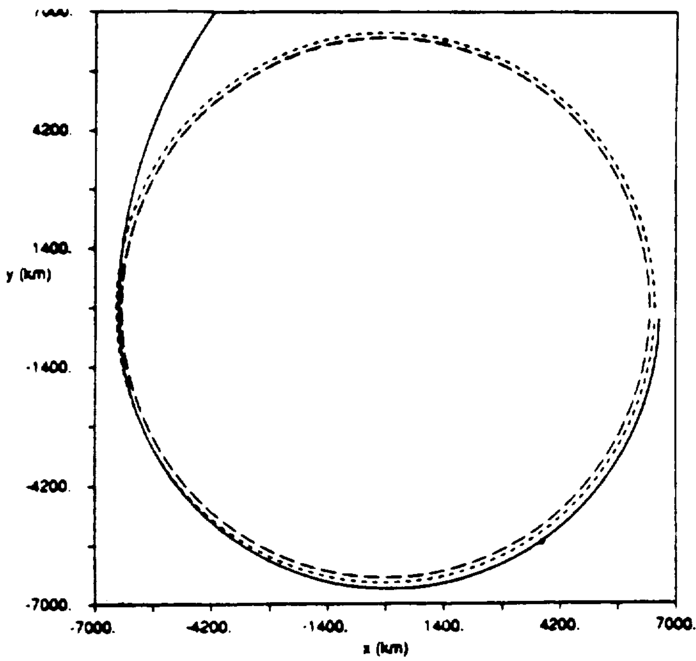


Figure 4.2. Blow up of portion of trajectory from Fig. 4.1 near earth.

Effects of Drag

The drag force on the spacecraft is controlled by the parameter $\frac{m}{C_D A}$, called the ballistic coefficient. The smaller the ballistic coefficient, the larger the drag force. Blunt ASTV's have low ballistic coefficients, and streamlined ASTV's have high ballistic coefficients. We found that with a high ballistic coefficient, the ASTV must come quite deep into the atmosphere (around 60 km) to end up at LEO. With a low ballistic coefficient, the ASTV can come in with a shallow entry angle, and can stay in the less dense upper atmosphere. This parameter has a dominant effect on heating rates. Large ballistic coefficients produce high heating rates and low ballistic coefficients produce low heating rates. A low ballistic coefficient is essential to keep heating rates low enough for a reusable TPS.

Effects of Lift

The lift force on the spacecraft is determined by the ratio L/D. The effects of lift on the trajectory are not as easy to understand as the effects of drag. Figure 4.3 shows three trajectories, with no lift, positive lift, and negative or earthward lift, all with the same entry angle. In this and the following graphs, the trajectories are shown in coordinates of altitude and polar angle, theta. Positive lift pulls the spacecraft up from the no lift trajectory. Along this trajectory the altitude is higher, and the drag forces less than the no lift case, so positive lift results in a higher apogee. Negative lift pulls the spacecraft deeper into the atmosphere, resulting in larger drag forces. In this case the spacecraft does not leave the atmosphere. These trajectories demonstrate that even a small amount of lift (L/D = 0.25) can cause drastic changes in trajectory. By rolling the spacecraft, the direction of the lift force is changed, and this can be used to control the trajectory of the spacecraft.

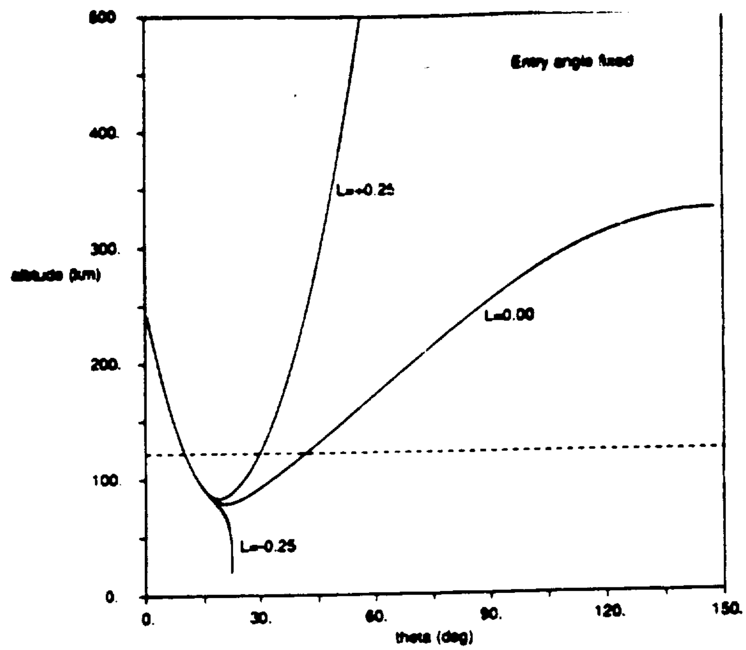


Figure 4.3. The effects of lift with entry angle fixed.

However, we are only interested in trajectories which end up at LEO. Figure 4.4 shows three trajectories with positive, negative, and no lift, with the entry angle for each adjusted to give apogee at LEO. Now it is apparent that negative lift allows the ASTV to stay in the upper atmosphere, but with positive lift the ASTV is required to come in at a steeper angle to get the required ΔV. This strongly increases heating rates and decelerations.

4.1.3 Heating

The aerodynamic heating of the aerobrake is caused by the extremely large speeds involved. The ASTV enters the atmosphere at speeds around 10.1 km/s, which roughly corresponds to a Mach number of 30. At these speeds and in the rarified upper atmosphere, the flow around the aerobrake is very difficult to analyze. There are two types of heating involved: radiative heat transfer, and convective heat transfer. The radiative heat transfer is caused by the extremely high temperatures in the detached bow shock in front of the spacecraft. This high temperature gas radiates a significant amount of heat, and approximately half of this radiation is incident upon the spacecraft. We have assumed that the radiative heat transfer rate is constant while the ASTV is in the atmosphere, and is predicted to be 10 W/cm^2 by reference 4.1.

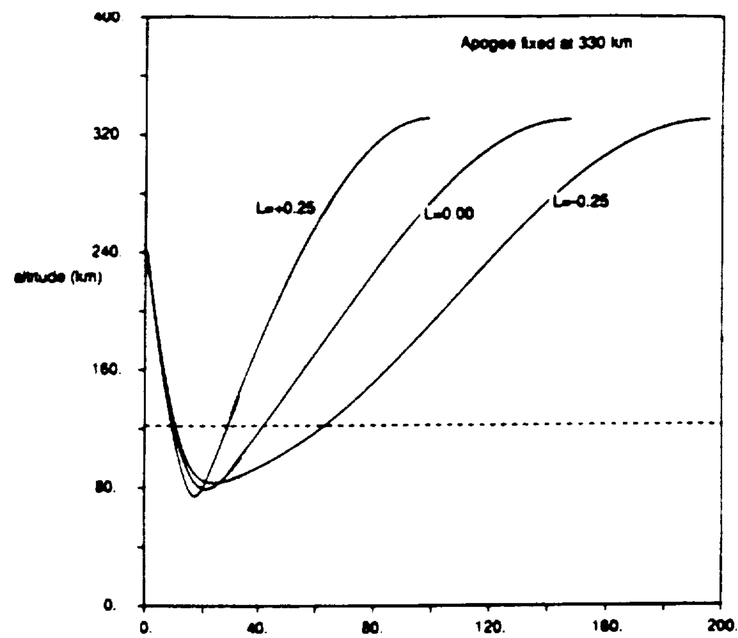


Figure 4.4. Entry angle adjusted so apogee of new orbit is at LEO

The convective stagnation point heat transfer rate is calculated from the engineering correlation formula [ref. 4.1]:

$$\frac{dq_s}{dt} = 1.83 \times 10^8 \rho^{0.5} v^3 \frac{1}{\sqrt{R_N}} \qquad \frac{W}{cm^2 \ m^{0.5}}$$

This equation shows that the heating is proportional to density and velocity. Therefore, the deeper the ASTV comes into the atmosphere, because the densities are higher, the heating rates will also be higher. Figure 4.5 shows heating histories for the three trajectories shown in Figure 4.4. Figure 4.5 also shows that L/D can be used to reduce heating rates if it is used in the negative direction, toward the earth. This is because the trajectory with L/D = -0.25 has the highest minimum altitude of the three, and is the most shallow trajectory.

Using the engineering correlation formula above and the trajectory computing program, it is possible to find the maximum heating rates for several values of ballistic coefficient and lift-to-drag ratio. The amount of available lift (maximum L/D) is roughly determined by the ballistic coefficient. Using a general relation between ballistic coefficient and L/D given by reference 4.2, and pointing all available lift in the negative, or earthward direction, we have found trajectories for a wide range of ballistic

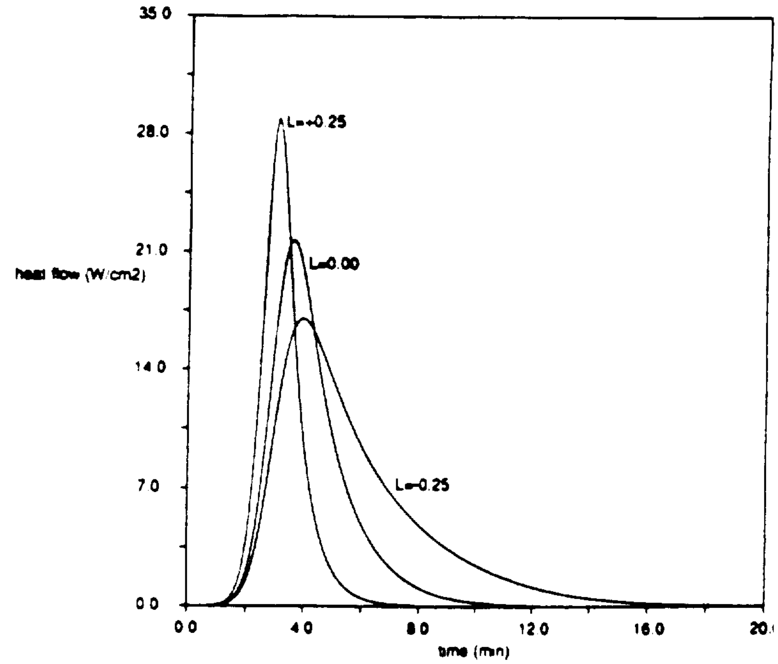


Figure 4.5. Heating rates during trajectories shown in Figure 4.4

coefficient. The maximum total heating rates (convective plus radiative) along these trajectories are shown in Figure 4.6. Each point on this curve represents a trial run of the program. The horizontal line shows the maximum allowable heating rate for FRCI-40. This line sets the upper limit on ballistic coefficient, about 300 kg/m^2. The lower limit is set by the maximum size limit for the ASTV, which is defined by the payload of the HLLV (see Logistics and Support). This is a diameter of 15 meters and corresponds to a ballistic coefficient of about 30 kg/m^2. These two limits restrict the choice of aerobrake to a low-lift design, since all mid-lift and high-lift designs have ballistic coefficients above 300 kg/m^2.

4.1.4 Decelerations

The deceleration the ASTV and its crew feels is simply the vector sum of the lift force and the drag force, divided by the mass of the ASTV.

$$a = \frac{F}{m} = \sqrt{L^2 + D^2}$$

To calculate the lift and drag forces, we have used:

$$L = \frac{1}{2}\rho\, v^2\, C_L\, A \qquad D = \frac{1}{2}\rho\, v^2\, C_D\, A$$

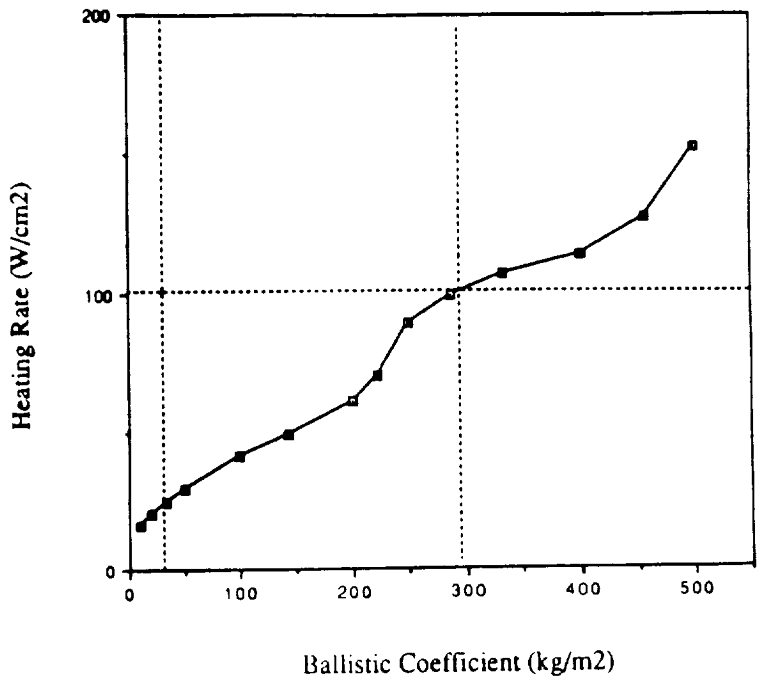


Figure 4.6. Maximum total heating rates and ballistic coefficient limits.

Figure 4.7 shows the decelerations for the three trajectories in Figure 4.4. Similar to the heating, the deeper the spacecraft comes into the atmosphere, the higher the decelerations. All of the designs with ballistic coefficients we considered for our design had maximum decelerations along the trajectory of 2.5 or less, which is within the limits sustainable by the crew.

4.2 Aerobrakes

4.2.1 Types

There are several different aerobrake designs, and the choice of aerobrake depends on the specific mission it is expected to perform. The different types can be classified by lift or by ballistic coefficient. For our evaluation it is convenient to classify them by lift. We have grouped these designs into three classes: low-lift, mid-lift, and high-lift aerobrakes. The amount of lift also controls the amount of drag for a specific design: high-lift designs have low drag because they are streamlined, and low-lift designs are blunt and have high drag.

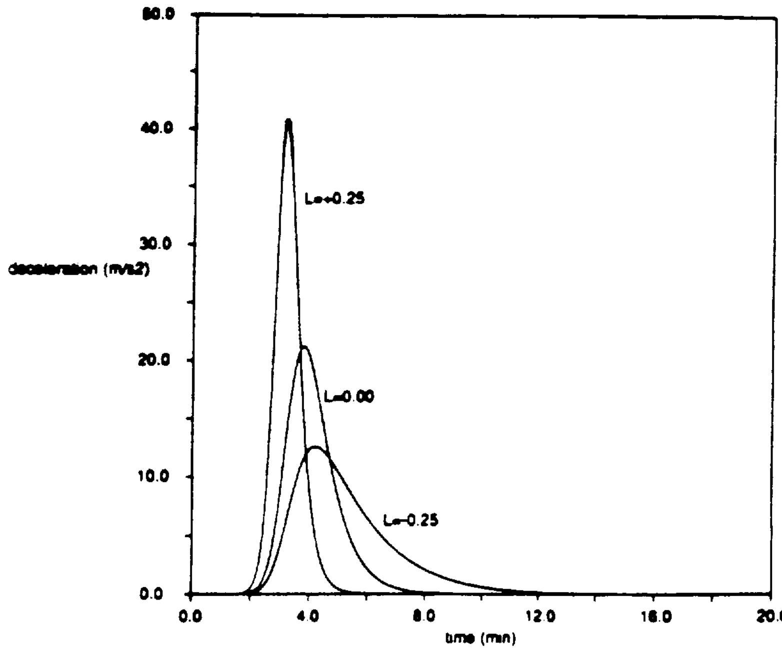


Figure 4.7. Decelerations (m/s^2) for the trajectories in Figure 4.4.

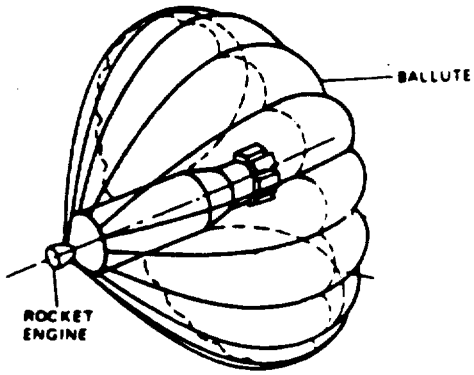
Various Aerobrake Designs (L/D)

Low-lift (0-0.5)	Mid-lift (0.5-1.5)	High-lift (1.5 and up)
ballute	biconic	lifting body
truss		
cone		

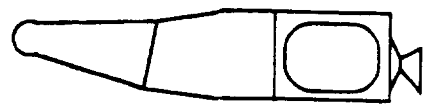
Table 4.1. Types of aerobrakes and lift classifications.

Low-lift Designs

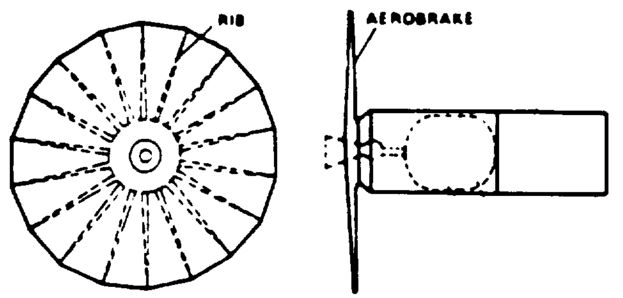
A ballute is a large inflatable structure, which when inflated surrounds the ASTV and produces an ellipsoidal nose shape with the ASTV rocket nozzle at its apex (Figure 4.8a). During the atmospheric pass, the rocket engine is fired forward with low thrust, producing a shock layer with a large separation layer near the nozzle. Because the ballute has zero lift, drag modulation by changing the engine thrust level is the only method of controlling the trajectory through the atmosphere. The exhaust plume significantly reduces the aerodynamic heating to the ballute, so a relatively low heat resistant material can be used. Nevertheless, because of its large size, the ballute is heavy, about 1200 kg. The ballute is not a reusable TPS. After leaving the atmosphere, it is discarded, and a new ballute would be used for each mission. Because of its large mass and non-reusability, the ballute is not economically feasible. Another major problem with the



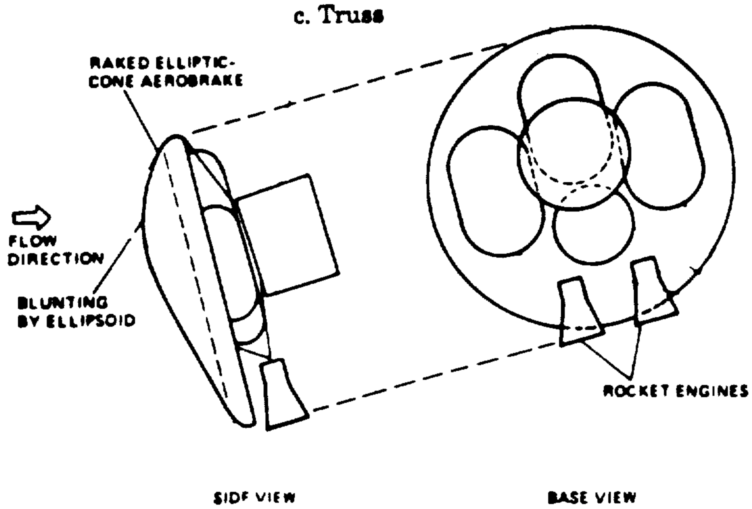
a. Ballute



b. Biconic



c. Truss



d. Spherical raked-cone

Figure 4.8. Various aerobrake designs.

ballute concept is that because the ballute surface is flexible, it changes shape during the atmospheric pass, which makes it unstable [ref. 4.1].

A truss, also called a lifting brake, is an additional structure attached to the STV to increase atmospheric drag. An example of a ribbed truss in shown in Figure 4.8c. One of the main advantages of this type of device is that it can be folded for launch and then unfolded once in LEO. This means the entire vehicle may be able to fit within the Shuttle's cargo bay or in other fairly small launch vehicles. It can also be re-folded in order to return the STV to Earth. However, Project Argo does not require these capabilities, and rigid designs are more mass efficient.

Rigid cone designs are very similar to the truss design except they are integral parts of the ASTV and they are not foldable. The advantage of this design is that the aerobrake can contribute to the structure of the ASTV. A raked-cone design is shown in Figure 4.8d. The cone is raked into a blunted cone with the stagnation point off center in order to give aerodynamic stability at an angle of attack, providing lift. For stability, the ASTV must fly at a constant angle of attack. Course corrections can be made by simply rotating the vehicle with reaction control jets to change the direction of the lift force.

Mid-lift Designs

The biconic (Figure 4.8b.) is a design where there is no aerobrake separate from the rest of the spacecraft. Instead, the components are configured in a way so the entire spacecraft is covered with a heat resistant material. As the name indicates, a biconic is made up of two cones aligned at a slight angle to produce mid-range lift, about L/D = 0.5 to 1.5. As described in Trajectory Analysis, the heating rates for this type of design are higher than those a non-ablative TPS can take. Biconics use ablative materials for insulation, which can handle the higher heating rates required. Thus biconics are suitable for one-time entry vehicles such as planetary entry probes.

High-lift Designs

The high-lift designs are called lifting bodies. These spacecraft are very streamlined, and use aerodynamic surfaces to produce high-lift. The Space Shuttle is an example of a lifting body. These designs necessarily have low drag and sharp leading edges, both leading to high heating rates. These designs are not appropriate for missions with entry velocities as high as Project Argo. Lifting bodies are advantageous for LEO to LEO missions, where high-lift provides maximum trajectory control and the ability for large plane changes.

4.2.2 Choice of Aerobrake Design

As discussed in Trajectory Analysis and the preceding section, the constraint of reusability limits the choice of specific aerobrake to a low-lift design. Of the possible low-lift designs, the ballute has stability problems, and the spherical raked-cone is more mass efficient than the truss. Therefore, Project Argo uses a spherical raked-cone aerobrake design.

By minimizing the heating rates, we can reduce the mass needed for the TPS. As shown in Figure 4.6, the lowest heating rates correspond to small ballistic coefficients. Since the ballistic coefficient is $\frac{m}{C_D A}$, and the mass and C_D of the vehicle is fixed, small ballistic coefficients are obtained by increasing the frontal area of the aerobrake. The largest diameter which will still fit in the HLLV is 15 meters. The mass of the ASTV will depend on the mass of payload retrieved from GEO, called "payload down." The following is a summary of our spherical raked-cone aerobrake parameters.

$\frac{L}{D}$	=	0.25	
Diameter	=	15 m	
C_D	=	1.5	
$\frac{m}{C_D A}$	=	$29.5 \frac{kg}{m^2}$	for 0 kg down
		$48.4 \frac{kg}{m^2}$	for 5,000 kg down
		$67.2 \frac{kg}{m^2}$	for 10,000 kg down

Figures 4.9 through 4.11 show the trajectories, heating rates and decelerations for the three values of ballistic coefficient shown above. The highest ballistic coefficient corresponds to the deepest trajectory and highest heating rates. All heating rates are within the allowable limit of 100 W/cm^2. The decelerations are all under 2.5 g's.

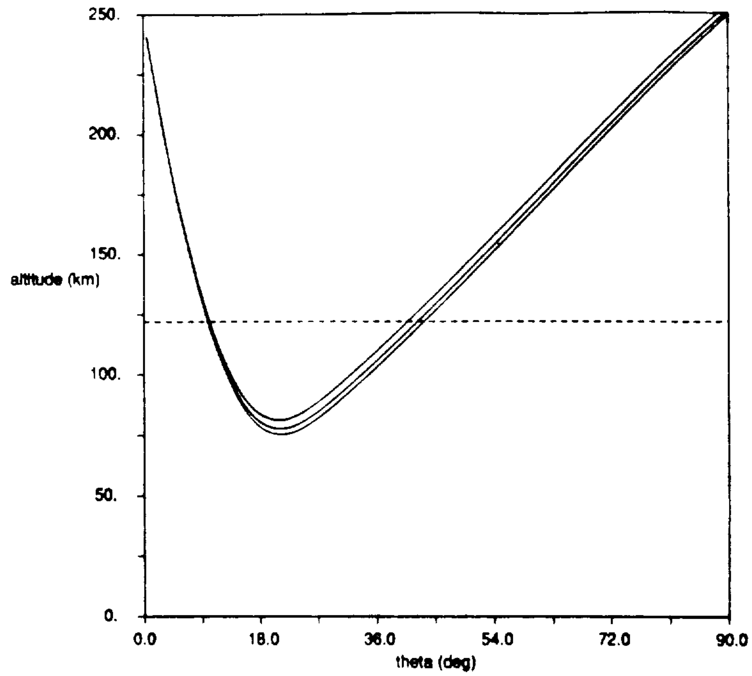


Figure 4.9. Trajectories for 0, 5000, and 10,000 kg down.

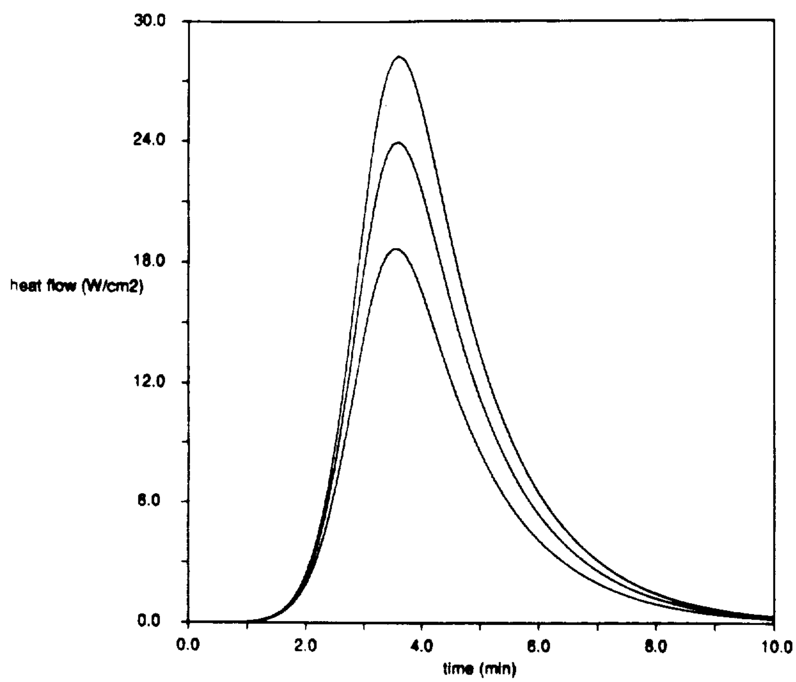


Figure 4.10. Heating rates for 0, 5000, and 10,000 kg down.

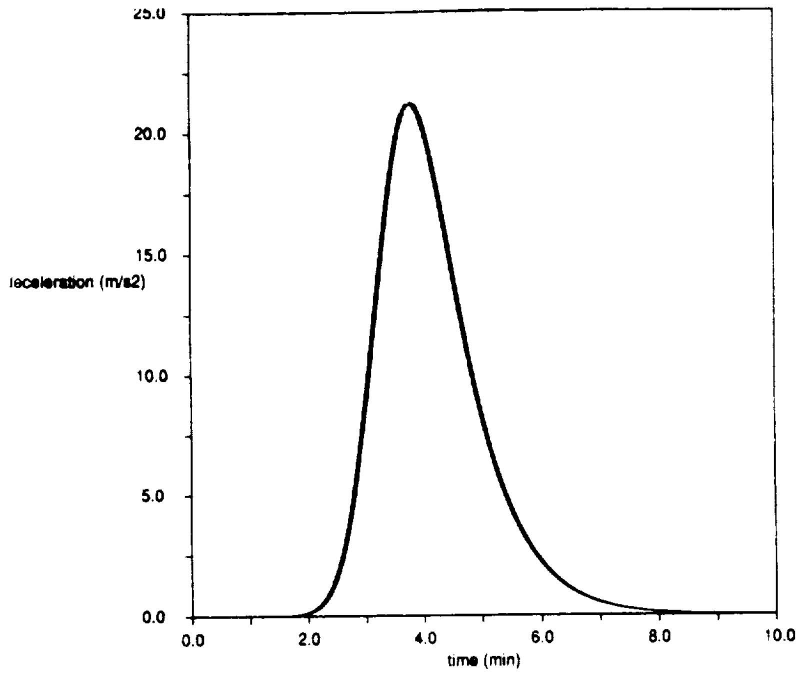


Figure 4.11. Decelerations for 0, 5000, and 10,000 kg down.

Aerodynamic Characteristics

The stability of an ASTV is a major concern. NASA Johnson Space Center [ref. 4.3] has analyzed the aerodynamic data for the spherical raked-cone configuration as shown below.

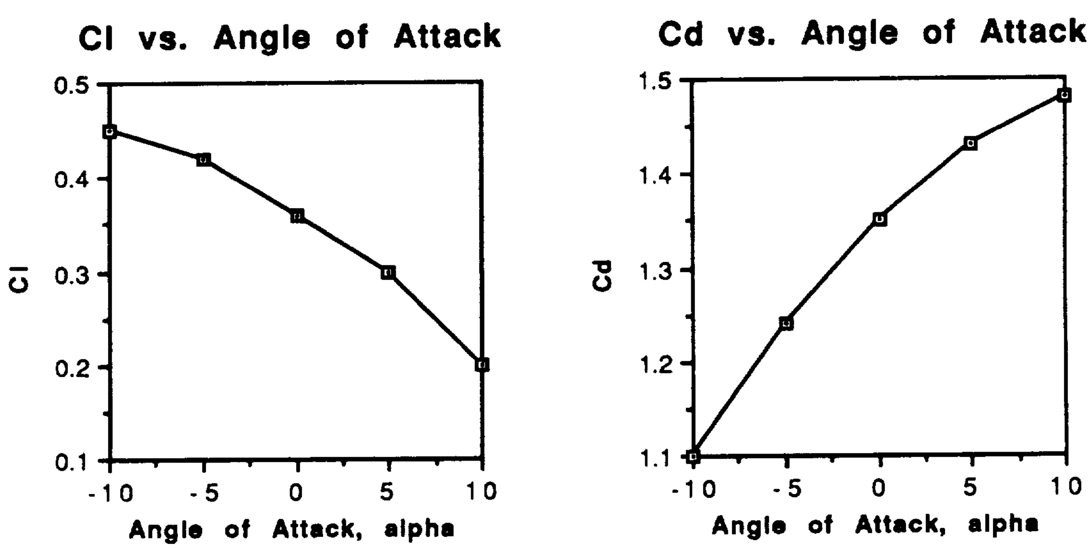


Figure 4.12. Lift and Drag Coefficients

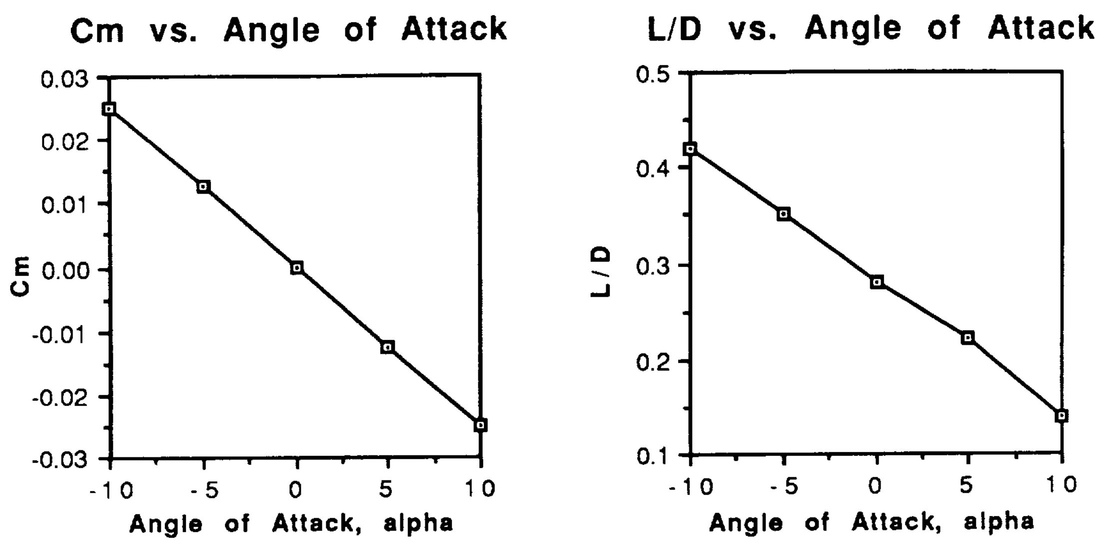


Figure 4.13. Moment coefficient and Lift-to-Drag ratio.

The angle of attack, α, is measured from the equilibrium angle of about 5° [ref. a]. The fact that $\frac{dC_m}{d\alpha}$ is negative shows that the aerobrake is stable.

Plane Change

The ability of a lifting aerobrake to provide plane change in the atmosphere is another fuel saving aspect of ASTV's. However, Project Argo assumes that all the plane change is provided propulsively at GEO by the rocket engines. Several reasons for this follow. The analysis of trajectories including plane change is more difficult, so we have assumed only planar motion. Project Argo is a low-lift ASTV, so only part of the necessary plane change could be obtained from lift. Finally, the ASTV must use a large part, if not all of its lift, to correct for atmospheric uncertainties and errors in entry angle.

4.3 Thermal Protection System

Several factors are important when chosing the thermal protection system (TPS) for the ASTV. The TPS must have adequate heat resistivity, high tensile strength, low recession rates, serviceability, and low density. Also, the TPS must be reusable, which excludes ablative materials. Mechanical cooling systems are not used because of excess mass and reduced serviceability. The two most promising systems are Rigid Surface Insulation (RSI), which consists of Fibrous Refractory Composite Insulation (FRCI) tiles, and Tailorable Advanced Blanket Insulation (TABI), which is a flexible TPS.

The flexible system is one continuous sheet which is directly bonded to the structure. Nicalon would serve as the emitting layer, with a ceramic felt filler, and a Nextel back face to increase tensile strength. TABI provides some advantages over FRCI such as: lower mass per unit area, smoother surfaces, integral construction, and a variable density. But, there are some important drawbacks to the TABI system. Most importantly this is an unproven technology; very little data exists to validate any reusability aspects or performance in a hostile environment. One such unsolved problem is the effect of hot gas flow through the material. TABI is still a research material, with little information on how the material will be manufactured or repaired in a space environment [ref. 4.4]. On the other hand, FRCI is a proven technology with its successful use on the Space Shuttle. Finally, TABI was designed for usage on retractable aerobrake systems, not on a rigid spherical raked-cone [ref. 4.5].

Both systems offer a small mass with excellent thermal protection capabilities, but for our system the RSI is the best choice. Rigid thermal protection systems are better suited for the high temperatures, approximately 2300 K, and high heating rates which will be encountered during entry. A rigid system consists of tiles which are bonded to a strain insulation pad which in turn is bonded to the aerobrake structure. The TPS should withstand up to 50 passes through the atmosphere, but allowances should be made in case of tile failure. The tiles will be tailored to fit on the graphite-polymide shell of the aerobrake, standardized to a certain number of shapes. This will allow quick and easy replacement at Space Station *Freedom*. Research can determine areas where debris damage or wear is most likely, and tiles then can be stored in space for simple refurbishment. The rigid TPS will also provide added strength to the aerobrake structure, which will be needed since the rest of the spacecraft is supported by the aerobrake.

4.3.1 Aerobrake Structure

Two criteria were used when selecting the material for the aerobrake structure: low mass and high temperature resistance. The three materials which fit these requirements are titanium, aluminum, and graphite polymide. Titanium can handle much higher temperatures than the above materials. Unfortunately we cannot take advantage of this because the thermal capabilities of the tile adhesive is the limiting factor [ref. 4.5]. The two remaining materials have small masses, with thermal masses of 2.82 for aluminum and 2.48 for graphite polymide [ref. 4.6]. But the main factor in determining the system was temperature capabilities. Here graphite polymide is the clear winner. Graphite polymide can withstand temperatures of 290 C while aluminum can only handle 175 C [ref. 4.7]. The aerobrake face will consist of a graphite polymide honeycomb structure with thin sheets of graphite polymide on either side. The honeycomb structure allows for shaping flexibility and small mass while providing sufficient stiffness. Figure 4.14 below is a cross-section of the structure including thickness and mass relations [ref. 4.4].

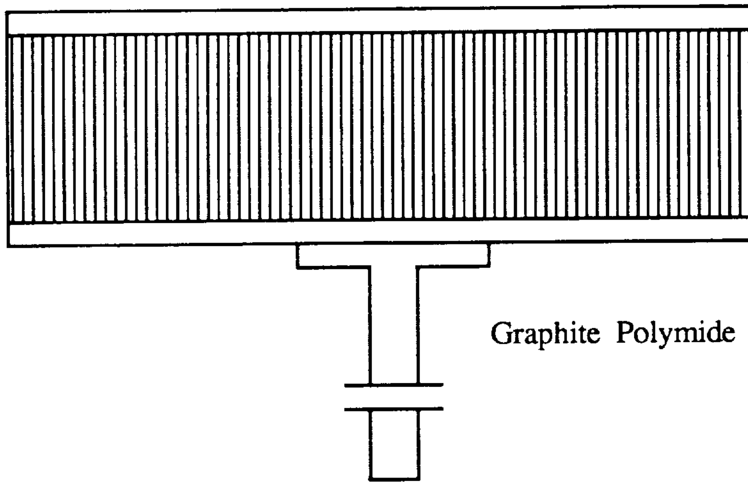


Figure 4.14. Aerobrake supporting structure.

Using the above values with an area of 177 m^2 we have the following mass:

Graphite polymide skin:	300 kg
Honeycomb structure:	100 kg
Support Structure:	400 kg
Total Structure Mass:	800 kg

Figure 4.15 shows the support structure configuration for the aerobrake [ref. 4.3].

4.3.2 Thermal Insulation

The total TPS will consist of several layers of materials. Heat dissipation will be handled by Fibrous Refractory Composite Insulation (FRCI) tiles with an Reaction Cured Glass (RCG) coating. Tiles will then be mounted on a strain insulation pad (SIP) using a silicone adhesive. For support, the system will be mounted to the above mentioned graphite polymide structure. The following is a more detailed discussion of each section.

Fibrous Refractory-Composite Insulation

The heat absorbing portion of the TPS will be made of Fibrous Refractory-Composite Insulation (FRCI) tiles. This material is a combination of silica and aluminoborosilicate fibers. Varying the amount of aluminoborosilicate can have a direct relationship on the properties of the material such as heating, recession, and tensile strength. Project Argo

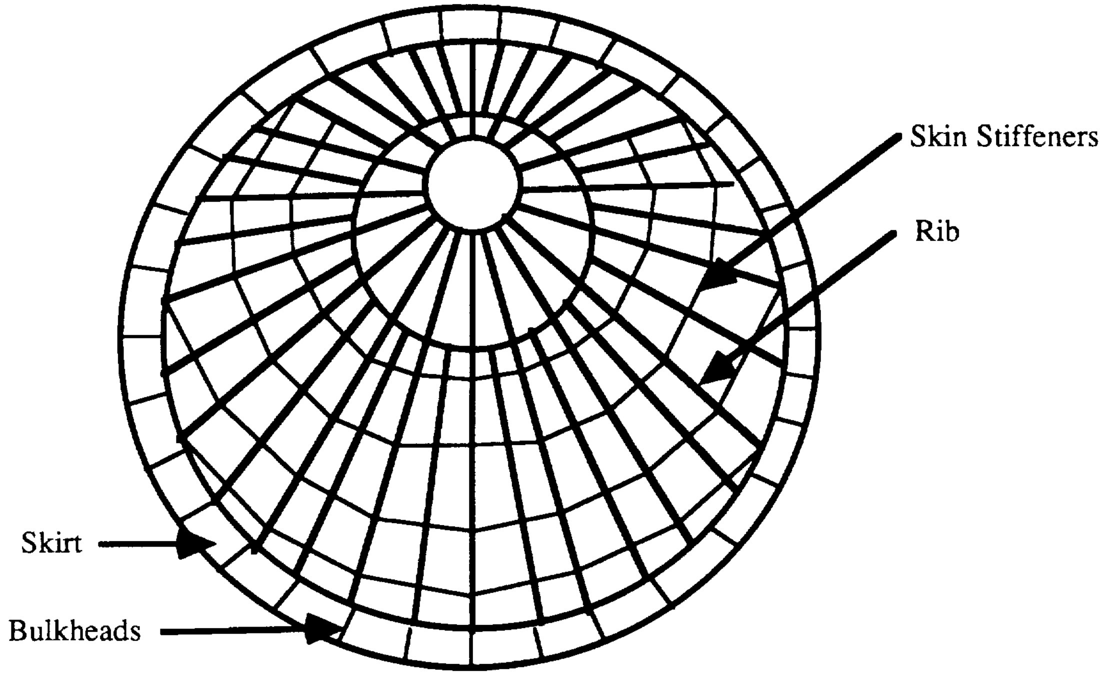


Figure 4.15. Supporting rib layout.

will use FRCI-40 tiles, which contains 40% aluminoborosilicate fibers. FRCI-40 tiles can handle temperatures up to 2600 K, provide a thermal conductivity rate of approximately $2.12x10^{-5}$ - $5.5x10^{-5}$ (W/mK), and can withstand a maximum heat transfer rate of 130 W/cm^2 [ref. 4.8]. The upper limit was set at approximately 100 W/cm^2 for the heat transfer rate. The primary reason for chosing FRCI-40 over other combinations such as FRCI-20 or FRCI-60 was the mechanical properties of the material. The mechanical properties can be varied by changing the percentage of aluminoborosilicate fibers in the tiles. The maximum value for tensile strength and modulus of rupture is obtained with 40% aluminoborosilicate fibers [ref. 4.9]. Increasing the mass percentage beyond 40% will reduce surface recession, but it will also greatly reduce tensile strength [ref. 4.10].

Tile surface recession is another concern due to the high heating rates that will be experienced. Generally, the greater recession, the less heating the tiles can withstand. The most severe recession occurs during long exposure times. In that case a tile with a greater aluminoborosilicate content, such as FRCI-60 might be required. But, the added mass penalty, due to a larger density, far outweighs the advantage. During re-entry, the ASTV will only see a heating pulse for approximately 10 minutes. Research has shown that a decrease in FRCI tile density results in little recession during short time intervals [ref. 4.11]. Therefore, FRCI-40 should be able to provide sufficient protection against surface recession without unnecessary mass.

The tile density for the aerobrake is a light 220 kg/m^3, with an average tile thickness of 1.67 cm. Mass savings was a prime concern in the aerobrake design. As a result, a low density tile was chosen to minimize mass while still providing good thermal and mechanical properties. Mechanical and temperature performance has been shown to increase with increasing density [ref. 4.9]. But, too high a density will push the TPS mass high without a substantial gain in usable performance. While the added performance would be nice, in the long run the added mass would hurt the overall efficiency of the ASTV. We feel that by minimizing density, while utilizing a higher aluminoborosilicate percentage, the tiles will provide sufficient thermal and mechanical performance while keeping the mass low.

Reaction-Cured Glass Coating

The FRCI tiles will be coated with black Reaction-Cured Glass (RCG) made of borosilicate glass. The coating is very important because it provides the aerobrake with the necessary emitting capability to maintain a suitable temperature during atmospheric entry [ref. 4.9]. An RCG coating should be able to dissipate 85% of the heat energy in radiation back into the atmosphere [ref. 4.12]. The coating will provide the TPS with an emissivity of 0.8-0.85 and will be about 0.254 mm thick. One problem with any RCG coating is its compatibility with the FRCI tile. If the thermal expansion coefficients of the RCG and FRCI tiles are not compatible the coating will fail by cracking and detaching from the tile. Research shows that the RCG coating with FRCI-40 appears to have compatible thermal expansion coefficients. On the other hand, FRCI-60 and FRCI-80 showed failure in a much shorter period of time[ref. 4.9]. The RCG will be the limiting factor in TPS lifetime. Research has shown that the RCG/FRCI system can withstand about 50 mission heating cycles before a substantial decrease in coating adherence [ref. 4.13].

Strain Insulation Pad

The tiles will be bonded to a Strain Insulation Pad (SIP) of Nomex felt. This pad will be bonded to the graphite polymide structure. The purpose of the pad is to absorb the thermal shocks and acoustic impulses caused by atmospheric entry, then transmit the aerodynamic loads to the structure [ref. 4.12]. The pad will prevent tiles from being torn from the shield due to structural deflections and failing due to stress or strain. The bonding agent will be a silicon adhesive with a temperature limit of 600 K. Below is a three dimensional view of the TPS layers [ref. 4.5].

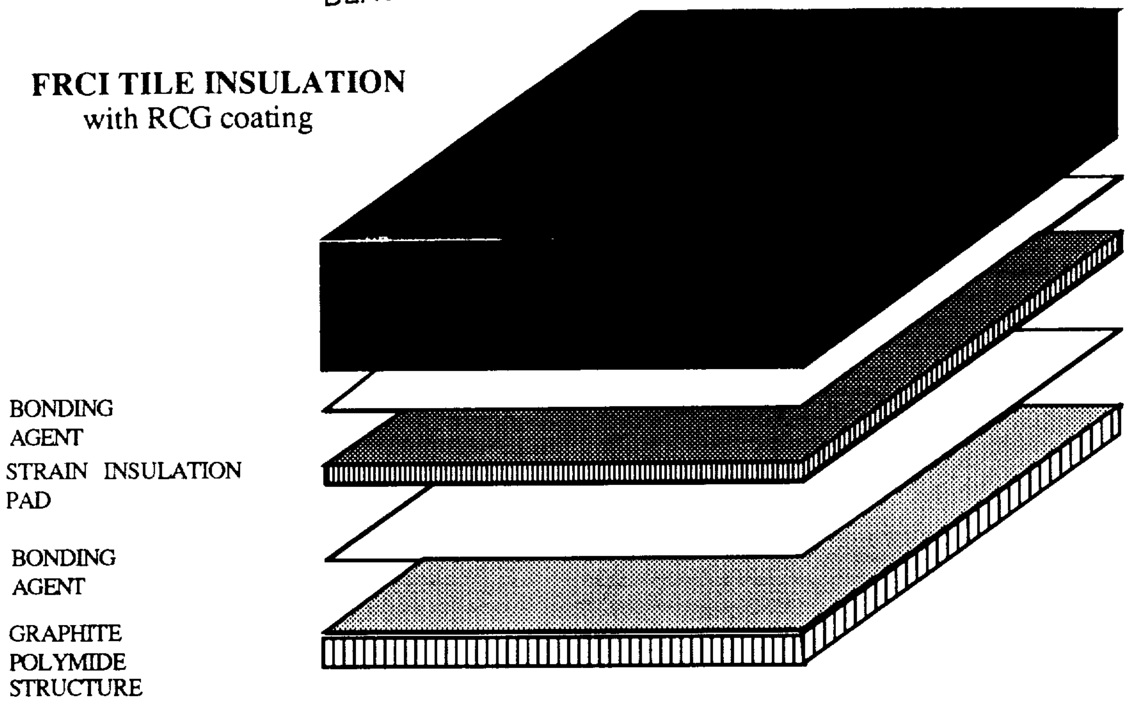


Figure 4.16. Exploded view of TPS layers.

Filler Bar

Due to the use of the tile system, gaps will be present between the tiles. Obviously to avoid damage to the aerobrake itself, the gaps must be as small as possible. Therefore, we will keep the gaps to a distance of 0.625 mm-0.70 mm. Underneath the gaps will be a filler bar made of Nomex felt with an RCG coating. This material will provide added protection for the graphite polymide against any heating loads which flow into the gaps. Below is a side view of the TPS [ref. 4.3].

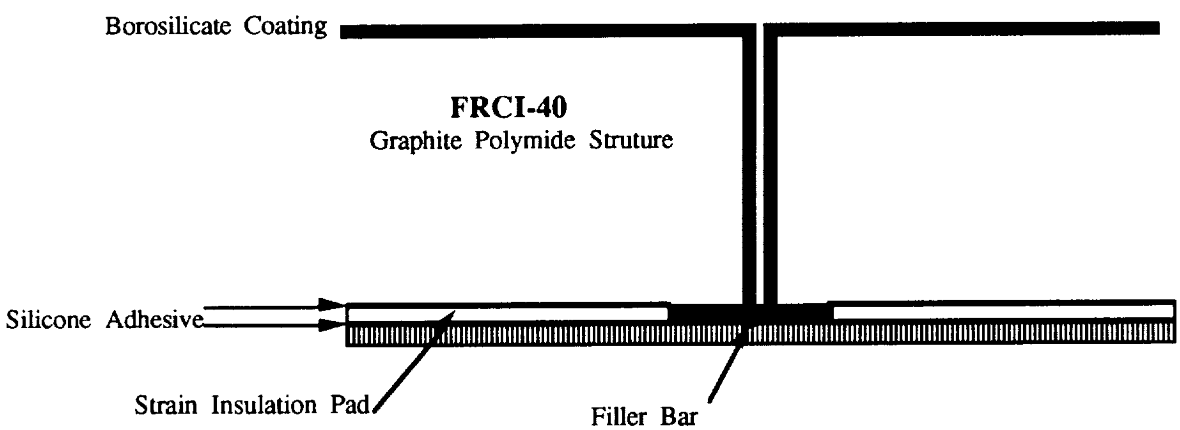


Figure 4.17. TPS attachment configurations.

Finally, it is necessary to calculate the overall mass of the TPS. Using information from reference 4.3 and an area of 177 m^2 the following values are obtained:

Material	Mass/Area	Thickness	Mass
G.Poly. Skin	0.845 kg/m^2	0.254 mm	300 kg
Honeycomb filler	0.561 kg/m^2	6.35 mm	100 kg
RCG Coating	0.392 kg/m^2	0.254 mm	70 kg
Silicone Adhesive	0.294 kg/m^2	Negligible	100 kg
Strain Pad	0.814 kg/m^2	0.254 cm	10 kg
FRCI-40 Tiles	1.242 kg/m^2	1.67 cm	650 kg
	Graphite Polymide structure mass:		400 kg
		Total Aerobrake Mass:	1630 kg

Table 4.2. TPS masses

4.4 References

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Chapter 5

Propulsion

5.0 Summary

The nature of the STV's mission makes great demands on the propulsion system. Transporting 10,000 kilograms of payload from LEO to GEO in a short period of time requires the propulsion system to be very powerful. At the same time, the high costs of bringing STV propellant into orbit demands that this system be as fuel efficient as possible. Once in orbit, the STV must be highly maneuverable in order to effectively place and retrieve satellites, and dock with the space station. The propulsion group is responsible for selecting the main engines, attitude control system, and propellant storage system of the STV so that the above requirements are satisfied.

The high thrust required to complete our mission in a short time is unavailable from nuclear or solar methods of propulsion. Chemical propulsion is the only method capable of providing thrust of the magnitude required to complete our mission. The choice of the main engines to be used hinged on the engines' fuel efficiency (specific impulse or Isp). Of all chemically fueled engines, those that burn liquid hydrogen and liquid oxygen provide the highest thrust with the highest Isp's. Therefore we selected an advanced chemical engine of this type made by Rocketdyne to propel the STV. A detailed description of this engine is given in Section 5.1.3. Two of these engines will be used to provide safe redundancy while enabling the STV to move from LEO to GEO in about five hours.

Hydrogen and oxygen propellants must be stored as cryogenic liquids in order to be used in the engines. The tanks that hold these propellants must be strong but lightweight to minimize structural mass. Additionally, to keep the propellants in a liquid form (minimizing "boiloff") the tanks must be well insulated from solar radiation and heat sources. Finally, the shape of the tanks must facilitate easy fuel acquisition for the engine feed lines. The above qualities are realized in tanks constructed of Aluminum-Lithium 2090, insulated by Polyester Dacron Tuft, and shaped like prolate ellipsoids. For more details, see Section 5.2.

In order to effectively manipulate and place satellites in orbit, the STV has to be equipped with an attitude control system which will adjust its rates of pitch, roll and yaw. Attitude control thrusters will also play a crucial role in controlling the ASTV during aeroassisted braking. While a compressed cold gas system with its non-corrosive exhaust would be very desirable for the STV, the high mass of this system made it unusable. Instead, smaller and lighter hydrogen/oxygen thrusters will serve this purpose. Extra care will be required when operating this corrosive system in the vicinity of satellites or the space station, but the fuel savings associated with the system's lower mass will be significant. A more detailed description of this system is given in Section 5.3.

The overall propulsion system configuration will be similar for both the CSTV and ASTV in that they will each contain the same components.

Both vehicles will have two main engines and 32 thrusters. The thrusters will be located in four modules each containing eight thrusters. Propellant for both vehicles will be stored in two liquid hydrogen tanks and two liquid oxygen tanks. Propulsion system diagrams for both vehicles are given in Section 5.4.

5.1 Main Propulsion System

5.1.1 Comparison of Possible Main Propulsion Systems

Three types of propulsion systems were considered for use in the STV. These are nuclear, solar, and chemical. There are two types of nuclear systems: the ion-xenon system and the nuclear electrical system. There are three types of chemical systems: those that use solid fuel, those that use a hybrid (mixture of solid and liquid) fuel, and those that use liquid fuel.

Nuclear (Ion-Xenon) Propulsion

This type of propulsion produces a very low level of thrust. For an STV with a mass of 3000 kg, a trip from LEO to GEO and back would have a travel time of almost 150 days. Our STV is more than twice this mass, thus making the nuclear ion propulsion system out of the question for our nominal mission length of seven days.

Nuclear (Electrical) Propulsion

Because of low thrust levels, this propulsion system has the same flaw as the ion propulsion system. Also, electrical nuclear propulsion is very dangerous during the atmospheric phase of the aeroassisted mission. A catastrophic failure during the atmospheric phase could result in radioactive debris falling over a wide area.

Solar (Electrical) Propulsion

Time of travel is again a factor in ruling out the use of electrical propulsion. The time required for transfer from LEO to GEO for a solar powered STV is about 25 days.

Chemical (Solid) Propulsion

Solid propellant is easily and economically stored. The major problem associated with the use of solid propellant is controlling the burn. Once ignited, all of the solid propellant must finish burning, with one exception. The burning can be controlled if we control the back pressure in the rocket. Although this type of system could be used, it is not very reliable, and it costs more than liquid or hybrid propulsion systems.

Chemical (Liquid) Propulsion

The major advantages of liquid chemical propulsion are the high Isp, controllability of the burn, and the availability of engines. Also, the velocities required by the STV can be achieved in a reasonable length of time as a result of the high thrust provided by liquid chemical propulsion. A problem associated with liquid propellant, however, is the higher cost of manufacturing, transporting, and storing it.

Chemical (Hybrid) Propulsion

This type of system uses both solid and liquid rocket propellants, the solid being the fuel and the liquid being the oxidizer. This system allows for the advantages provided by solid propellant and the burn control provided by liquid propellants. Economically, the hybrid system is better because it requires half of the components needed for liquid chemical propulsion and also solid fuel is much easier to manufacture and store than liquid fuel. The major problem with the hybrid engine, however, is its low Isp relative to liquid chemical engines. Also, there are no hybrid engines currently available and with very limited research, it does not appear any will be available in the near future.

Final Choice of Propulsion System

We chose liquid chemical propellant for the following reasons:

1) The mission length of seven days requires a high level of thrust which this system produces.

2) For a liquid propellant system, we can choose from several available engines to obtain the optimum package.

3) The liquid propellant provides control of thrust level and the rate at which fuel is burned.

5.1.2 Comparison of Engines

We compared engines that use cryogenic fuels and engines that use storable fuels. A list of the engines and some of their specifications are shown in Table 5.1 and Table 5.2 [ref. 5.1].

It should be noted that the Isp for liquid oxygen/liquid hydrogen (LOX/LH2) engines is much higher than that of monomethyl hydrazine-nitrogen tetroxide (MMH-NTO) engines as can be seen in the tables. This factor makes (LOX/LH2) engines more desirable for STV use.

Storable monomethyl hydrazine-nitrogen tetroxide (MMH-NTO) engines:

Engine	Vehicle	Manufacturer	Thrust(N)	Isp(s)
RS2101C	Viking	Rocketdyne	1,350	292
8096	Agena	Bell	71,350	300
TR-201	Delta	TRW	43,700	302
SST-OMS	Shuttle	Aerojet	26,750	316
Transtar	-	Aerojet	16,700	328
XLR-132	-	Aerojet	16,700	340

Table 5.1

Cryogenic liquid oxygen/liquid hydrogen (LOX/LH2) engines:

Engine	Vehicle	Manufacturer	Thrust(N)	Isp(s)
RL10A-3-3A	Centaur	Pratt & Whitney	74,000	447
RL10-IIB	-	Pratt & Whitney	67,000	460
RL10-IIIB	-	Pratt & Whitney	33,500	470
Advanced	-	Pratt & Whitney	33,500	485
RS-44-3	-	Rocketdyne	67,000	492

Table 5.2

5.1.3 Choice of Engine

Our choice for the main engine is Rocketdyne's RS-44 (see Figure 5.1) [ref 5.2]. Its high values of thrust and Isp allow the mission to be completed within the defined time limit with a minimal amount of fuel necessary. The specifications are shown in Table 5.3.

Thrust..................................	67,000 N
Isp......................................	492 sec
Throttling..........................	30:1
Length:	
extended.....................	332.2 cm
retracted.....................	88.0 cm
Exit diameter.....................	162.8 cm
Mass..................................	185.0 kg
Gimballing........................	6 degrees
Lifetime............................	20 hrs
....	500 startups
Maintenance free lifetime.....	4 hrs
....	100 startups

Table 5.3

C - 2

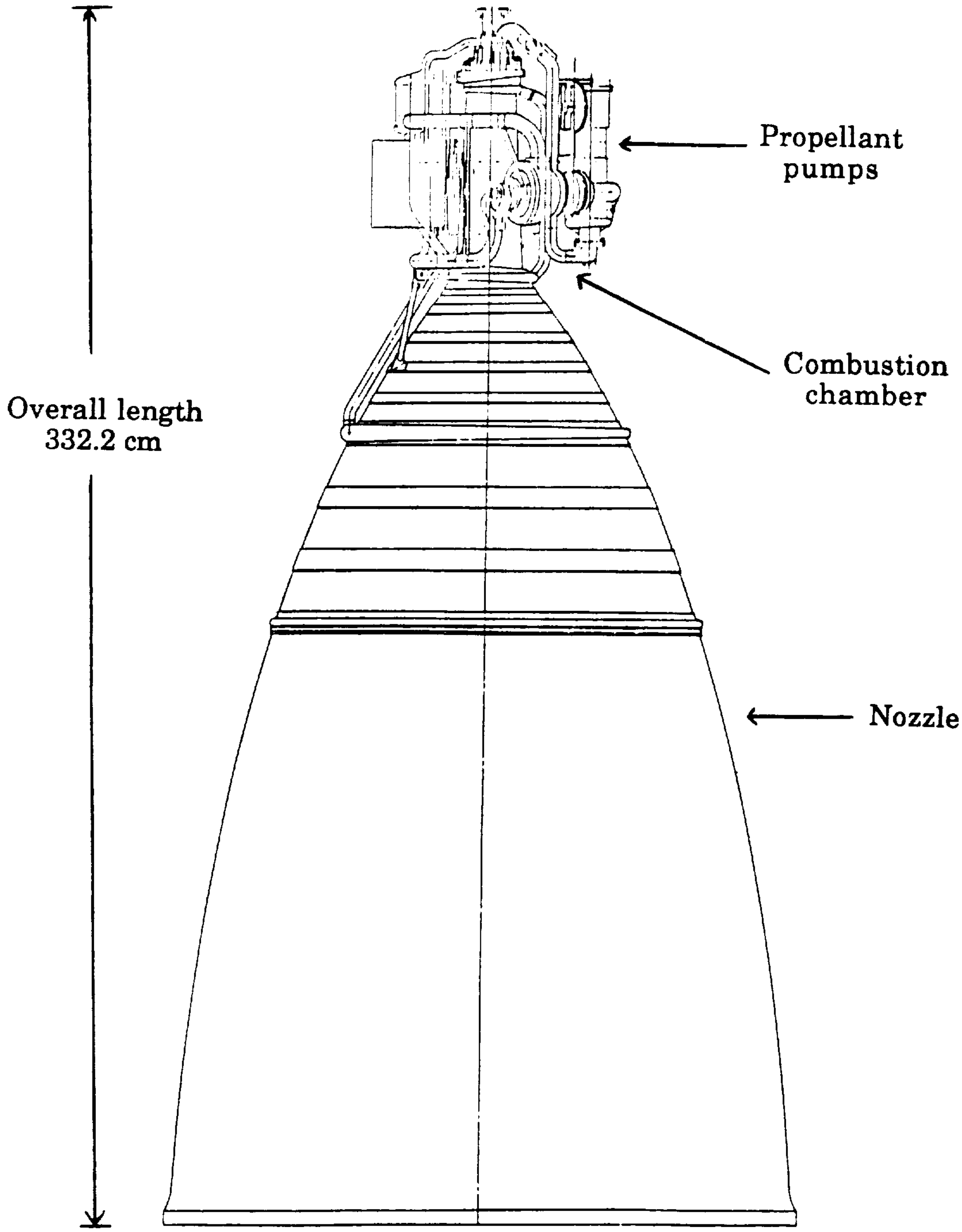


Figure 5.1 Rocketdyne RS-44 engine

This engine has two additional features, a gimballing capability and a retractable nozzle, which make it attractive for use on the STV. The gimballing capability of plus or minus six degrees allows the thrust vector to be changed as needed. As the center of mass of the vehicle changes during the mission due to payload changes and propellant usage, the thrust vector can be changed so that it is always directed through the center of mass.

The configuration of the engine nozzle will differ in the all-propulsive and the aeroassisted missions. On the aeroassisted STV, the retractable nozzle is necessary because of limited space behind the protection of the aerobrake. However, this feature is not necessary on the all-propulsive design and will be not be used. By eliminating the retraction system, the overall mass of the all-propulsive vehicle is reduced.

To optimize the safety/cost trade off, two engines will be used. This configuration will provide a reasonable safety margin while keeping production and maintenance costs as low as possible.

5.2 Fuel Tank Design

5.2.1 Tank Shape

The best shape for propellant tanks on the space transfer vehicle is the prolate ellipsoid. The nearly spherical prolate ellipse combines reliable orientation of propellant with efficient use of volume.

The optimal shape for any fuel tank is a sphere, since the sphere will enclose the largest volume of fuel for its given dimension (surface area). Therefore, the fuel tanks on the space transfer vehicle were chosen to be as close to spherical as possible.

As shown in Figure 5.2, fuel tends to stick randomly to the inside of the spherical tank in a zero gravity environment. There is no one place to hook up the fuel lines where a supply of fuel is guaranteed.

In the case of the prolate ellipsoid, however, fuel always orients itself in the small, highly curved ends of the tank due to surface tension as pictured in Figure 5.3 [ref. 5.3]. So, in zero gravity, placing fuel lines at one end of the ellipsoidal tank will guarantee a supply of fuel for the engines.

Another advantage of ellipsoidal tanks is realized during vehicle maneuvering. When a vehicle changes directions, fuel in its tanks tends to slosh in the opposite direction. If the fuel tank has an ellipsoid shape, as opposed to a spherical shape, its steeper walls will not allow the fuel to slosh as far away from the engine feed line outlets. To further inhibit fuel slosh, four baffle plates will be positioned inside each tank.

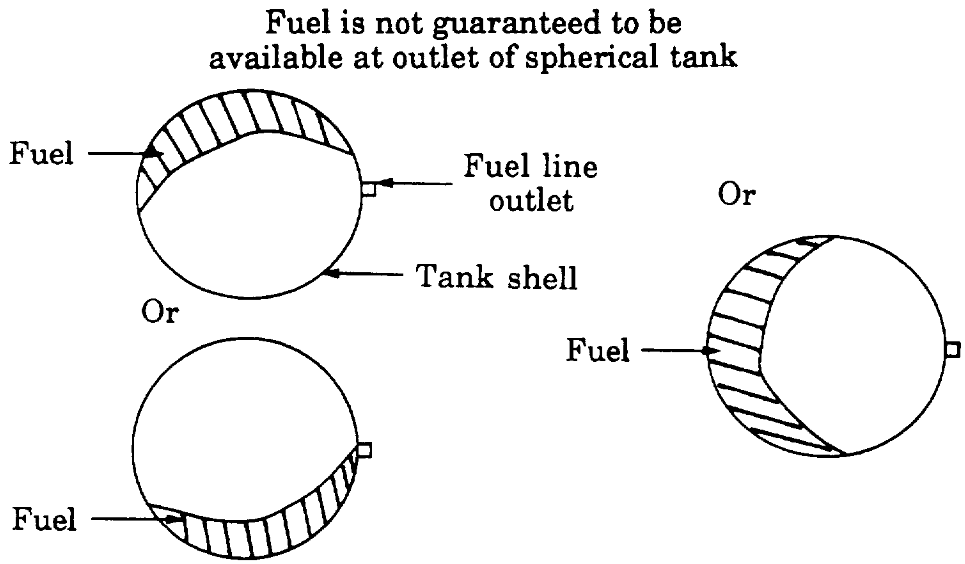


Figure 5.2 Zero gravity orientation of fuel in spherical tanks

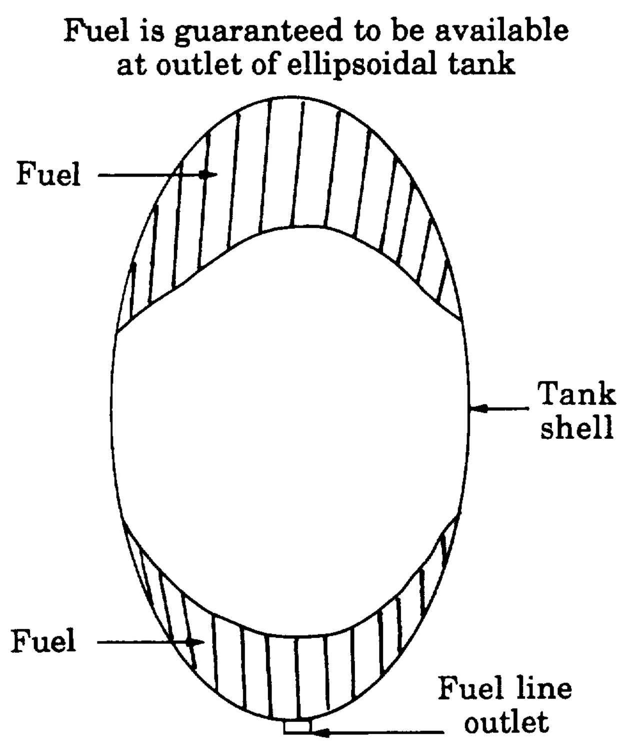


Figure 5.3 Zero gravity orientation of fuel in ellipsoidal tanks

To maintain a constant pressure within the tanks, a small amount of propellant will be bled from the high pressure pumps of the main engines. This ensures a constant back pressure of 35 kPa in the tanks which will keep the fuel flowing. To prevent internal tank pressure from becoming too high, each tank will be fitted with a boiloff vent.

5.2.2 Tank Shell Material

Keeping structural mass of the space transfer vehicle to a minimum will save fuel required to move the vehicle. For this reason, the best material to construct the walls of the fuel tanks is a lightweight and strong aluminum-lithium alloy.

Below is a table of lightest aluminum alloys, with their associated strengths (E) and densities (d).

Alloy	E (kPa)	d (kg/m3)
Al-Li 2090	704	2676
Al 2219-T62	640	2967
Al 2219-T63	640	2967
Al 2219-T65	640	2967

Table 5.4

It is clear from the table that the alloy Al-Li 2090, which is made by the Alcoa Corporation, provides the most strength (highest E) at the best weight (lowest d). The other alloys, while more ductile, are weaker and heavier overall.

5.2.3 Tank Insulation

Insulation for the tanks must protect the stored fuel from any source of heat and be lightweight to conserve fuel. Dacron Tuft (Superfloc) insulation is recommended to insulate the fuel tanks because it is thermally efficient, lightweight and the least costly of all reasonable methods.

Sources of heat include solar radiation, the engines and the crew compartment of the ship (the Spacecraft Configuration and Integration group has designed the engines and crew module to be close to the fuel tanks).

Insulation for spacecraft is traditionally characterized by a parameter called the "d*k" coefficient. This parameter is the insulator's density (d), multiplied by its thermal conduction coefficient (k). The best insulators for use on the fuel tanks will have the lowest product of these two factors. Dacron Tuft has the lowest density for its d*k coefficient of any currently available insulators.

A wrapping of 120 layers of this insulation will effectively shield the fuel tanks from heat. Dacron Tuft (Superfloc) insulation consists of a layer of heat resistant dacron polyester tuft floc, sandwiched between 0.025 mm thick sheets of radiation reflecting aluminum (see Figure 5.4). The 120 layers of this insulation will have a total thickness of 8.75 cm. It is manufactured by the TRW Corporation.

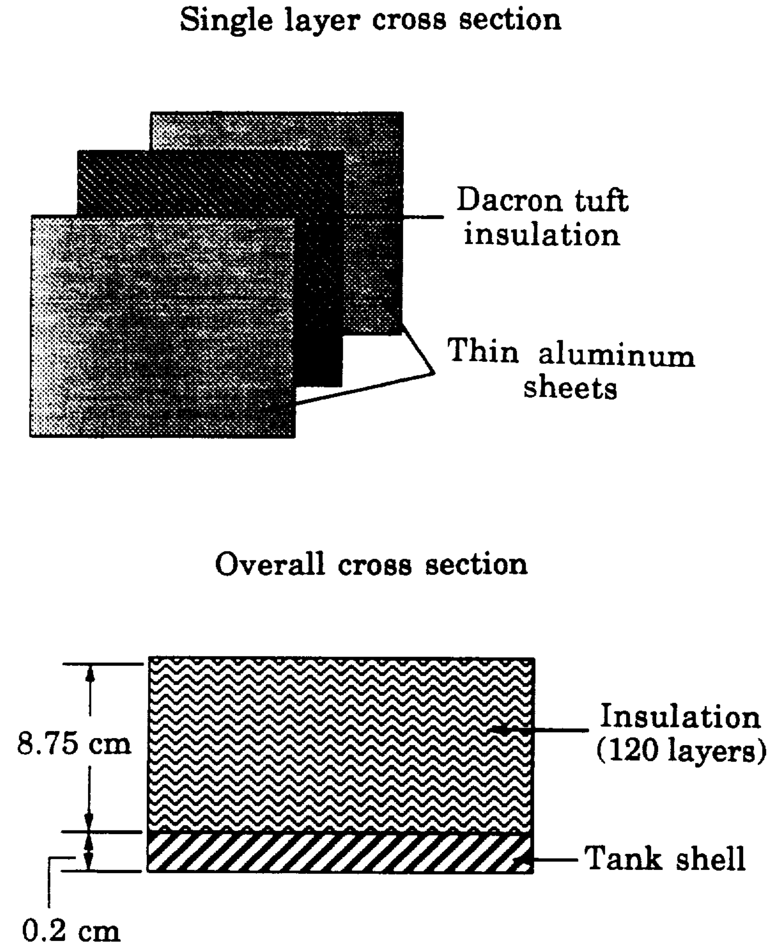


Figure 5.4 Dacron tuft fuel tank insulation diagrams

5.2.4 Tank Specifications

CSTV, 4 tanks, all ellipsoidal

Hydrogen tanks (2)

volume	83 m^3 each
major axis length	8 m
minor axis length	4.5 m
shell mass	62 kg each
baffle mass	10 kg per tank
insulation mass	221 kg each
thickness	0.2 cm

Oxygen tanks (2)

volume	31 m^3 each
major axis length	6.5 m
minor axis length	3 m
shell mass	32 kg each
baffle mass	8 kg per tank
insulation mass	114 kg each
thickness	0.2 cm

Total system mass.................................. 894 kg

ASTV, 4 tanks, all ellipsoidal

Hydrogen tanks (2)

volume	65 m^3 each
major axis length	11 m
minor axis length	3.4 m
shell mass	42 kg each
baffle mass	10 kg per tank
insulation mass	149 kg each
thickness	0.2 cm

Oxygen tanks (2)

volume	24 m^3 each
major axis length	7 m
minor axis length	2.6 m
shell mass	27 kg each
baffle mass	8 kg per tank
insulation mass	96 kg each
thickness	0.2 cm

Total system mass.................................. 664 kg

5.3 Attitude Control System

The attitude control system is responsible for small course corrections and maneuvers near satellites and the Space Station. The system must provide a total ΔV of 200 m/sec for all translational and rotational motions and complete control about the roll, pitch, and yaw axes.

5.3.1 Comparison of Possible Attitude Control Systems

Table 5.5 lists possible attitude control systems with advantages and disadvantages of each system. [ref 5.4]

System	Advantages	Disadvantages
Compressed gas	Non-corrosive exhaust	Low thrust High system mass
Liquid fuel	High thrust and Isp Low system mass	Corrosive exhaust
Resistojets	High Isp	Low thrust Greater power needed
Momentum wheels	Precise control	High mass Thrusters needed to despin

Table 5.5

5.3.2 Choice of Attitude Control System

For the STV, a liquid propellant system was chosen for attitude control maneuvers. The liquid propellant system will give the high thrust needed for orbital maneuvers and necessary corrections during aerobraking but care will be needed for maneuvers near satellites and the Space Station due to the corrosive nature of the exhaust from this system. A compressible gas system with its non-corrosive exhaust would be desirable for these maneuvers but, due to mass considerations, this type of system could not be used.

Since a liquid propellant system has been chosen, the best type of fuel must be determined. Available choices are monomethyl hydrazine-nitrogen tetroxide (MMH-NTO) or liquid oxygen/liquid hydrogen (LOX/LH2). MMH-NTO, being non-cryogenic, is much easier to store than LOX/LH2 but gives lower thrust for a given mass flow rate. With the STV main engines using LOX/LH2, a second system of fuel tanks can be eliminated if the liquid propellant thrusters use the same fuel. Greater thrust and the fact that an LOX/LH2 system is already on board makes LOX/LH2 thrusters the optimal choice.

The thruster chosen for the attitude control system is the Aerojet AJ10-167 [ref 5.5]. See Figure 5.5. A list of parameters for this thruster is given in Table 5.6.

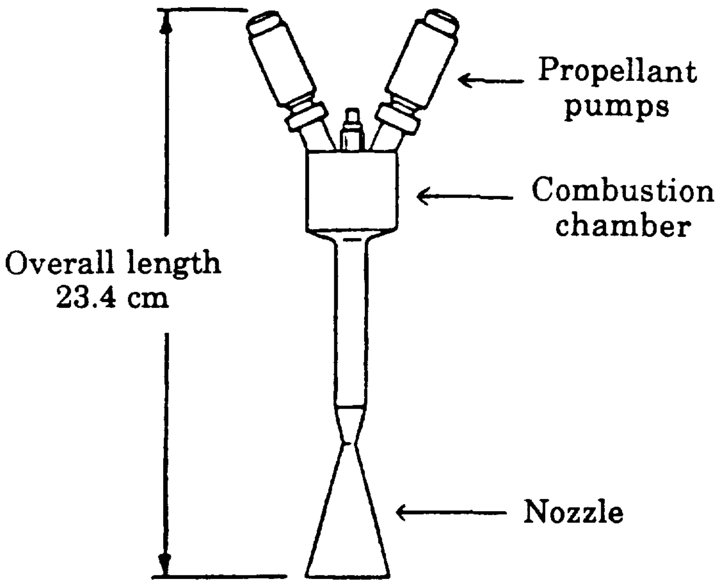


Figure 5.5 Aerojet AJ10-167 thruster

Thrust....................................	111 N
Propellant...............................	LOX/LH2
Mixture ratio..........................	3.0
Isp..	400 sec
Propellant flow rate................	0.284 kg/sec
Dry mass...............................	1.72 kg
Power....................................	1.4 watts

Table 5.6

Propellant for the attitude control thrusters will be stored in the same tanks as the main engines. As these thrusters do not include pumps, additional pumps will have to be included for the attitude control system. For a total ΔV of 200 m/sec, the following propellant quantities will be needed.

	Mass	Volume
LH2 (fuel)	575 kg	8.2 m^3
LOX (oxidizer)	1725 kg	1.5 m^3

The attitude control system for the CSTV and the ASTV will consist of 32 thrusters, propellant pumps for these thrusters, and accumulator tanks for both liquid oxygen and liquid hydrogen. Propellant will be drawn from the main tanks into the accumulator tanks where it will then be pumped to the thrusters. Accumulator tanks are used to reduce the number of fuel line connections which must be made to the main tanks.

The following table lists the mass breakdown for the attitude control system.

Thrusters	55 kg
Pumps	70 kg
Plumbing	20 kg
LH2 accumulator tank	25 kg
LOX accumulator tank	5 kg
Total system dry mass	175 kg

Table 5.7

5.4 Overall Propulsion System Configuration

To allow easier understanding of the configuration diagrams, the attitude control system will be considered separately.

5.4.1 CSTV

The main propulsion system of the CSTV consists of two liquid hydrogen tanks, two liquid oxygen tanks, and two main engines. Each engine will be connected to each propellant tank, and both LH2 tanks will be connected as will both LOX tanks. This allows the tanks to drain evenly so that the center of mass of the vehicle remains along the centerline. The connecting lines are valved at each end in a way that ensures valve failure will not cause propellant leakage. Each line contains one fail open and one fail closed valve. A fail open valve is designed so that if it fails, it will fail in an open position. Likewise, a fail closed valve is designed so that if it fails, it will fail in a closed position. If a fail open valve fails, flow through the line can still be controlled with the fail closed valve. If the fail closed valve fails, flow is cutoff and another line must be used. This ensures that if both valves fail, propellant will not leak out of the system. Figure 5.6 shows the location of the main engines, tanks, propellant lines, and valves for the CSTV.

The attitude control system for the CSTV will consist of 4 attitude control modules each containing 8 thrusters. In each module, Thrusters will point in 4 directions. To provide redundancy, thrusters will be paired so that 2 point in each direction. Thrusters at these locations will provide complete translational and rotational control about all three axes. Figure 5.7 shows thruster location and thrust vectors for the CSTV.

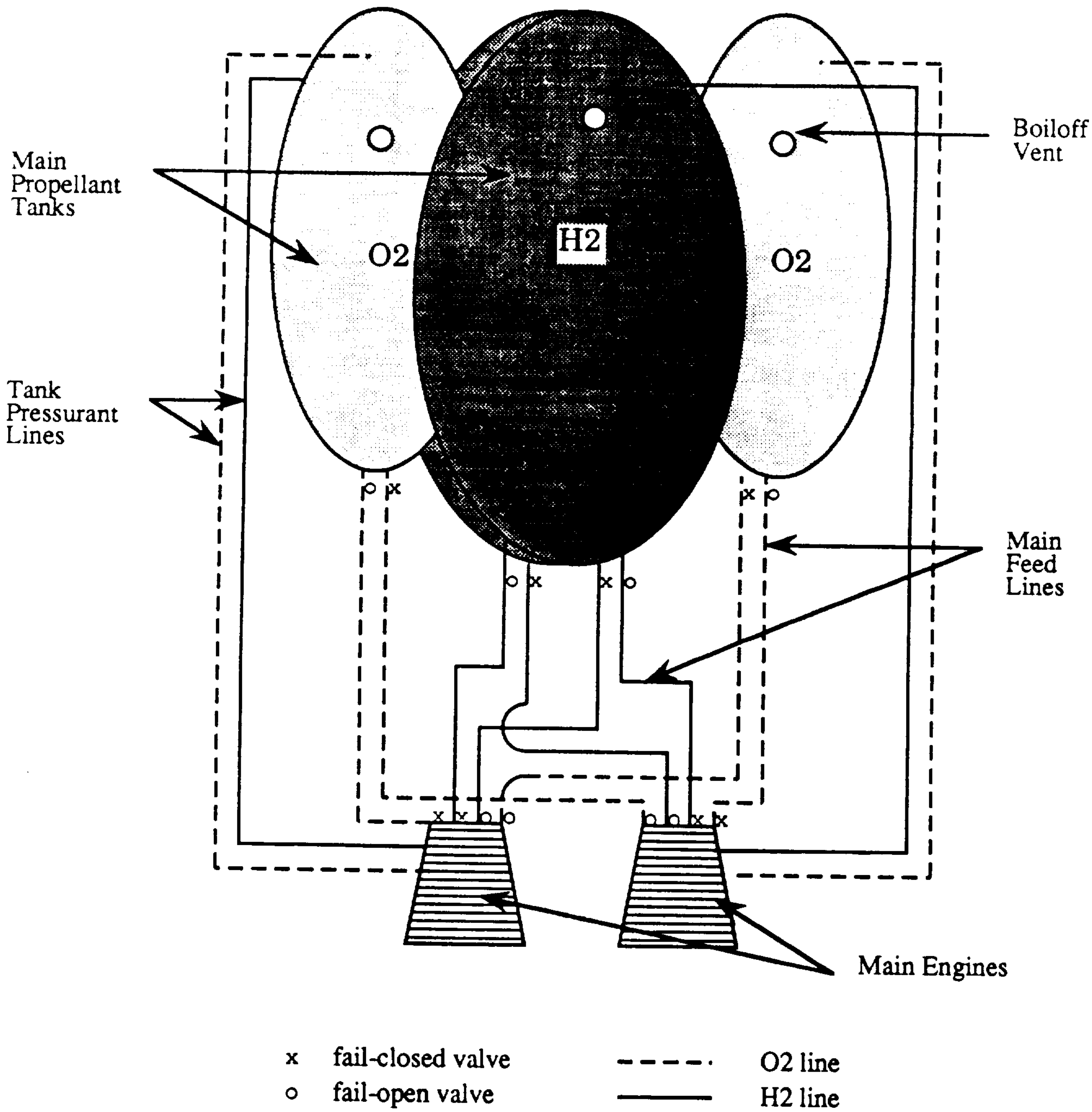


Figure 5.6 CSTV main propulsion system configuration

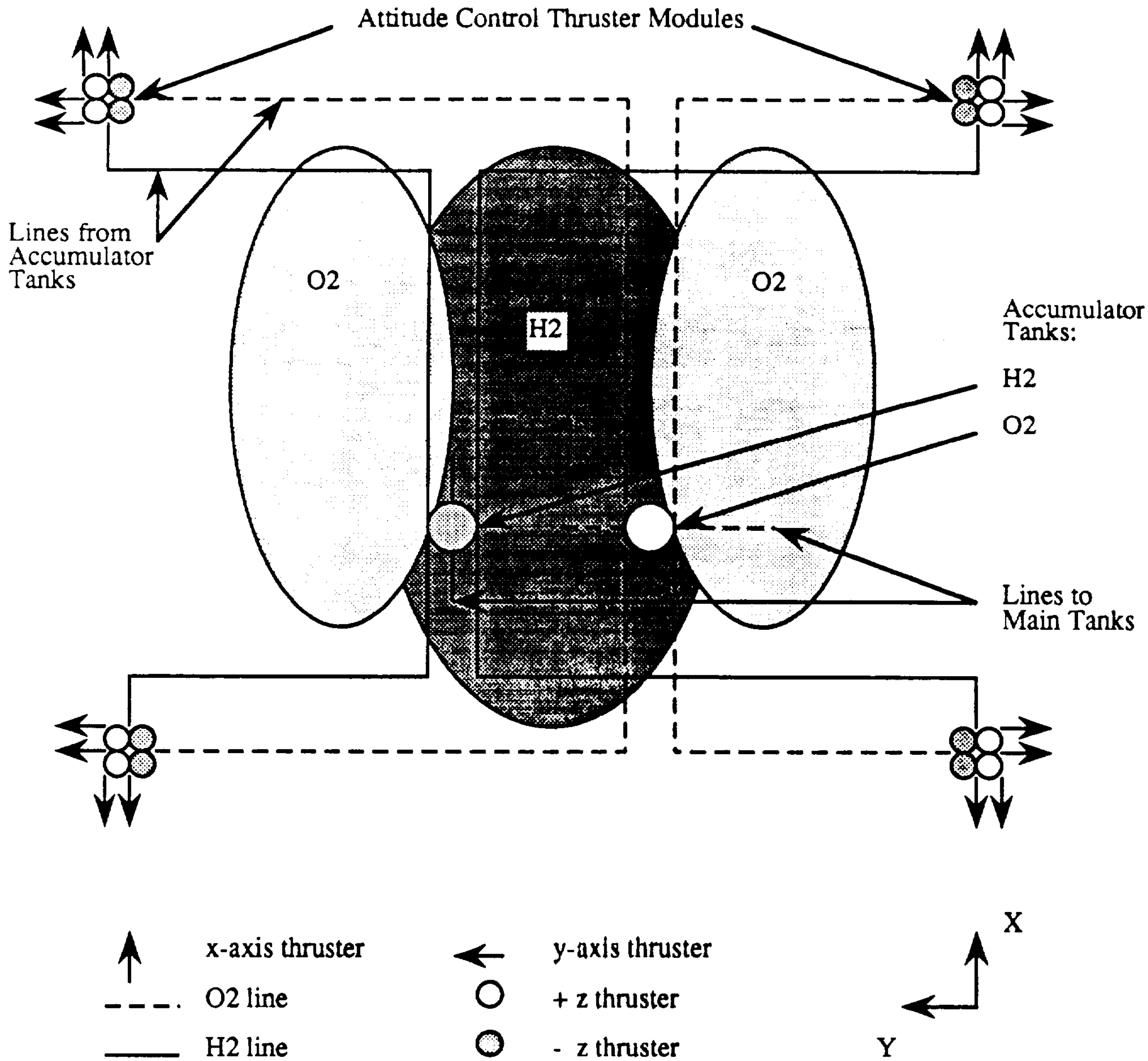


Figure 5.7 CSTV thruster location and thrust vectors

5.4.2 ASTV

The main propulsion system on the ASTV is similar to that of the CSTV in that it contains the same components. Notable exceptions are longer propellant tanks to keep the center of gravity low in the aerobrake, and retractable main engine nozzles. A similar fail open/fail closed valve system will be used to prevent leakage. Figure 5.8 shows the main propulsion system configuration for the ASTV.

The attitude control system for the ASTV is similar to that of the CSTV in that it will consist of 4 attitude control modules each containing 8 thrusters. Again, to provide redundancy, thrusters will be paired so that two thrusters point in each of 4 directions in each module. These modules will be located 90 degrees apart along the edge of the heat shield. Since thrusters cannot fire through the heat shield, some thrusters will have to be angled so that the thrust vector has a downward component. Although this method is somewhat inefficient, it is the only way to provide upward vertical motion. See Figure 5.9 for location of thrusters and thrust vectors. Thrusters are located to provide complete translational and rotational control about all three axes.

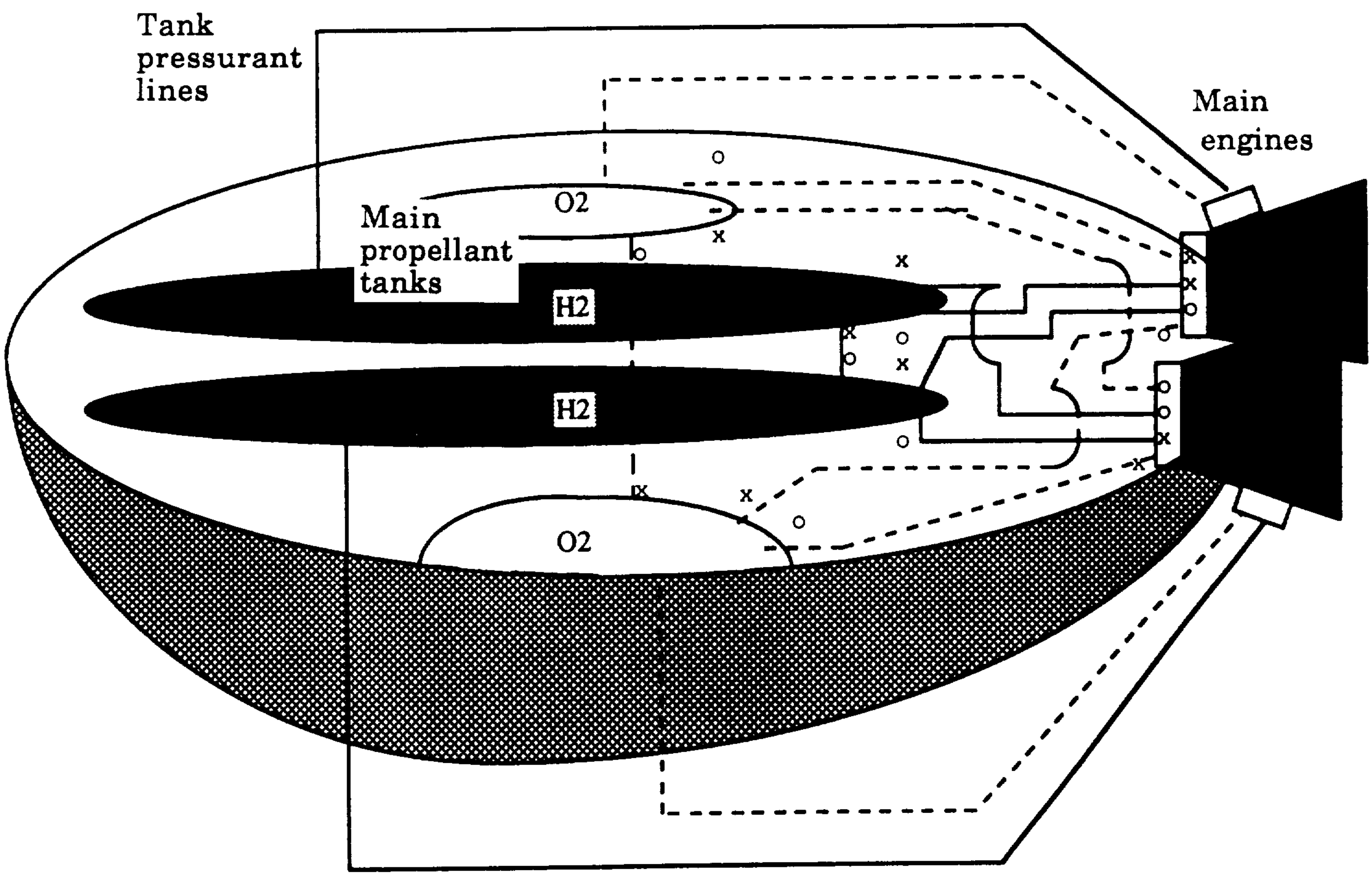


Figure 5.8 ASTV main propulsion system configuration

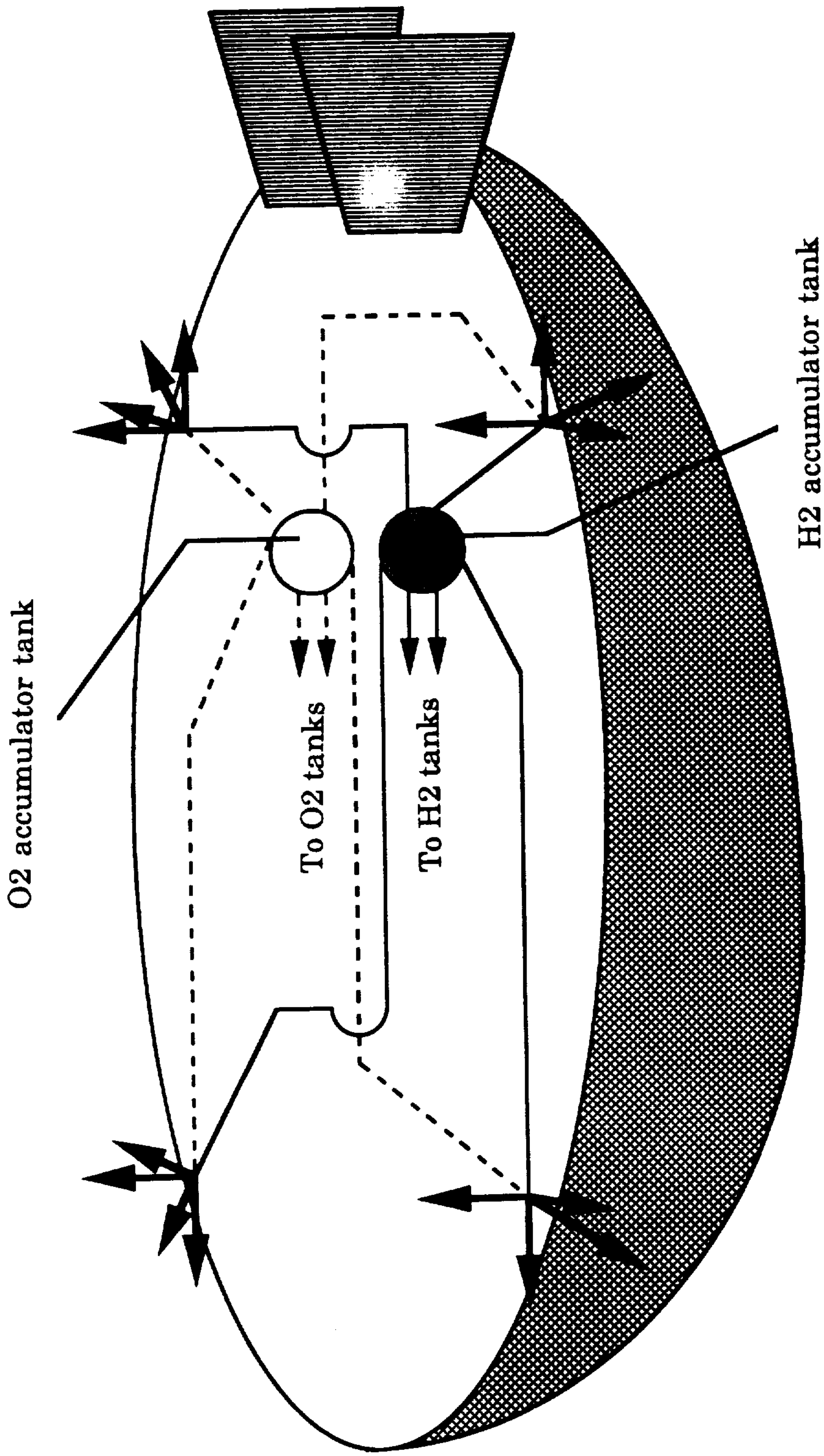


Figure 5.9 ASTV thruster location and thrust vectors

5.5 Mission Specifications

5.5.1 Fuel Requirement for Nominal Mission

Fuel requirements for the STV were obtained using the following equation:

$$M = \{\exp [(\Delta V / (Isp \times g)] - 1\} \times (D + P)$$

where M is the mass of fuel required
ΔV is the change in velocity
Isp is the specific impulse
g is earth's gravitational acceleration
D is the dry mass of the vehicle
P is the payload mass

The nominal missions are:

All-Propulsive:

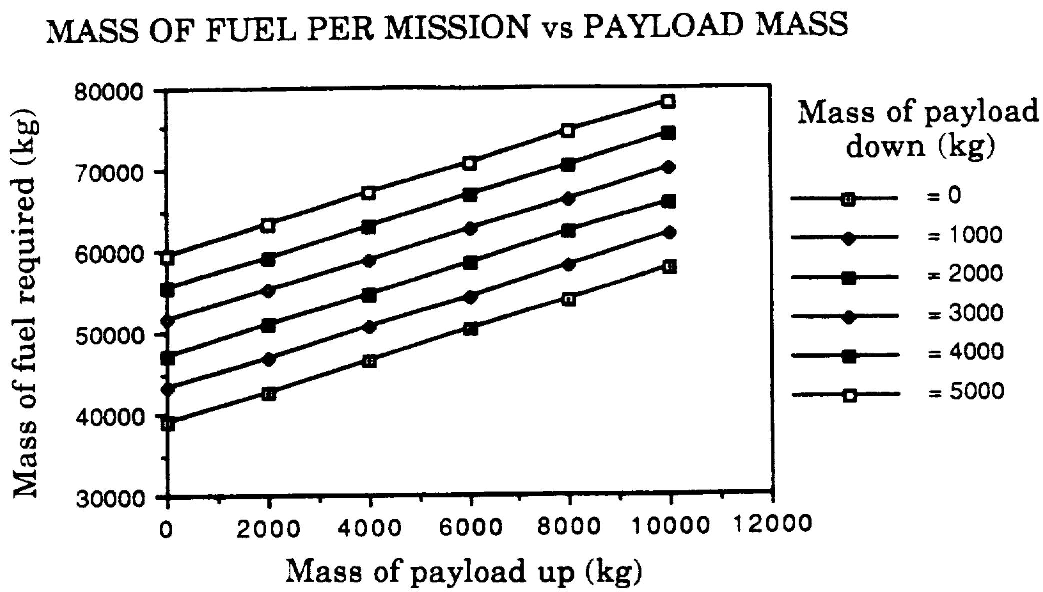
1) Total ΔV = 9395 m/s
2) Isp = 492 sec
3) Dry mass = 6341 kg
4) Payload mass is 10,000 kg up and 5,000 kg down
5) Mass of fuel required = 77,925 kg

Aeroassisted:

1) Total ΔV = 7354 m/s
2) Isp = 492 sec
3) Dry mass = 7841 kg
4) Payload mass is 10,000 kg up and 5,000 kg down
5) Mass of fuel required = 56,280 kg

Figure 5.10 shows graphs of propellant mass requirements for various mission specifications for both vehicles.

CSTV (Dry mass = 6341 kg)



ASTV (Dry mass = 7841 kg)

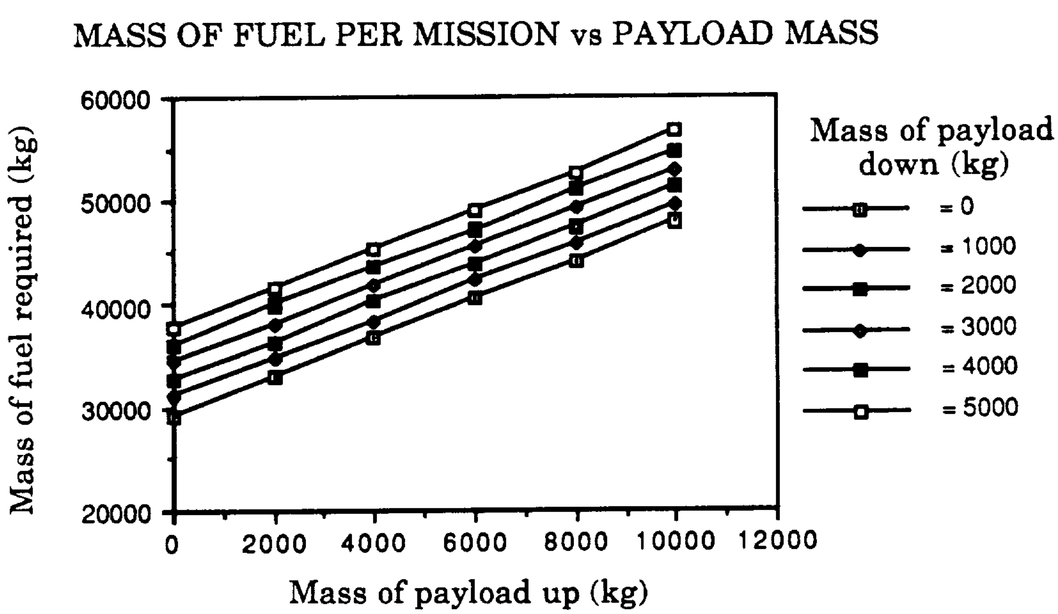


Figure 5.10 Graphs of propellant mass requirements

5.5.2 Engine Burn Times

The engine burn times were obtained from the following equation:

$$tb = M \times g \times Isp / T$$

where
- tb is the burn time required
- M is the mass of fuel
- g is earth's gravitational acceleration
- Isp is the specific impulse
- T is total engine thrust

For the nominal all-propulsive mission the total burn time is 47 minutes and there are seven startups, which corresponds to an engine lifetime of 25 missions.

For the nominal aeroassisted mission the total burn time is 34 minutes and there are nine startups, which corresponds to an engine lifetime of 35 missions.

5.6 References

5.1 *Jane's All the Worlds Aircraft*; John W. R. Taylor, editor; Jane's Publishing, Inc.; New York, New York, 1987

5.2 *Rocketdyne's Engine for the Orbital Transfer Vehicle*; Lasers and Advanced Engineering, Canoga Park, California; Rocketdyne Division, Rockwell International; February, 1984

5.3 General Dynamics, OTV Fuel Tank Study, Volume II, Book 3

5.4 Martin Marietta, RCS Study, Volume I

5.5 *Liquid Rocket Engines*; Aerojet Liquid Rocket Co.; Sacramento, California; 1975

Chapter 6

Power and Communications

6.0 Summary

The systems described in this chapter are identical for both the aeroassisted and all-propulsive versions of *Argo* in terms of design and mass. Differences in location of the antennas and radiators are described in Chapter 2 - Spacecraft Configuration and Integration.

Power Generation System

The STV's power requirements are 3.70 kW nominally, 5.70 kW maximum, and 1.90 kW for life support only. The total energy required is 1235 kW-hr (including a 4-day emergency life support). The power system is a hydrogen-oxygen chemical fuel cell system with three fuel cells and supply systems. Only one system is in operation at any time and is required for safe return to LEO and Space Station *Freedom*. Each fuel cell system has a mass of 95 kg and a volume of 0.13 m^3. The fuel is stored in the auxiliary fuel tanks which contains 419 kg liquid oxygen and 52.4 kg liquid hydrogen. Water is generated as a by-product and goes directly to the life support system. The fuel cell systems are located in the command module. Heat generated by the fuel cells and other equipment in the command module will be removed by cold plates through a 30 kg water cooling system. The heat is then transferred to a freon gas cooling loop and radiated into space by 21.3 m^2 of radiator surface area on the outside of the STV.

Power Management System

The power system must be managed efficiently so that a minimum of excess energy is produced. The astronauts enter into the central computer the times of their power intensive activities. The power system provides only the amount of power necessary to fulfill these requirements by controlling the fuel flow rate to the fuel cells. Regulators maintain proper voltage by expending excess power in the form of heat. They are cooled by the fuel cell cooling system. Direct current transformers provide power at voltages different from the supply voltage of 150 V to specific users with those needs. Direct current circuit breakers are located at each user to protect the system from malfunctioning loads. Direct current switches can separate any user from any of the three redundant power buses. Failures are detected by monitoring circuit breakers and power bus ammeters. Failures are automatically isolated with the switches and reported to the astronauts.

Power Distribution System

The power generated by the fuel cell system requires a distribution system with minimal power loss and mass. This design criteria is met with the following system. The two major groups of power users, the command module and propulsion groups, are supplied with power with a radial system architecture. The individual components within each group are supplied with a ring system. The power is transmitted at 150 V in direct current with a bipolar link. This link is an aluminum coaxial cable with an cross-sectional area of 1 cm^2 and a flat rectangular shape. Approximately 21 meters of cable is required for one bus. For redundancy, there are three identical, independent power buses. This results in a total distribution system mass of 18 kg. The maximum power loss in the distribution system is 5.1 W, with the maximum concentration occurring at 0.8 W/m. No special cooling system is required to mitigate the cable heating.

Communications

The system will use K-band communications for primary data transfer in combination with the Tracking and Data Relay Satellite System. There is an omni-directional setup for communications with detached payloads and targets in the immediate vicinity of the STV consisting of three 0.50 m diameter coil antennas, two for K-band and one for S-band. Two 0.75 m diameter K-band dish antennas will handle the long range communications. The system will account for and rectify problems concerning Doppler effects on transmissions and signal reception. This system will also track possible targets with radar, utilizing the same equipment as the long range K-band system. This will provide telemetry information to support rendezvous.

Data Management System

The data management system will coordinate the data from a number of major subsystems into the central computer core. Each subsystem will be controlled by its own data processor. These processors provide the main computers with the data that runs the entire STV. Each processor is linked to a network interface unit which connects the subsystems to one another and to the main computers via a fiber optic ring system. Data will be stored on erasable optical disks, and workstations will interface the crew with the STV through the central computers. The software of the STV will include expert systems in order to ease the workload of the astronauts.

Guidance

Guidance is the control of the spacecraft to specific attitude and velocity constraints. While in space, the crew enters the desired state of the spacecraft in space or target parameters for the atmospheric flight. The guidance system calculates the necessary trajectories with fuel optimal maneuvers and commands jets to fire. The system calculates the spacecraft state using information from the Inertial Measuring Units (IMUs). Furthermore, the system has a mass estimator that uses data regarding maneuvers and their resulting effects on the spacecraft state to calculate total mass, center of mass location, and moments of inertia.

Navigation

Navigation is the determination of the spacecraft position, velocity, attitude, and time. The position, velocity and time of the spacecraft are determined by the Global Positioning System. The attitude of the spacecraft is determined by three 3 kg star trackers on board. Knowledge of these quantities is maintained by three 15 kg IMUs, each of which contain three laser rate gyroscopes and three accelerometers. The navigation quantities will be updated before each major burn and the atmospheric flight segment. The velocity uncertainty for the atmospheric flight in this system is ± 0.06 m/sec; position uncertainty is ± 18.5 m; attitude uncertainty is ± 0.03 degrees.

System Mass Chart

SYSTEM	MASS (kg)	VOLUME (m^3)	LOCATION
Power	**904.6**		
Fuel Cells x 3	95.0 x 3	0.13 (each)	Command Module
Radiator	100.0	0.64	Outside STV
Distribution System	18.0	-	Entire STV
LO2	419.2	-	-
LH2	52.4	-	-
Internal Cooling	30.0	-	Command Module
Communication	**55.0**		
Antennas	40.0	0.35	Outside STV
Tranceivers x 2	7.5 x 2	0.13 (each)	Command Module
Data Management	**76.0**		
Processors x 6	1.0 x 6	0.01 (each)	Each Subsystem
Storage Disk Drives	10.0	0.03	Command Module
NIUs x 6	5.0 x 6	0.03 (each)	Each Subsystem
Workstations (total)	30.0	0.50	Command Module
Guidance	**55.0**		
IMUs x 3	15.0 x 3	0.02 (each)	Command Module
Star Trackers x 3	3.0 x 3	0.06 (each)	Outside STV
Total Mass = 1090.6 kg			

Table 6.1

6.1 Power

Designing the total power system for the STV involved the following requirements: (1) the power and energy requirements of the STV were determined, (2) a power system was designed to satisfy those requirements, and (3) a system was developed to manage and distribute the electrical power to the STV components as required.

6.1.1 Power Requirements

The power requirements for the STV can be seen in Table 6.2, listed as per system and subsystem. The energy requirements and comments on usage are also listed. The maximum power, which is required when the STV is at geosynchronous orbit and is operating the remote manipulator arm, is 5.70 kW. The nominal power required for life support and safe return to low earth orbit and Space Station *Freedom* is 3.70 kW. For life support only, the power requirement is 1.90 kW.

6.1.2 Power Systems Examined

Nuclear

One of the power generation systems considered was a nuclear system. It has the advantages of long life and total self-containment. Therefore, it does not require constant consumption or replacement of bulky fuels. Unfortunately, the system has the major disadvantage of emitting a hazardous level of radiation. Therefore, it requires shielding to protect crew, equipment and payload. The shielding must be all encompassing and extensive since the STV will operate in close proximity to the Space Station, satellites, and vehicles which may contain personnel and highly sensitive equipment. This translates directly to an exceptionally massive shield. Preliminary estimates to bring the radiation down to a tolerable level required a 2000 kg [ref. 6.1] shield, which is prohibitively heavy. Another danger was for the aeroassisted version of the STV. The atmospheric portion of the trip generates extreme heat, which could result in extensive shielding damage. In the case of catastrophic failure, high levels of radiation could be released in the atmosphere. Therefore, a nuclear power source was discarded as a possible power source for the STV.

Solar Dynamic

Solar dynamic power generation was also studied as a possible power source for the STV. This system concentrates the sun's light on a receiver, thus creating heat that drives an engine (i.e., Rankine, Brayton, free piston Stirling, etc.). Solar dynamic systems are small, efficient, and can be run during eclipse time due to heat storage in the receiver. These are all advantages over the solar photovoltaic system. It was decided not to use solar dynamic power generation for three reasons: (1) weight, (2) incompatibility with the STV, and (3) limited technology. The weight of a solar dynamic system exceeds one thousand kilograms, much more than the STV can or should handle. Also, all systems to date supply a minimum of 25 kW [ref. 6.2] of power, which is much more than the STV needs, thus making it an impractical system. Finally, no dynamic system has ever been flown or

STV Power Requirements

SYSTEM	POWER (kW)	ENERGY (kW-hr)	COMMENT
Power	**0.35**	**58.8**	**Continuous**
Fuel Pumps	0.15	25.2	
Radiator Pumps	0.15	25.2	
Distribution Loss	0.05	8.4	
Communications	**1.00**	**168.0**	**Continuous**
Tranceivers	0.80	134.4	
Antenna Boom	0.20	33.6	
Data Management	**0.60**	**100.8**	**Continuous**
Computers	0.30	50.4	
NIUs	0.10	16.8	
SDPs	0.10	16.8	
Work Stations	0.10	16.8	
Guidance	**0.18**	**25.6**	
IMUs	0.15	25.2	Continuous
Star Tracker	0.03	0.4	≈ 12 Hours
Life Support	**3.55**	**572.4**	
Atmosphere	0.225	37.8	Continuous
Cabin Structure	1.325	222.6	Continuous
RMS	2.000	312.0	GEO only
Propulsion	**1.30**	**118.2**	
Engine	0.28	0.3	≈ 1 hour
Warm-up valves	0.30	0.3	≈ 1 hour
Ready valves	0.70	117.6	Continuous
Attitude Control	**0.05**	**8.4**	**Continuous**
7-Day Mission Energy 1052.2			
4-Day Emergency Life Support + 182.4			
Total Energy Required 1234.6			

Table 6.2

tested in space. Therefore, due to the infancy of the technology and the other reasons listed above, the solar dynamic power system was ruled out as a method of power generation for the STV.

Solar Photovoltaic

The third type of power system studied for the STV was solar photovoltaic. This system directly converts the sun's energy into direct current electricity. Solar photovoltaic is a well-developed technology which has both low mass and the ability to supply power without using consumable fuel. However, there are three major disadvantages: (1) loss of power in eclipse, (2) radiation degradation, and (3) problems with pointing and tracking the array. Solar photovoltaic arrays only provide power when they are exposed to sunlight. Therefore, the arrays cannot provide power during the period of time that the STV is in the Earth's shadow. During these times a secondary power source consisting of batteries or fuel cells is used. This secondary power source substantially increases the total mass of the power system. The second problem is that the large amounts of radiation in orbit slowly degrade the solar cell's efficiency by as much as 10% a year [ref. 6.3]. The third disadvantage is that the solar array must always be kept pointing to within 3-4 degrees of the sun. This constraint could potentially limit the maneuverability of the STV. Therefore, due to these unanswered problems, solar photovoltaic was discarded as a possible power source for the STV.

Chemical (Fuel Cell)

A fuel cell system was decided upon as the power system for the STV. This was chosen for several reasons: (1) it is mass competitive for the length of mission and power requirements of the STV, (2) it is a readily available technology and is highly reliable (including no need for deployment, which greatly decreases the chance of breakdown), (3) it offers good redundancy, and (4) it provides water as a by-product which can be used by life support systems [ref. 6.4]. The disadvantages are that it needs fuel and heat management. However, the type of fuel cell was selected to be a hydrogen/oxygen type, so the fuel type is the same as for propulsion. Also, life support requires heat management, so the use of a fuel cell only requires an increase in the size of a radiator, rather than an entirely new system.

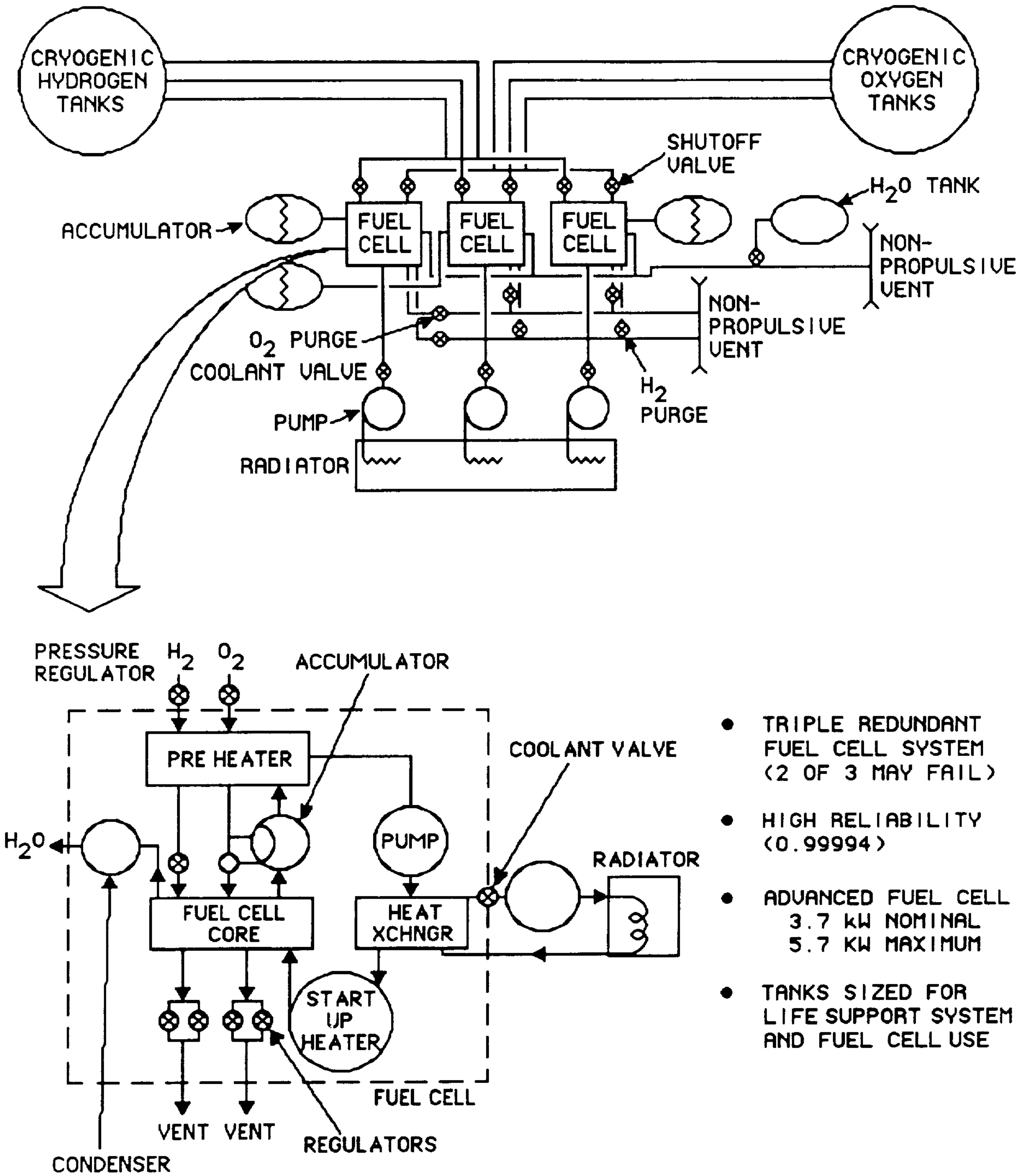
6.1.3 Fuel Cell System Specifications

Fuel Cells

The power system, as shown in Figure 6.1, consists of three identical fuel cells which are each capable of supplying maximum power requirements. This is for redundancy since it is a life critical component. All of the vital components, such as the pumps, piping, etc., are triple redundant so that a failure will not be catastrophic to the mission. Each fuel cell has a mass of 95 kg [ref. 6.5] and a volume of 0.13 m^3 (46 cm x 46 cm x 61 cm) [ref.

6.6]. These masses and volumes include piping, pumps, mountings, etc., as shown in Figure 6.1. Nominally, only one fuel cell and supply system are in use at a time.

Power Generation System (PGS)



- TRIPLE REDUNDANT FUEL CELL SYSTEM (2 OF 3 MAY FAIL)
- HIGH RELIABILITY (0.99994)
- ADVANCED FUEL CELL 3.7 kW NOMINAL 5.7 KW MAXIMUM
- TANKS SIZED FOR LIFE SUPPORT SYSTEM AND FUEL CELL USE

Figure 6.1 [ref. 6.7]

Fuel

The fuel required for the fuel cells is stored in the auxiliary tank system, which is shared with the life support system. There are three tanks each of liquid hydrogen and oxygen with a triple redundant supply system. If two tanks fail, the mission would be abandoned immediately, leaving sufficient fuel to maintain the necessary life support systems in the remaining tank. The masses for the tanks are included in the Spacecraft Configuration and Integration section of this report. The mass of fuel required for a nominal mission requiring 1235 kW-hr (for a 7-day mission plus 4-day emergency life support) is 419 kg of liquid oxygen and 52.4 kg of liquid hydrogen. The system produces 34 kg of water a day at its nominal power requirements. This water goes directly to the life support system.

Location in Command Module

The fuel cells are located in the temperature-controlled command module for several reasons: (1) a defective component can easily be replaced or repaired, (2) the fuel cell requires a temperature between 0° and 120° Celsius [ref. 6.4], (3) the water produced as a by-product can go directly to life support as required, (4) the fuel tanks will be near the command module, (5) most of the power is used by life support, data management, and other equipment which will also be located in the command module, and (6) the radiator is shared between life support and the power systems, so close proximity is desirable.

Thermal Control System

The fuel cells and other equipment in the command module generate 8.98 kW of heat. This heat is removed by cold plates through a 30 kg water cooling system. The heat is then transferred outside the command module to freon gas, which circulates through radiators to expend the heat into space. The radiators are 21.3 m^2 [ref. 6.8] in surface area and 3 cm thick. They are covered by louvres which rotate to reflect away any impinging sunlight. The total mass of the freon system and radiators is 100 kg [ref. 6.9]. For the aeroassisted version of *Argo*, the radiators will not be able to expend heat during the atmospheric portion of the flight. Instead, the heat will be stored in the water cooling system until the STV is in space again. The temperature of the command module during this segment will not exceed the limits imposed by the life support section of this report.

Power Generated

The power generated is in the form of direct current at 150 V, as required by the power distribution system.

6.1.4 Power Management

User Scheduling

To most efficiently use fuel resources and minimize excess power system heat production, the power produced should not exceed the power required. This is achieved by astronaut inputs of the times of their activities into the central computer. The central computer transmits this data to the power system processor. The fuel pumps control the fuel flow rate of the fuel cell to maintain power production equal to power requirements. The requirements should be known ahead of time because there is some lag time in changing the power level of the fuel cell system.

Regulators and Transformers

The power produced will always be slightly greater than power needs to provide a margin for a sufficient power supply. Furthermore, the power system will supply power at the maximum level expected over an interval of time, rather than at the exact level needed continuously. Therefore, it is necessary to expend the excess power using regulators. These are located at the distribution system origin on each power bus. They convert the excess power into heat which is removed with the water cooling system.

Some of the users may require power at a voltage different than that which is supplied by the distribution system. This need is expected to be required for a minority of the users. Therefore, direct current transformers will be located at each user that requires a specialized voltage. These will not need any special cooling system.

Circuit Breakers and Switches

It is possible for power users to malfunction and create a short in the distribution system. This would be catastrophic because all power would be lost through this short; consequently, life critical systems would not have sufficient power. To prevent this scenario, a direct current circuit breaker is located at each power user. These automatically separate the user from the distribution system if excess current flow occurs. A failure may also occur in any of the three redundant power buses that supply power to the users. If this should occur, it is necessary to separate that power bus from the distribution system. This is done using direct current switches located between each power bus and user, and between the fuel cells and power buses.

Failure Detection and Isolation

Failures in the users are located by constant monitoring of the status of the circuit breakers. If a circuit breaker should activate, its action would be reported to the astronauts and other systems so that either a repair could be made or a redundant replacement could be turned on. The condition of the power buses is monitored with ammeters and voltmeters located on each. If an irregularity is observed, the power bus is separated from the system with the direct current switches and the action is reported.

6.1.5 Power Distribution

Architecture

The power users can be separated into two groups by location - the command module group and the propulsion unit group. It is optimal to supply each with an independent radial system because less wire is needed. Conversely, it is optimal to supply power to the components within each group with one wire through a ring system.

Transmission Type and Voltage

The choice in transmission type was between direct and alternating current. Given a maximum allowable voltage, alternating current has a transmitting efficiency that is only 66% of direct current. Alternating current also creates more noise that can affect the STV electronics. Furthermore, it requires conversion from the power source type and then back to the user type, resulting in a loss in efficiency. Therefore, direct current will be used for transmission.

The power will be transmitted at 150 V. It is best to use the highest voltage possible since that will result in the lowest current to be distributed and consequently a lighter distribution system. This is because reducing current reduces cable size. The maximum allowable voltage on the STV is 150 V since the space environment begins to adversely affect the system at higher voltages [ref. 6.10].

Link Type and Description

A bipolar link which has two conductors, one positive and one negative, will be used. The danger of shock or electrocution from this system is minimal. The link used is a coaxial cable, which is a cable with one conductor within the other. Since the conductors produce magnetic fields that are identical in shape and equal in magnitude, the electromagnetic noise is effectively cancelled out. The conductors in the power cables will be made of aluminum since it is lighter for a given length and resistance than copper and silver. The mass of the cable is a function of the area, since the length is determined from the vehicle dimensions. The quantity Δpower/Δmass can easily be derived for the distribution system. For optimal mass, this quantity should equal that of the power generation system. This gives the mass of the triple redundant system as 18 kg, 10.5 kg in the command module and 7.5 kg from the command module to the propulsion unit. The cross-sectional area for each cable can be calculated from the mass, and it equals 1 cm^2 for this system. About half of the area is used for insulation.

Cable Shape and Length

The power cables will have a flat rectangular shape. It is desirable to have a shape that is easily mountable. Most of all, the cable surface area should be maximized to dissipate heat as fast as possible. These requirements lead to the selection of a flat rectangular shape instead of the usual round shape. The total cable length necessary to distribute the power is 21 meters per bus, 12 m in the command module and 9 m to the propulsion unit. This is a function of the geometry of the STV and the distribution architecture.

Power Loss

When maximum power is being used by the STV, the power loss in the distribution system is 5.1 W. The power generation system must produce this much additional power to compensate. The power loss is not equally distributed; therefore, there are places of maximum power loss. The losses at these points do not exceed 0.8 W/m, so no specific cooling system is required. The system will rely on simple heat transfer to the surrounding parts of the vehicle to dissipate the heat.

6.2 Communications

The communications system has many requirements. It must handle any type of information (video, sound, data, etc.) deemed necessary for the mission. The system must communicate with Earth and space-based systems, attached or detached payloads, and other possible targets. It must produce/receive a sufficient signal to noise ratio to ensure coherent signals to/from these target sights. It must account for and rectify problems concerning Doppler effects on transmissions. This system must also be integrated with the navigation system to track possible targets and support precise rendezvous.

6.2.1 Communications With Ground Systems

Ground-based systems consist of any system that requires communication directly with an Earth-based Antenna System (EAS). The advantage of this type of system is that it does not rely on an outside system, such as the Tracking and Data Relay Satellite System (TDRSS), for communications with an EAS. Unfortunately, there is an interference problem with the atmosphere at high frequencies, and even low frequencies are subject to interference during adverse weather conditions. At these lower frequencies, larger antennas are required. Therefore, the STV will not nominally support direct communication with an EAS.

6.2.2 Communications With Space Systems

Space-based systems consist of any system that has a Space-based Antenna System (SAS). The primary SASs in consideration are TDRSS and the Space Station. TDRSS can communicate with S-band, but is more versatile in the K-band range. It can send wide band information in this mode and has the capability of higher Bit Per Second (bps) send/receive rates. The Space Station will support space to space communications. However, at different orbits, direct communication with the Space Station will be impossible to maintain for extended periods. Therefore, the system will primarily utilize TDRSS for communications.

6.2.3 System Overview

S-band

It is useful to examine the frequencies and rates that are utilized individually. Various combinations of voice, video, telemetered data, etc., can be sent to or from the STV utilizing the S-band communications link. There will be the capability for two different bit transmission/reception rates. In the high bps mode, the STV can receive a 72 kbps digital data stream and transmit a 192 kbps digital data stream. In the low bps mode, the system can receive 32 kbps of data. The low bps transmission will consist of a 64 kbps data stream [ref. 6.11, 6.12]. Part of the signal could consist of telemetered information from a payload or other source which would require data relay.

Signal Type

The system will also be able to utilize a frequency modulated signal for transmission at bandwidths up to 4.5 MHz. This data can include: recorded voice, real-time closed-circuit TV, payload control signal, main engine data, and digital or wideband analog data from a payload (either attached, detached, or similar target).

K-band (radar)

The K-band communications system has the advantage of a frequency range within that of radar, thus allowing for the system to double as a radar tracking unit. If one K-band antenna is being used for communications, the second antenna system can be simultaneously used for radar. When in the radar mode, the system is capable of detecting, acquiring, and automatically tracking a passive target at a range of ~20 km, and an active target up to ~550 km. The tracking is effective down to a range of 30 m. These values are for objects on the order of 1 m^2 in size which have Swerling Case 1 scintillation characteristics [ref. 6.12], which are basically measurements of the radar reflectivity of the objects. In the proscribed range, the radar should acquire the target in a minute, or less, after a search along the expected target vector. Once a target has been acquired, the system should provide line of sight range to the target, range rate, angles relative to the STV rendezvous axis, and angle rates.

K-band (communications)

For the communications mode, the antenna and base will be the same as that for the radar system. As in the S-band system, there will also be two different modes for communication. In Mode 1, up to 52 megabits per second (Mbps) of information may be transferred from the STV. The information may come from an attached payload as operational data, stored data, experiment data, or real-time operational data. Mode 2 transmissions can consist of 4.2 MHz analog (TV) data from either the STV or an attached payload, or 7 Mbps of payload digital data, stored data, experiment data, or real-time operational data from the STV. For reception of information, the signal can carry 2 Mbps of medium bandwidth data with operational data for the STV or payload [ref. 6.12]. For all transmissions that are relayed with TDRSS, a triple convolution encoding process is used for signal identification and security purposes.

Continuous Transmission

Continuous transmission requires complex equipment since the STV will move with respect to the TDRSS satellites during many phases of its mission. As this occurs, there will be a lapse time where the antenna will have to switch communications from one satellite to another. In order to get around this "hand off time", another independent communication assembly is on board.

Global Positioning System

A final operational bandwidth is required for the Global Positioning System (GPS). This is a satellite system that consists of beacon satellites with known orbits and positions as functions of time. The position of the STV will be determined with respect to these satellites by receiving signals of their positions. Since this system plans on employing L-band transmission links, the STV must have an antenna to receive the signal at a sufficient ratio to calculate position. The K-band is sufficient to support the GPS system. Therefore, it will be utilized in this capacity. As can be seen in the system specifications above, the entire communications system, and position update system is completely redundant.

Transceiver

Now that the system itself is laid out, a transceiver choice should be made. Regardless of the choice, the transceiver will rectify Doppler effects caused by motion relative to the STV. The most important factor to consider is the power of the system. It is important that the transceiver be able to transmit with sufficient signal strength for necessary operations, while also being able to receive weak signals coherently. The trade-offs to consider are the mass of the system and the size of antennas required for the power level chosen. This leads to the selection of two 0.4 kW transceivers with a mass of 7.5 kg each. This results in a maximum transmitting power of 0.8 kW. The units are located in the command module.

6.2.4 Antennas

The system described above requires an overall antenna system to focus the power of the transceivers. For primary communications, the K-band frequency range will be used. This will require a set of antennas whose ranges overlap, since the nominal mission will require communications from LEO, GEO, and in transit. There may also be a requirement to relay data to or from a target or payload in the S-band range. All of the

Antenna Specifications

Antenna System	Band	Diameter (m)	Mass (kg)	STV Transmit (M Hz)	STV Receive (M Hz)
1	K	0.75 Dish	6.5	K band 15003.4 ±1.6	K band 13775 ±1.6
2	K	0.75 Dish	6.5	K band 15003.4 ±1.6	K band 13775 ±1.6
3	K	0.50 Omni	0.5	K band 15003.4 ±1.6	K band 13775 ±1.6
4	K	0.50 Omni	0.5	K band 15003.4 ±1.6	K band 13775 ±1.6
5	S	0.50 Omni	0.5	S band 2200 to 2300	S band 2020.0 to 2123.5
6	L	Shared Dish with Antenna System 1 or 2		L band Specified to GPS	L band Specified to GPS

Table 6.3 [ref. 6.11, 6.13]

communications will be done through TDRSS, and the system design assumes that TDRSS will be fully functional and have global coverage.

All of the above communications situations (i.e., ground via TDRSS, payload, etc.) must be planned for in the communications layout. It is advantageous to look at the shuttle system for several reasons: (1) the technology is all proven, tested, and space-rated, and (2) if the shuttle and STV share common components, they could use the same reserve of spare parts. Unfortunately, the mission parameters are different; therefore, so are the communication requirements. Consequently, redundancy may not be possible for every component [ref. 6.14].

The system can use TDRSS or GPS for tracking and position update purposes. The STV will use one of the large dishes for L-band reception of position update information.

Having chosen a transceiver system consisting of two 0.4 kW units, there is a maximum transmitting power of 0.8 kW. TDRSS high-gain, single-access K-band equipment will be used for communication purposes. This will allow smaller hardware on board the STV. The details of the system are outlined in Table 6.3. The antennas will be made of a Carbon epoxy material since it is used in many current space-rated antenna systems and has good structural and mass properties.

6.3 Data Management System

The data management system (DMS) provides command, control, and data processing for all the systems in the STV. The DMS consists of processors, storage devices, networks, workstations, and software. The DMS architecture, as shown in Figure 6.2, will be in the form of three central computers, a network system, and several subsystems which are controlled by their own individual processors [ref. 6.15]. The three central computers form a triple redundant system which coordinates all the STV's subsystems. Each central computer is individually capable of performing all the STV's functions; however, the three work in parallel and compare their data so that at least two of them must agree before a subsystem is told what to do. This information is then sent over the network to the subsystems. The subsystems then complete their assigned tasks and report back to the central computers. The central computers are themselves controlled by their own software and any input from the astronauts through the workstations.

6.3.1 Processors

The standard data processors (SDPs) are the heart of the DMS and provide the actual computational power that runs the entire STV. They will be located in all the subsystems as well as in the central computers, and should be the same ones used on the Space Station in order to maintain compatibility and low cost. Each subsystem has its own processors, so that once it receives its instructions, it can perform the assigned task without further using the central computers. This is necessary because the network is the bottleneck of the system and the more processing each subsystem does itself, the less it will have to use the network and the faster it will be. Using individual processors also helps isolate the subsystems so that if the central computers do fail, the subsystems can still operate. Microprocessor technology is advancing so fast that it is impossible at this moment to predict the actual parameters the STV's processors will have. However, any type of processor must be resistant to radiation damage. Since there are a limited number of chips with this qualification on the market, the processors will probably have to be either a military version or custom developed.

6.3.2 Storage Devices

Any storage device for the STV must be able to store large amounts of data, have high data transfer rates, and provide rapid access to the stored data. These storage devices must also be reliable, resistant to radiation, and have low volume. There are two types of devices that meet these requirements: magnetic hard disks and erasable optical disks. The advantages of magnetic disks are that they are a completely known quantity and there are already several space qualified versions. While erasable optical disks are a new development, they have several important advantages over magnetic disks. They have increased resistance to radiation and electromagnetic noise. The disks are portable and easily transferrable. The disks can also hold video as well as sound formats, therefore eliminating the need for magnetic tape. Because of these advantages, erasable optical disks are the storage system of choice.

6.3.3 Network

The network will consist of a fiber optic double ring system to communicate between subsystems. Fiber optics are advantageous since they are resistant to radiation, are extremely light, and have high data transfer rates. Between the subsystem and the fiber optic line will be a network interface unit (NIU). Each subsystem will have its own NIU to facilitate redundancy. These units gather data from the individual subsystems, via SDPs, and translate it into workable data to be transferred to the central computers. Refer to Figure 6.2 for a schematic. These network interfaces assume the burden of the communication between subsystems, so that each SDP will be free to concentrate on its own specific responsibilities. Each NIU has a mass of 5 kg. The NIUs, and all electronic circuitry, will be shielded from radiation and protected from overheating. Two types of shields were investigated for radiation protection: Indium Tin Oxide (ITO) and a Kapton Blanket [ref. 6.16]. The ITO was given preference because the Kapton Blanket generates some low level discharges which might affect current flow in nearby electronics. Finally, each NIU will be protected from extreme heat which could cause problems to the efficiency of the circuits. This is especially important to circuits in "hot" areas such as the propulsion unit. By shielding the NIUs and cooling the hardware, heating will not cause a significant problem. It should be noted that the NIUs make up the backbone of the the STV's avionics system. They coordinate the information from each subsystem to be relayed and managed in the central computers located in the flight deck.

STV Network Diagram

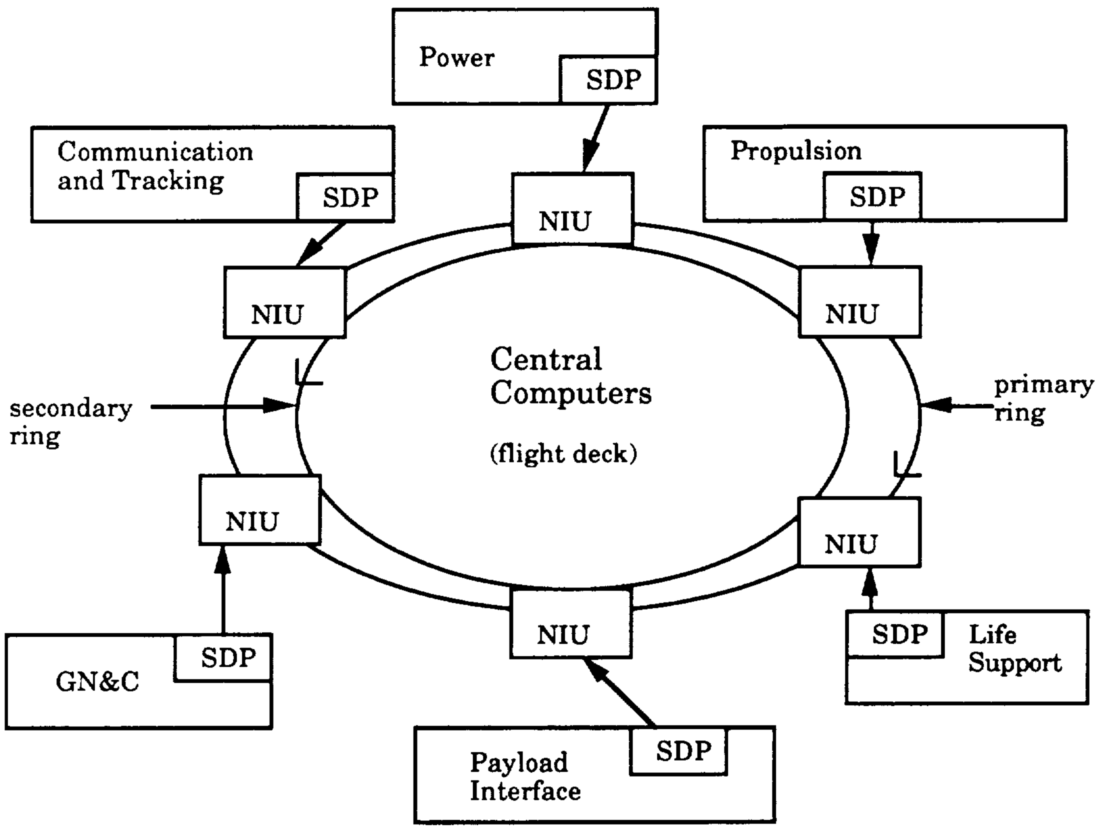


Figure 6.2 [ref. 6.17]

6.3.4 Workstations

Workstations are the interface between the astronauts and the STV. It is from here that the astronauts will receive information and issue commands. The main output devices will be a series of color, flat screen liquid crystal displays (LCDs). LCDs are used because they take up less room and power then conventional cathode ray tube displays, while color is necessary in order to better integrate the information on the screens for easy viewing. These displays will take the place of all the dials and meters as well as serve as the main output device from the central computer. This will make it easier for the astronauts to find the information they want since they only have to call it up on the display. The astronauts can also program the displays to show the information in whichever format they feel is most useful at that time. The main input devices will be a keyboard and a trackball for cursor movement. Keyboards are the most versatile of all input devices while a trackball is useful for controlling the cursor and needs only a very limited amount of space.

6.3.5 Software

The STV computers will be controlled by software stored on erasable optical disks. Besides the normal software needed to run the system, there will also be special expert system programs. Expert systems try to imitate the experience and reasoning powers of experts in solving problems [ref. 6.18]. The main use of these systems will be in the location, diagnoses, and repair of breakdowns in the equipment, both on the STV and in any satellites that need repairing. Therefore, before the STV goes out on a mission, it will first be given the appropriate expert systems to help the astronauts in their tasks, whether it be to repair a satellite or to check one out before its placement in orbit. Expert systems can also be used in schedule planning, stowage location, and system monitoring. The STV may also have voice control options. This will be especially useful when the astronauts are performing labor intensive tasks such as EVA and have their hands in use. The astronauts can then use a voice driven menu on a display screen in their helmets to perform operations.

6.4 Guidance and Navigation

Guidance is the control of the spacecraft to specific attitude and velocity constraints. The guidance system consists of several computer algorithms, as illustrated in Figure 6.3, and interfaces with the attitude control system and Inertial Measuring Units (IMUs). These IMUs are part of the navigation system, whose purpose is the determination of the spacecraft position, velocity, and attitude. This information is periodically updated by a navigation satellite system and star trackers.

6.4.1 Guidance

Crew

The crew enters into the guidance computer the desired state for the spacecraft. They might desire a roll, translation, or a combination of the two. Also, they may want the spacecraft to track a particular direction, such as toward the center of the Earth.

Maneuver States Computer

The maneuver states computer determines the trajectory to get from the current spacecraft state to the desired state. It takes input from the state estimator algorithm for the current state and the input for the final state from the crew. The final state may also be determined by the algorithm itself if the crew has selected a track option where the spacecraft automatically tracks a particular direction, or if the spacecraft is in the atmospheric flight portion of the flight. The maneuvers calculated are optimized for minimum fuel usage.

Maneuver Command Generator

The maneuver command generator determines which maneuver jets of the attitude control system should be used, and when they should be activated. The combination of jets selected is driven by the constraint to minimize fuel use and the number of jet cycles. Also, it is possible to restrict certain jets from use to prevent contamination of nearby spacecraft. The algorithm will select the fuel optimal combination from the remaining jets.

Maneuver Jet Error Corrector

The maneuver jet error corrector maintains the spacecraft on the desired state when the deviation is small. These deviations may be a result of jet thrust deviances from nominal, or small perturbations from other sources. This algorithm has the capability to generate maneuver commands to make necessary corrections.

Guidance/Navigation Scheme

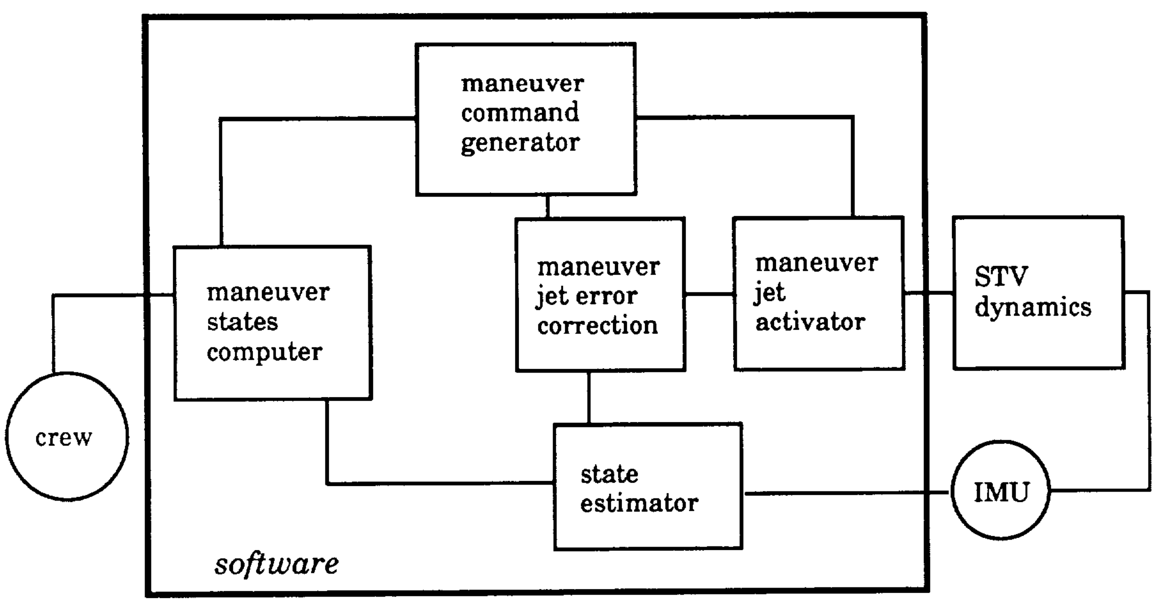


Figure 6.3 [ref. 6.19]

Maneuver Jet Activator

The maneuver jet activator is the subroutine which actually activates the specific maneuver jets. It continually monitors all of the jets to ensure that they are in their correct mode. Also, this algorithm maintains knowledge on which jets have failed and commands their redundant partners.

State Estimator

The state estimator determines the inertial state of the spacecraft and the mass characteristics. It inputs rotation rates and acceleration rates from the IMUs. A coordinate transformation is performed to derive them in an inertial coordinate frame. It integrates these to obtain the spacecraft position, velocity, and attitude. The mass characteristics of the vehicle are determined from combining knowledge of how the vehicle should have responded to a particular maneuver to how it did respond. From this information, the state estimator determines the mass, center of mass location, and the moments of inertia.

6.4.2 Navigation

Navigation Updates

Knowledge of position and velocity is critical before any propulsive burn is performed. More importantly, the atmospheric segment of the mission requires exceptional accuracy. The navigation quantities will come from the GPS, which consists of 18 satellites in 12 hour period orbits [ref. 6.20]. This system can be used from GEO altitudes down to the Earth's surface. The spacecraft does not transmit any signals to the GPS. It only needs to receive L-band signals to navigate. Range and range rate data from four satellites is determined by timing the arrival of beacon signals and their frequency shifts. The range measurements allow determination of position, and the range rate data allows determination of velocity. The data in the beacon signal contains information on the GPS movement, and this allows clock updates. The accuracy of this system is given in Table 6.4.

Navigation Accuracy

Location	± Position (m)	± Velocity (m/s)	± Attitude (deg)
GEO	141.5	0.47	0.03
In transit	73.8	0.25	0.03
LEO	18.5	0.06	0.03

Table 6.4 [ref. 6.21]

Attitude Updates

Knowledge of the spacecraft attitude is important in order to correctly perform burns and successfully reenter the atmosphere. The attitude is determined from knowledge of the spacecraft position, which comes from the GPS, and the relative positions of stars. The accuracy for the attitude is also given in Table 6.4. These star positions are measured by three star trackers located on the outside of the command module. Each has a mass of 3 kg, a volume of 0.01 m^3 (15 cm diameter and 13 cm long) and a power requirement of 9 W [ref. 6.22].

Inertial Navigation

Navigation from update to update is achieved using IMUs. These contain three accelerometers and three laser rate gyroscopes in a strapdown system. Each IMU has a mass of 15 kg, a volume of 0.02 m^3 (23 cm x 26 cm x 31 cm) and a power requirement of 50 W [ref. 6.22]. The guidance system takes the linear and rotational accelerations from the IMUs to determine the spacecraft position and velocity.

Alternate Navigation Updates

Range and range rate information can also be obtained from TDRSS. The use of this system will be limited below altitudes of 1200 km [ref. 6.20], which is why it is not the primary system. Navigation information can also be obtained from ground radar systems if necessary; however, this reduces accuracy. Finally, the star trackers can be used in stellar-refraction measurement geometry. This method would be accurate, but requires excessive time and is restricted by the relative Sun position.

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6.22 Courtesy of NASA Lewis Research Center, Cleveland, Ohio.

Chapter 7

Life Support and Human Factors

7.0 Summary

Life Support and Human Factors is responsible for the safety and comfort of the crew on the nominal mission and any expected derivations. For the two separate designs, all-propulsive (CSTV) and aeroassisted (ASTV), we designed pressure vessel environments which were equipped with all of the necessary life support systems for the safety and comfort of 2 men on the 7 day nominal mission with a 4 day emergency reserve. The nominal environment and Air Revitalization System are the same for both designs, as well as the system masses. However, different g-loading vectors and positioning requirements demanded 2 separate configurations with different cabin and radiation shielding masses.

We have chosen a Earth-like sea-level environment for the pressurized module to provide safety and comfort for the crew and to maintain compatibility with the Space Station. The life-critical systems were designed for triple redundancy for our man-rated vehicle. Because mass is an important factor in the operating cost of the vehicle, our design philosophy focused on minimizing total mass but without sacrificing safety.

Our food and waste systems are basically advance models of current designs used by the Space Shuttle and to be used by the Space Station. Standard nutritional menus, consistent with the Space Station, will be used, although most of the foods will be of the dehydrated or semi-dehydrated type to save mass.

When a vehicle leaves the protection of LEO, radiation shielding becomes a large factor in the safety and protection of the crew. The Van Allen radiation belts with its high proton flux and the GEO environment with a large electron flux produce radiation levels too high for the human body to tolerate. Therefore, it was necessary for our group to supply the crew with an adequate amount of radiation protection. Although many types of shielding were explored, the simple use of aluminum to absorb and deflect radiation particles was found to be the most practical for our needs. We also found it necessary to include an extra-massive radiation shelter which is designed to protect the crew from a solar flare. For mass considerations, this shelter was structurally integrated with the airlock, and the entire unit can be replaced with an less massive airlock without the integrated shelter for periods of low solar activity.

Satellite deployment, repair, and retrieval will be included in most of the mission envelopes. Therefore, the ASTV and CSTV were designed with extravehicular activity (EVA) in mind. Our crew cabins were designed so that the crew had easy access to the radiation space suits and the airlock, and could also maneuver the payload from inside the cabin using a remote manipulator system. A manned maneuvering unit (MMU) and a remote tele-robotic servicing unit will support the EVA's.

A summary of our design is detailed in Table 7.1.

Life Support Summary	
Nominal Environment:	
Pressure	55 - 101 kPa
Atmosphere	79% N_2/21% O_2
Temperature	16 - 32° C
Humidity	35 - 55%
Air Revitalization System:	
Oxygen/man	0.84 kg/day
Carbon dioxide removal system	Lithium Hydroxide
Odor Control	Activated Charcoal
Radiation Protection:	
Regular Shielding	2 gm/cm2 Al-Equivalent
Solar Flare Shielding	10 gm/cm2 Al-Equivalent
Life Support System Mass	
CSTV	2918 kg
ASTV	3018 kg
Maximum Power	2.05 kW

Table 7.1

7.1 Environmental Control and Life Support Systems

7.1.1 Cabin Environment

The nominal cabin environment will be the same for the CSTV and the ASTV. This environment consists of four main factors; pressurization, atmospheric composition, temperature and humidity, and carbon dioxide removal. All of these factors must be well within human limits, with an extra safety margin as an added constraint. The nominal environment is as listed:

Pressurization:	Variable between 55 kPa and 101 kPa
Atmosphere:	79% Nitrogen and 21% Oxygen
Temperature:	Variable between 16° and 32° C
Humidity:	Variable between 35 and 55%
CO_2 Partial Pressure:	0.4 kPa, nominal (1.0 kPa, max.)

The Environmental Control and Life Support Systems (ECLSS) will be an open system, which means that none of the atmospheric, life support, food, or water systems will be regenerable. An open system was chosen over a closed or partially-closed system because, for the length of our nominal mission, the extra expense and mass of a regenerative system does not prove to be cost effective or mass efficient. If the mission length increases above a couple of weeks, a partially regenerative system would prove to be worthwhile.

7.1.2 Air Revitalization System

The Air Revitalization System (ARS) provides the crew with a conditioned environment that meets both life support and crew comfort requirements. The ARS is responsible for 3 major functions: (1) maintaining the cabin pressure and controlling the oxygen/nitrogen mixture ratio; (2) removing the excess carbon dioxide; and (3) controlling the temperature and humidity, and removing odors and harmful trace contaminants. The ARS is identical for both the all-propulsive and aeroassisted versions of *Argo*.

Cabin Atmospheric Pressurization and Composition

The cabin atmosphere will be pressurized to 101 kPa with a 79% Nitrogen/21% Oxygen composition during the non-EVA phases of the nominal mission. While the nominal pressurization level and composition of the atmosphere was chosen primarily to maintain compatibility with the environment of the space station, a two-gas system maintained at sea-level conditions will also reduce the risk of fire and possible oxygen toxicity in an oxygen rich environment. The nominal partial pressure of O_2 will be 22.1 kPa and in no case will the partial pressure of O_2 be below 15 kPa due to safety considerations.

Prior to an EVA phase of a mission, the cabin atmosphere will be depressurized to 55 kPa which is compatible with the internal pressure level of the space suits carried by the *Argo*. Slowly depressurizing the cabin atmosphere to the level of the space suit will eliminate the pre-breathe phase that is currently standard procedure for Space Shuttle astronauts when they prepare for an EVA. Even though the Space Shuttle orbiter's cabin atmosphere can also be reduced from 101 kPa to 65 kPa for EVA's, the current space suits used by Shuttle astronauts have only an internal pressure of 28 kPa. Before donning their space suits, Shuttle astronauts must pre-breathe pure oxygen for 3 hours to reduce the risk of suffering aeroembolism, also known as "the bends." The space suits used by the

Argo's crew are advanced, hard-shelled models capable of maintaining an internal atmosphere of 55 kPa which will allow the crew to don the space suits without a pre-breathe procedure (see Section 7.5.2).

Each crew member will consume 0.84 kg of oxygen per day. For a nominal two man, seven day mission with a 4 day emergency reserve, a total of 21.8 kg of oxygen is required, which includes the replacement of 0.21 kg of oxygen lost per day in cabin leakage. In consideration of weight savings, the oxygen used by the ARS will be stored in the same cryogenic storage tanks used by the fuel cell power generation system. A single, triply redundant piping system will transport the oxygen from the storage tanks to the crew module. Then oxygen will be distributed to the fuel cells and the ARS.

The nitrogen used for the nominal atmosphere will be stored in gaseous form in two 22,750 kPa storage vessels, each with a mass of 10 kg and holding 28.8 kg of N_2. In addition to maintaining cabin pressure, nitrogen is also used to pressurize the potable and waste water systems. An allowance of 0.8 kg of nitrogen per day is made for cabin leakage.

Carbon Dioxide Removal

Each crew member produces about 1 kg of CO_2 per day. Since CO_2 is poisonous at high concentrations, it is critical to keep the partial pressure of CO_2 below 1.0 kPa. The nominal partial pressure of CO_2 in the *Argo*'s cabin atmosphere will be maintained at 0.4 kPa to allow for a safety margin. A lithium hydroxide (LiOH) system, a molecular sieve system, and an electrochemical depolarized carbon dioxide concentrator system (EDC) were considered for use in the *Argo* for carbon dioxide removal.

Currently, LiOH is used by the Space Shuttle for controlling the partial pressure of CO_2 in the orbiter's crew module. Cabin air is passed through cannisters of LiOH which absorbs the CO_2. LiOH offers the advantage of a simple and proven system. However, the LiOH cannisters must be discarded after each flight because they cannot be regenerated after they are saturated with CO_2.

Two regenerable systems were considered. The first system, which uses molecular sieves, has been flight proven and used on the *Skylab*. Molecular sieves, made out of synthetic zeolites, are similar in concept to LiOH [ref. 7.1]. To remove the CO_2, air is passed by the ventilation system over molecular sieves where CO_2 is absorbed. The advantage of molecular sieves is that they can be regenerated during flight by exposure to vacuum, during which the absorbed CO_2 is desorbed and outgassed to the vacuum. Power is required to heat the sieves in order to increase the CO_2 desorption rate.

An EDC CO_2 removal system is currently being considered for an extended duration orbiter mission [ref. 7.2]. The EDC system uses a process where H_2 and O_2 reacts in an electrolyte to remove CO_2 from the air and vents it overboard, similar to the reaction in a power generating fuel cell. For mission over 9 days for a two man crew, EDC can be shown to save mass over the LiOH system. However, EDC uses consumable H_2 and O_2 and produces heat, and thus is more complex than either of the two absorbent chemical systems previously discussed.

In Table 7.2, we have listed the various properties of the different systems as they are scaled for a nominal mission with a 4 day reserve. From the table, we see that not only does the LiOH system have the lowest total system mass (total = reusable + consumable), but the consumable mass of the LiOH system, which consists of the lithium hydroxide cannisters themselves, is bettered only by the EDC system. Thus we have chosen to use the lithium hydroxide CO_2 removal system for the *Argo*.

Comparison of CO_2 Removal Systems
(Nominal 2 man/7 day mission with 4 day reserve)

System	Reusable Mass	Consumable mass	Power Required
LiOH [ref. 7.3]	0.0 kg	26.4 kg	None, except for ventilation fans
Molecular Sieves [ref. 7.3]	20 kg	41 kg	0.4 kW
EDC [ref. 7.2]	30 kg	20 kg	0.05 kW

Note: Consumable mass is defined to be the mass of all materials which need to be resupplied at the end of each mission. This includes the mass of H_2/O_2 required to generate the power for the system and any H_2/O_2 required by the system for other purposes (EDC).

Table 7.2

Temperature, Humidity and Odor Control

Each crew member will produce 0.137 kW of metabolic heat which is added to the cabin atmosphere. Excess heat is also generated by the action of electrical devices such as fan motors and avionics. Also added to the atmosphere is 1.82 kg of water per person through respiration and perspiration and 0.82 kg of water through the action of the LiOH, which produces water vapor when it absorbs carbon dioxide. Therefore, to maintain the nominal cabin environment, both heat and water must be removed from the atmosphere. By the use of a combined heat exchanger/dehumidifier, we can cool the air and condense the evaporated moisture from the air at the same time.

The heat exchanger will be a conventional plate-fin, liquid-cooled device, whose metal surface is kept by cold water (the cooling liquid) below the dew point. Water is condensed from the moist air onto the surface of the heat exchanger, where it is blown into a centrifugal water separator. The condensate is then pumped into waste water storage tanks. The mass of the temperature and humidity control system, not including the storage tanks, is estimated to be 10 kg. Total power used by the system is estimated to be 0.075 kW.

Dry cool air is then passed through an activated charcoal bed to remove odors from the air. 0.6 kg of charcoal is sufficient for our nominal mission [ref. 7.4]. Many harmful oxides such as CO or NO_2 will be absorbed by the lithium hydroxide, so due to the relatively short length of our nominal mission, we will not use a catalytic burner to remove other trace contaminates.

A flow diagram of the Air Revitalization System is presented in Figure 7.1. Cabin airflow rate will vary between 4.6 m/min to 12.3 m/min, with a nominal flow rate of 7.7 m/min. We can see from the diagram that in one pass through the ARS, CO_2 and odors are removed and the cabin air is cooled and dehumidified.

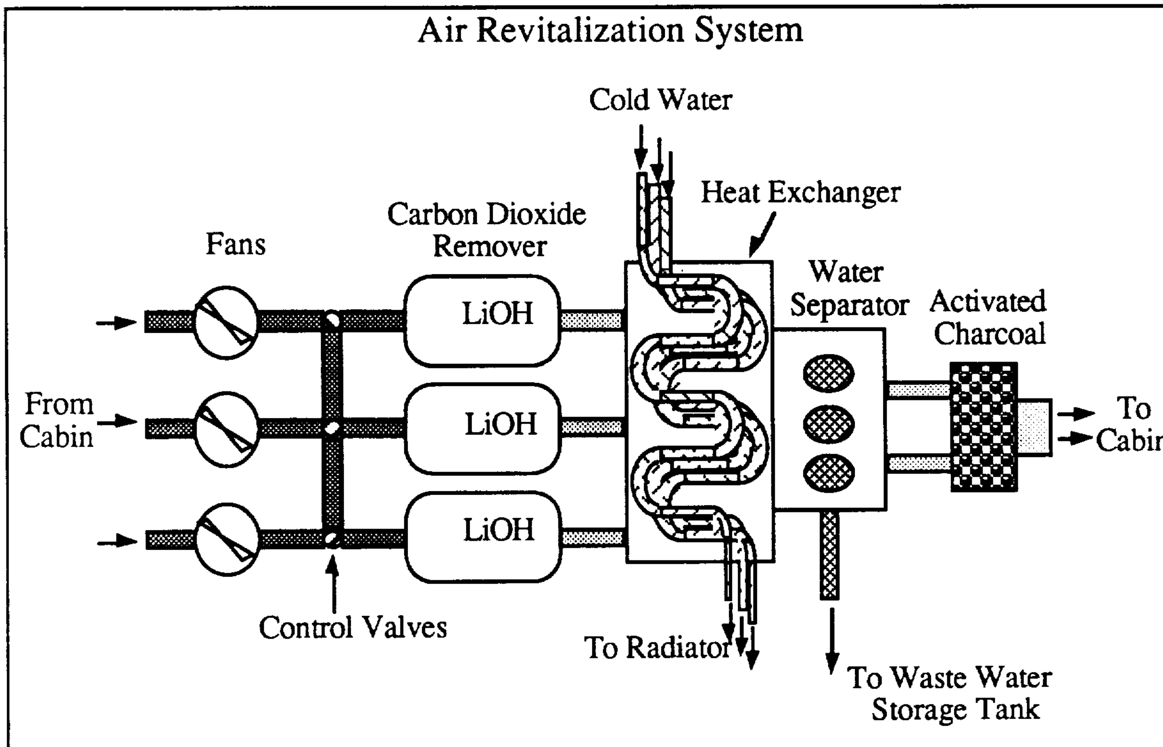

Figure 7.1

Redundancy and Safety Considerations

All of the subsystems of the ARS are triply redundant for safety considerations except the activated charcoal bed, which is fairly failure-proof and does not pose a hazard to human life if it does fail. In addition to the 4 day emergency supply of nitrogen and oxygen carried by the *Argo*, we also will have two 22,750 kPa tanks of gaseous N_2 and O_2 for emergency pressurization in case of the development of a leak in the cabin. Each tank will have a mass of approximately 40 kg, fully loaded.

Only one of the three fans and LiOH absorbing beds will be operating at any time. The cross-connected control valve system allows us to use any fan in conjunction with any LiOH bed (see Figure 7.1). When one LiOH bed is saturated with CO_2, its inlet valve is closed and another one is opened. Even though we use only one heat exchanger in our ARS, triple redundancy is maintained by the use of three independent water cooling loops in the exchanger. Should any of the cooling loops fail, the other two loops will be able to maintain the nominal temperature and humidity. However, should any two of the cooling loops fail, the crew will have to run in a down-graded mode, with non-essential avionics and other electrical equipment turned off to reduce the head load on the system.

Portable Halon 1301 extinguishers will be used for fires occurring within the crew module. Automatic, fixed extinguishers, controlled by smoke and heat detectors will be used in avionics, fuel cell, and storage bays.

7.1.2 Nutrition and Water Systems

Human life requires certain basic inputs in order to sustain itself. These inputs can be broken up into three categories: food (energy-in calories), water, and oxygen. The oxygen requirement has been dealt with in Section 7.1.1 of this report. This section will examine the other two requirements.

Dietary and Health Requirements

For the crew of *Argo*, an average of 2900-3000 calories will be provided per person per day. This matches the current levels on the Space Shuttle and projected levels on Space Station *Freedom*. The specific nutritional requirements for the crew will parallel and be integrated with those used in the Space Station. Two major problems observed with prolonged exposure to zero-g are calcium depletion and muscle deterioration. No way is yet known to counter this calcium loss, however exercise has been shown to reduce muscular atrophy. On STV missions lasting longer than three days, provisions will need to be made for the crew to exercise, keeping in routine with Space Station. This can be accomplished using one of a few simple elastic devices now under development.

There are two main sources of water intake. The most obvious and greatest source is drinking, either of water or any water-based fluid. The second source of water for the body is water available from food. The nominal allocation of drinking water is 2.35 kg per man per day. The water derived from food is 1.22 kg per man per day. This is the water that is needed for nutritional use only. Additional water is needed per day per astronaut for personal hygiene, food preparation and to take into account losses while preparing food, etc. 1.4 kg per day of wash water will be allocated for each crew member. This number is fairly arbitrary but has been chosen as being reasonable after looking at the amount used on previous space missions, the amount used in the Space Shuttle, and on projected allocations on the Space Station. The water derived from food takes into account all water used for food preparation minus some slight losses. The total water requirement/allocation for each astronaut, adding a small factor for losses, comes to 5 kg per day.

Food Preparation

When choosing the type of food system used on the STV, four criteria were considered. The first of these criteria was that the food must be easy to prepare within the small area provided and that the food preparation equipment be minimized with respect to volume and mass. The second consideration was compatibility with Space Station, if at all possible. The third was that, wherever possible, already developed systems should be used. The fourth and not the least important consideration was to make the food as appetizing as possible. The system that best satisfies all of these considerations is one derived from the Space Shuttle system.

The STV will operate to and from the Space Station. It would therefore seem most ideal if the food used on *Freedom* could be used on *Argo*. However, this is not the case. Most of the Space Station food will be "wet" i.e. it will be supplied frozen, refrigerated, or canned. The mass of the refrigerators and freezers needed to store "wet" food for a seven day mission is alone prohibitive, not to mention the large mass of the canned or frozen food itself. Currently the shuttle menu consists of foods which are dehydrated, thermostabilized, irradiated, intermediate moisture, natural form, and beverage foods. These foods are light-weight, easy to store and prepare, and are already being produced for the Space Shuttle.

The exact menu will be decided at the time of each mission. The menu will be integrated with the Space Station so that the nutrition and diet of the crew of the STV will match that of those still on the Space Station. This will aid in keeping the *Argo* crew in routine with the Space Station and in monitoring the health of STV crews. In an ideal case, while the Space Station crew are eating beef stew for dinner, the STV crew will also be eating an equivalent stew. For each mission, the crew of *Argo* will be provided with enough food for three meals a day, including snacks. The mass of this food, including packaging, is 0.91 kg per astronaut per day. This means the mass of the food for the nominal seven day mission is 12.74

kg. An additional four days of emergency food will also be aboard for every mission.

The food preparation will be accomplished in a minimal galley area. The galley will basically be a section of one wall in which there will be a pantry, an oven, food trays, water dispensers, and support items. The pantry will be able to store the 20 kg of food that the nominal mission will require. As the galley will not take up the entire wall, there will also be room for miscellaneous storage (see Figure 7.2). The oven will be a forced-air convection heater of the same type used on the shuttle. Cleaning of utensils and trays will be accomplished using sanitized "wet wipes" that contain a quaternary ammonium compound, as on the shuttle.

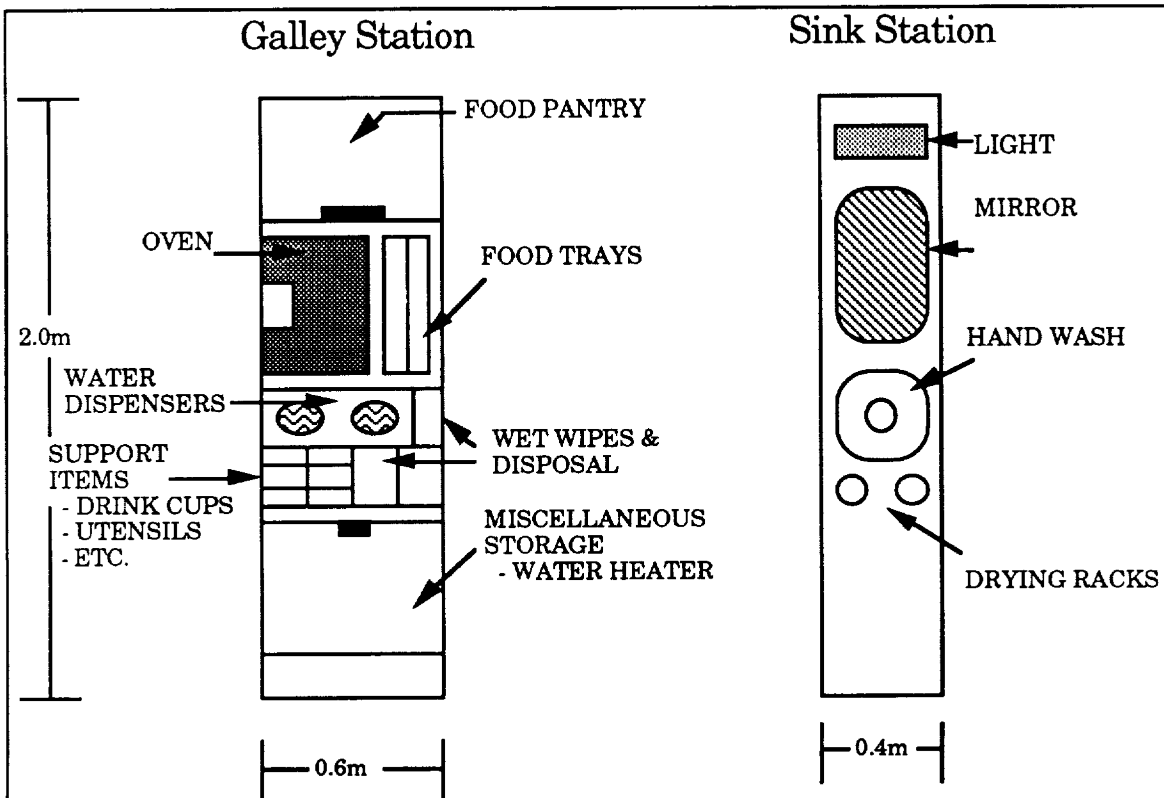

Figure 7.2

Water systems

The two main needs the astronaut has for water are (1) nutritional needs as outlined earlier, and (2) personal hygiene needs. The second is necessary to insure the health and comfort of the crew during any flight. The first need must satisfied by water that is pure enough to drink. As the STV, due to mass restraints, will have an open-loop water system, both wash and potable water will come from the same source. Waste water from the sink will not be recycled in any way. We will begin each mission with only 8 kg of water. As the mission progresses, water will be supplied by the fuel cells which produce approximately 34 kg of water per day. This more than meets the needs of the crew. The triple redundancy of the fuel cells also means triple redundancy in water supply.

The water will be pumped from the fuel cells to a tank capable of holding 7 days worth of water (70 kg). This tank will have sensors and plumbing so that once the tank is full, any excess water will be vented to space. This seven-day level will be constantly maintained.

Water for drinking will be provided by a water dispenser gun. One gun will be located in the galley section and another at the personal hygiene station. These guns will also be used for food reconstituting and washing respectively.

The personal hygiene system was based on the Space Shuttle, in much the same way the galley was. The personal hygiene station will consist of a hand washing area, a water gun, the hygiene water valve, a soap dispenser, drying racks for washcloths, a mirror and a fold-out panel light. The station will also have controls for draining water and temperature control. The hygiene water valve is a squeeze valve that provides water at ambient temperature. The water guns will be able to provide chilled water and hot water, heated by the water heater in the galley station. An air flow valve will connect the the station with the waste collection system. This air flow will create a suction to create a free-flow of water over the hands or a washcloth. The crew will take "washcloth baths" to clean themselves. The washcloths can then be dried in the rubber, slit drying rack located below the sink station. Each astronaut will have a personal hygiene kit which will include washcloths and towels, dental hygiene supplies and all other hygiene needs.

7.1.3 Waste Management

The crew cabin of the STV will provide a system to process all human wastes produced during a mission. This waste collection system is an integrated multi-functional system used to collect, process, and store solid and liquid wastes. The system is used the same way as a normal facility and performs the following general functions: (1) collecting, storing, and drying fecal wastes, associated toilet paper, and emesis-filled bags; (2) processing wash water from the personal hygiene station; (3) processing urine; (4) transferring the collected fluids to the waste storage tanks in the waste management system; (5) venting the air and vapors from the wet trash container and stowage compartment; and (6) the cabin water from the cabin heat exchanger.

Commode System

Three major systems used in past and present designs were examined: the *Skylab*, the Space Shuttle and the Space Station commodes. From these three designs the Shuttle commode was chosen because of its compactness, reliability and proven technology. Its self-contained design is useful for our vehicle needs. This commode is a unit of approximately 69x69x74 centimeters and it will be located on the pilot's left side tunnel for the all-propulsive and on the pilot's left underside for the aeroassisted vehicle (See Figure 7.3).

This waste collection system which resembles an Earth-like toilet accommodates both male and female crew members and consists of the commode assembly, the urinal assembly valving, instrumentation, interconnecting plumbing, the mounting framework and restraints. The unit has two major independent and interconnected assemblies: the urinal part and the commode. This compartment has a "sliding door" which will isolate the area from the rest of the cabin.

Designed for a zero-g environment, the commode differs from its Earth counterpart in that there are foot, waist and hand restraints provided to keep the crew member from floating and to maintain an adequate seal between the user and the seat. To make both solid and liquid wastes flow downward from the point of departure, a high velocity air stream is applied to pull and retain all wastes inside the unit. The waste collected is then vacuum dried, stored and chemically treated to prevent odor and bacterial growth. This toilet system will be serviced upon return to the Space Station.

Trash Management System

The trash management system consists of the storage and collection of wet and dry trash. Dry trash consists of disposables items such as wipes, tissues, and food containers while wet trash consists of items that could offgas. The facilities available to take care of the trash includes trash bags, trash bag liners, wet trash containers, and the stowable wet trash vent hose.

There will be two trash bags containing trash bag liners in the crew cabin. One bag will be for dry trash and the second bag will be for wet trash. At a specific time of day the dry trash bag liner will be removed from the trash bag and closed with a strip of velcro and stowed in a storage locker. The wet trash bag liner will be removed at a specific time of day and placed in a wet trash container. In the case that offgassing is evident inside the wet trash bag, the container will be connected to a vent hose in the waste management system and vented overboard.

All the information used to support this waste management system was obtained from the Space Shuttle News Reference packages published by NASA and by Rockwell International.

STV Commode

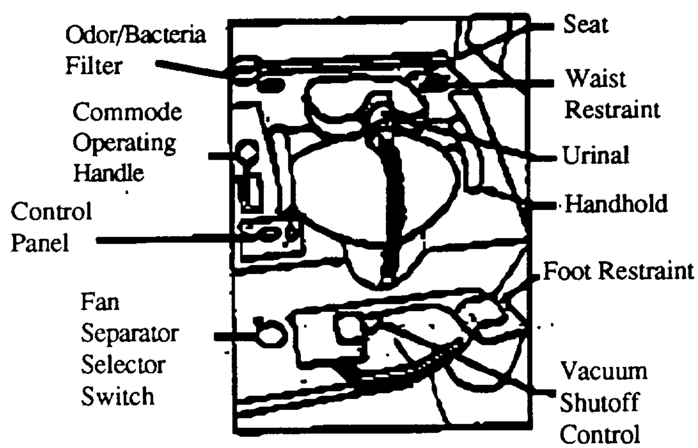

Figure 7.3

7.2 Radiation Protection

7.2.1 Radiation Environment

At LEO both radiation strength and flux are low due to the protection of the Earth's magnetic field. Therefore, astronauts require only minimal shielding. For example, Space Station *Freedom* has a mass radiation shield of 0.86 g/cm^2 aluminum equivalent (Al-eq.). Aluminum equivalent refers to using aluminum as a reference material for radiation protection. For example, if 2 g/cm^2 of material X gives 1 g/cm^2 Al-eq. of radiation protection, then it would take 1 g/cm^2 of aluminum to give the same level of protection as 2 g/cm^2 of material X. Since the Space Station will vary in altitude, the dose to the Space Station crew will vary from 0.02 to 0.3 REM/day [ref. 7.5].

The Van Allen belts are regions of energetic charged particles trapped by the Earth's magnetic field. Between LEO and GEO lies the most hazardous portion of the Van Allen radiation belts. The belts vary in shape, radiation strength, and flux with altitude, latitude, and longitude (see Figure 7.4) [ref. 7.6]. In general, the inner belt consists of high energy protons (up to 10^9 eV) and lies between 500 km and 10,000 km in altitude. The outer belt begins at 12,000 km and extends to approximately 60,000 km in altitude [ref 7.7]. Beyond 30,000 km the outer belt radiation is not severe. The electron flux, and therefore, the outer belt's boundary vary with solar cycle. The inner belt's high energy, high flux protons are the greatest threat to astronauts. Fortunately, the STV will spend only a small fraction of its mission in the Van Allen belts.

Van Allen Radiation Belt

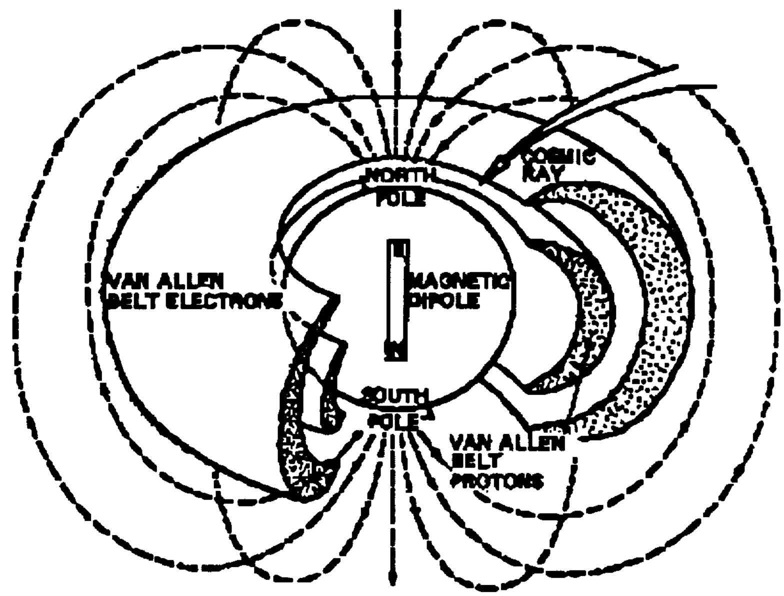

Figure 7.4

Although GEO is technically inside the outer Van Allen belt, it is considered free space. GEO is characterized by virtually no proton flux but a high flux of electrons. GEO's radiation environment varies with longitude and time by as much as a factor of 10 [ref. 7.8]. Since the STV will spend the majority of its mission in GEO, radiation shielding must primarily address this environment.

Cosmic radiation consists of 85% protons, 13% alpha-particles, and 2% heavy nuclei with energies from 10^6 eV to 10^{20} eV [ref. 7.7]. Since cosmic ray flux is small, it will not greatly contribute to the net radiation dose received at GEO.

7.2.2 Radiation Limits

Radiation is quantified by the Rad which is the amount of radiation needed to cause the absorption of 100 ergs of energy in 1 g of material. Radiation dose is quantified by the REM (Roentgen Equivalent Man) which is calculated using the formula REM = Rad x RBE. RBE is the Relative Biological Effectiveness of a particle, and will vary with type of particle, its energy, and the type of tissue [ref. 7.9]. Table 7.3 gives current NASA dose limits [ref. 7.5].

NASA Radiation Exposure Limits (REM)

PERIOD	BFO	EYE	SKIN
30 Days	25	100	150
Annual	50	200	300
Career	200+	400	600

BFO-blood forming organs

Table 7.3

Total radiation dose is the cumulative effect of incident and secondary radiation. Secondary radiation is caused by collisions between radiation particles and cabin walls, space suits, etc. Another form of secondary radiation is Bremsstrahlung (x-rays) caused by the interaction of energetic electrons and other matter. Since the GEO environment is primarily electrons, Bremsstrahlung radiation contributes significantly to total dose.

7.2.3 Radiation Shield Types

Three types of radiation shielding were investigated; electrostatic, magnetic, and mass shielding.

An electrostatic shield uses a radiation particle's charge to repel it from the vessel. It accomplishes this by using high voltage spherical grids. The grids must have electrical energy equal to the kinetic energy of the radiation particles that it must repel. An electrostatic shield must be spherical, must have a very large power source, and does not easily adapt to capturing and deploying payloads. For these reasons, *Argo* cannot use an electrostatic shield.

Magnetic shielding uses a powerful magnetic field generated by superconducting electromagnets to deflect charged particles. This is the same principal as the Earth's magnetic field protecting LEO spacecraft. A magnetic shield requires a toroidal shaped spacecraft, massive super-conducting electromagnets, and cryogenic storage for the magnetics. Due to these technical problems, and the necessary size and shape of our STV, magnetic shielding is not practical [ref. 7.10].

Argo will utilize mass shielding to protect the astronauts from harmful radiation. Mass shielding is simply the use of matter to block or absorb energetic charged or neutral particles. To minimize total vessel mass, only the crew cabin will have this protection. We arrived at the amount of mass shielding by optimizing total mass of the shield vs. dose received. 2 g/cm^2 Al-eq. is the optimal level of mass shielding [ref. 7.11]. As mass shielding increases in thickness, absorbed dose from incident radiation decreases while dose from Bremsstrahlung radiation increases. Therefore, large increases in shielding beyond 2 g/cm^2 Al-eq. will only slightly decrease the total absorbed dose while the total mass of the shield will rise sharply. 2 g/cm^2 Al-eq. translates to cabin walls of 0.74 cm thick. For the sake of redundancy, the design of the cabin walls will be two 0.37 cm thick pressure vessels. Thermal insulation, environmental ducts, and electrical conduits can be placed in between the pressure vessels.

Unfortunately, aluminum is not the best radiation protection material. Optimal mass shielding would use a laminate of a low Z outer layer and a high Z inner layer [ref. 7.12]. Z refers to the charge of a nucleus. An example of a laminate would consist of an outer layer of carbon fiber-epoxy composite (low Z) and inner layer of tantalum (high Z). Vacuum spaces, structural aluminum and insulation may be placed in between the layers so the laminate could serve as a redundant pressure vessel. An advantage of a laminate would be a significant mass savings. As of this report, hard data on the structural characteristics, mass, and protection abilities of laminates were not available. Therefore, this design uses only aluminum for its radiation shielding.

Since the radiation environment at GEO varies as a function of time and longitude by as much as an order of magnitude, an exact dose rate is difficult to quantify. Therefore, this shield was designed for the worst longitude, and the average dose at this longitude for our shield was multiplied by a factor of 10 to account for worst case variations. For a shield of 2 g/cm^2 Al-eq. the worst case dose rate will be 0.51 Rem/day BFO [ref. 7.11].

The Van Allen belts have substantially greater radiation strengths and fluxes. For a round trip from LEO to GEO and back using the proposed elliptical outbound transfer and the Hohmann transfer for the return trip, our shield will yield a dose of 2 REM/trip BFO [ref. 7.13].

Given a crew rotation of 90 days, 4 missions per crew rotation and a stay at GEO of 7 days/mission, the worst case absorbed dose of the STV crew will be 41 REM BFO. The average dose for the same profile will be 21 REM BFO. 41 REM BFO is within current NASA radiation limits.

7.2.4 Solar Flare Protection

Solar flares are unpredictable explosions of high flux radiation (primarily protons and alpha-particles) from the sun's surface. A major flare would yield a lethal dose of radiation to the crew of *Argo*. Therefore, steps must be taken for this contingency. Two plans were evaluated; a "storm shelter", and an immediate escape to LEO. LEO is protected from solar flares by the Earth's magnetic field.

An immediate escape to LEO would necessitate a plane change at LEO which uses much more propellant than a GEO plane change. The mass of this additional propellant is much greater than the mass of a storm shelter. An alternate plan to use the OMV as a tug or tanker to eliminate the need to carry the extra fuel was found to be logistically unsound. Therefore, a storm shelter was adopted for solar flare protection.

Argo's airlock was modified to serve as the storm shelter. The design and interior layout of the airlock/shelter is discussed in Chapter 2. The optimal amount of shielding for the storm shelter is 10 g/cm^2 Al-eq. [ref. 7.8].

In the event of a solar flare, the crew will have to spend a maximum of 20 hours in the shelter. During a major flare the crew will receive a non-lethal dose, but they will have to be rotated to Earth as soon as possible. *Argo*'s crew will have approximately five hours notice to reconfigure the airlock into the storm shelter [ref. 7.8]. The procedure will be to remove the space suits from the airlock to the crew cabin, install the false bulkhead, fold down the shelter's benches, and install the auxiliary control panel. Finally, the crew will provision the shelter with the necessary food, water, and Apollo-style bathroom equipment from the ship's stores. The crew will have minimal communication and control of the ship. The escape to LEO burn will consist of a Hohmann transfer with a GEO plane change, and will be done using remote-telemetry or on-board software. As the STV approaches LEO, the crew will be protected from the flare by the Earth's magnetic field, so they can leave the shelter, control LEO entry and dock as usual. To reduce dry mass, and fuel requirements, this airlock can be replaced by a lighter airlock without the massive shielding at times of low solar flare activity.

7.2.5 Mass Estimate for Radiation Shield

Taking into account the radiation protection afforded by the masses of the internal equipment, exterior avionics packages, and the propulsive package, the mass of the CSTV's radiation shield will be 550 kg. The ASTV's radiation shield will have a mass of 650 kg. The differences are due to different cabin dimensions, and different spacecraft configurations between the two designs. Fortunately, this shield can also serve as the cabin pressure vessel. Cabin structural supports and interior structure are not included in this estimate. The mass of the solar flare storm shelter will be 575 kg for both designs.

7.2.6 Surface Charging

Another danger from charged particles is the collection of charge on spacecraft surfaces. If large voltages build up on different surfaces of

Argo, arcing can occur, thus causing damage to the spacecraft [ref. 7.14]. Furthermore, potential differences can occur between EVA personnel and the STV. To eliminate voltage between spacecraft surfaces, all surfaces will have electrical continuity, and a conducting tether would protect EVA personnel from the same danger.

7.3 G-Loading Requirements

7.3.1 G-Tolerance of Humans

Life Support and Human Factors is responsible for ensuring that the crew of the *Argo* do not experience decelerations or accelerations that will endanger the life of the crew during the nominal mission. We are also responsible for configuring the crew in a position that is as comfortable as possible during the adverse g-loading. By using the data from the nominal mission, we were able to compare the g-loading experienced by the CSTV and the ASTV with data from previous space missions and centrifuge testing. To determine our mission limits, we scaled down reasonable centrifuge test data [ref. 7.15] for the added safety and maneuverability of the crew. We also took into account that the *Argo* crew will be in the weightless space environment for some time before the STV mission and will lose an unknown amount of their g-tolerance due to the loss of body fluids and muscle mass.

Our conclusion for the mission limits is that the STV crew should have a maximum acceleration-deceleration load of 4 g's. In addition, the crew should not experience a loading of over 1 g for more than 2 minutes.

7.3.2 All-Propulsive Requirements

The only excessive acceleration that the All-Propulsive STV will experience is when the engines are being fired for orbital transfers. This acceleration is under 1 g, so the CSTV is not limited by our restrictions. For comfort of the crew, however, the seats are configured in a way that directs the g-loading vector perpendicularly through the crew's chests (see Figure 7.5).

7.3.3 Aeroassisted Requirements

The Aeroassisted STV will be subject to a large deceleration as it enters the earth's atmosphere and undergoes the braking maneuver. The g-loading of this deceleration is much more substantial than the acceleration from firing the engines. Knowing the angle of entry and the velocity as the ASTV enters the atmosphere, we can plot the deceleration curves for the low L/D aerobrake (see Chapter 4). The maximum deceleration for the low L/D aerobrake is 2.5 g, and the time above 1 g is 1.4 minutes. This ASTV deceleration profile is acceptable by the limits which we established so we did not have to recommend any mission restrictions.

However, the magnitude of the deceleration makes it important that the crew is positioned with the deceleration vector directed in a way for the crew to tolerate the highest g-load possible. As a result, the main design constraint in the ASTV is that the crew are situated with the deceleration vector pointing straight through their chest, from front to back. With the crew configured in this manner (see Figure 7.6), there should be no difficulty in the crew tolerating the loading due to atmospheric braking.

7.4 Interior Design

The interior layout and design of the crew cabin was constrained by 4 major criteria: (1) g-loading during the engine burn and aeroassisted portions of the flight; (2) orientation of the crew cabin with respect to the payload bay; (3) zero-g design considerations; and (4) necessity of a solar flare radiation shelter. Because the location and orientation of the crew module are different between the all-propulsive and aeroassisted versions of *Argo*, the interior layouts of the module are quite dissimilar, although the criteria for which the layouts were designed remains the same. The final configuration of the all-propulsive and aeroassisted vehicles can be seen in Figures 7.5 and 7.6, respectively. An skeletal 3-D view can be seen in Figure 7.7.

G-loading Considerations

In Section 7.3, you will find extensive discussion on the g-loading specifications for both versions of *Argo*. The criteria which was set in Section 7.3 is that the acceleration vector or "g-vector" must pass perpendicularly through the crew member's chest from front to back.

Payload Bay Orientation

Another requirement for cabin layout is such that the crew must be able to visually observe payload operations and operate the remote manipulator system (RMS) from the RMS control station.

Zero-g Design Considerations

Zero-g considerations are too numerous to list, but they are discussed in the *Crew Systems and Crew Accommodations* section of reference 7.12. One factor is that the human body assumes a different posture in zero-g, somewhere between standing and sitting. To strap an astronaut, in space, into a chair designed for 1-g would cause discomfort and undo strain on the individual. Thus, we have chosen to use modified Apollo-type, reclined couches which conform to the natural position of the human body in zero-g. The command couches will also double as sleeping couches, since they should be quite comfortable and the length of the nominal mission does not require us to have separate sleeping quarters.

Other factors which need to be taken into consideration are the necessity for hand, foot and body restraints, and the need to protect vital switches and controls from inadvertent actuation.

Solar Flare Radiation Shelter

The necessity for a solar flare radiation shelter is discussed in Section 7.2.4. To reduce the weight of the actual shielding required, we have integrated the structure of the shelter with the structure of the airlock (see Chapter 2).

All-Propulsive Cabin Layout

From Figure 7.5, we can see that the couches are orientated with their backs perpendicular to the plane of thrust produced by the main propulsion system. The windows overlook the payload bay area in the front of the vehicle and allow visual observation of the operation of the RMS. There are also windows in position to allow observation of the berthing maneuver with the space station.

All command consoles are either accessible from either crew couch position or are duplicated for each crew member. The commode and galley areas are located to either side of the command couches. An auxiliary console allowing minimal control of the vehicle and communication with mission control will be installed within the shelter during solar flare conditions.

Aeroassisted Cabin Layout

Since there are two different acceleration vectors produced during an aeroassisted mission (main engine burns and aerobraking deceleration), and since these two vectors are not co-linear, we have chosen to orientate the command couches perpendicular to the vector with the greater magnitude. The deceleration vector produce by aerobraking is about two times greater than the acceleration vector produced by engine thrust (see Section 7.3). Thus, the orientation of the couches are determined (see Figure 7.6).

Like the all-propulsive version, windows are located to allow visual observation of both payload and berthing operations. Both the galley and the commode are located beneath and to the side of the couches. Again, emergency control and communication facilities will be installed in the shelter during solar flare conditions.

All-Propulsive Cabin Layout

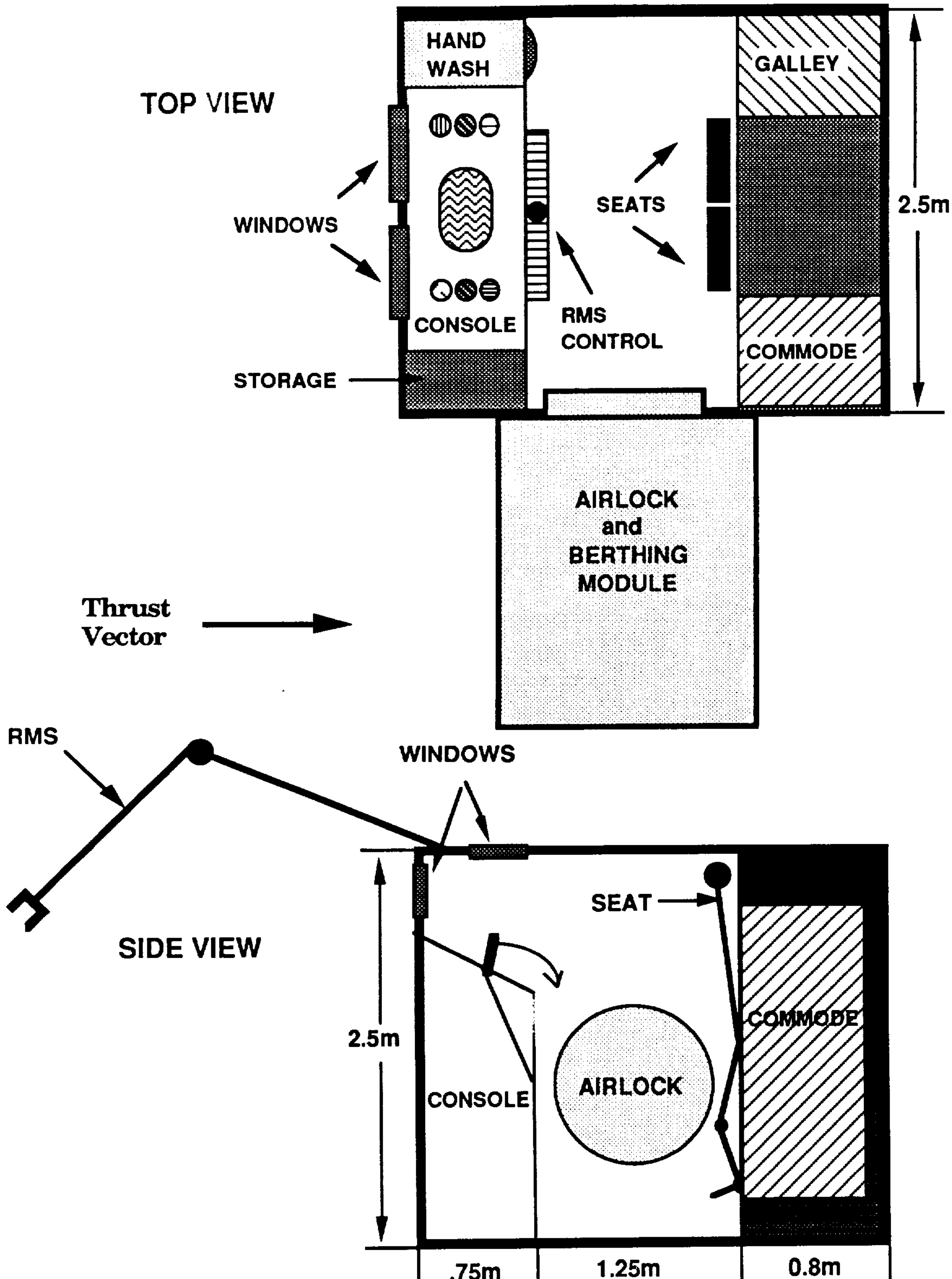

Figure 7.5

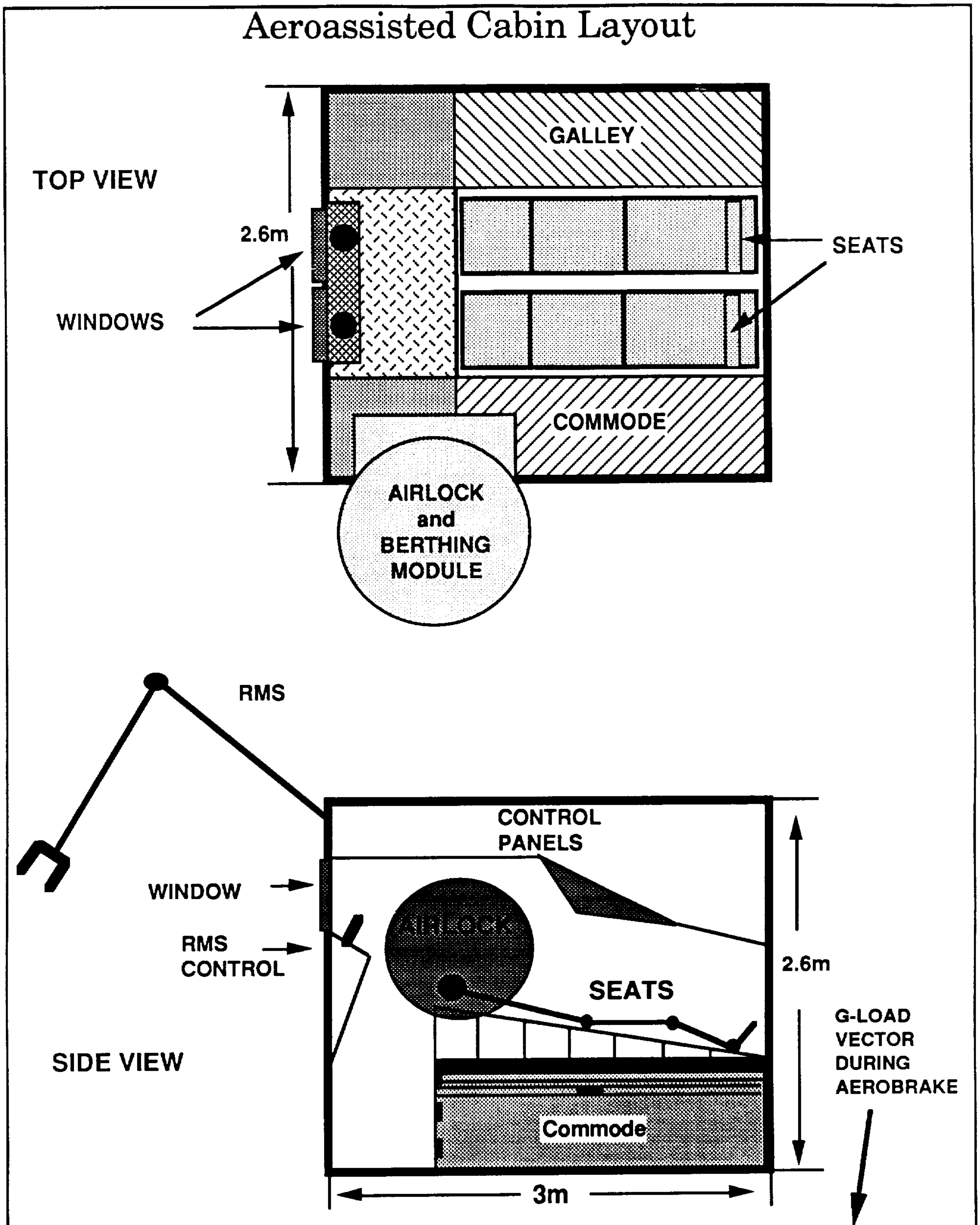
Aeroassisted Cabin Layout
TOP VIEW
GALLEY
2.6m
SEATS
WINDOWS
COMMODE
AIRLOCK
and
BERTHING
MODULE
RMS
CONTROL
PANELS
WINDOW
RMS
CONTROL
SEATS
2.6m
G-LOAD
VECTOR
DURING
AEROBRAKE
SIDE VIEW
Commode
3m

Figure 7.6

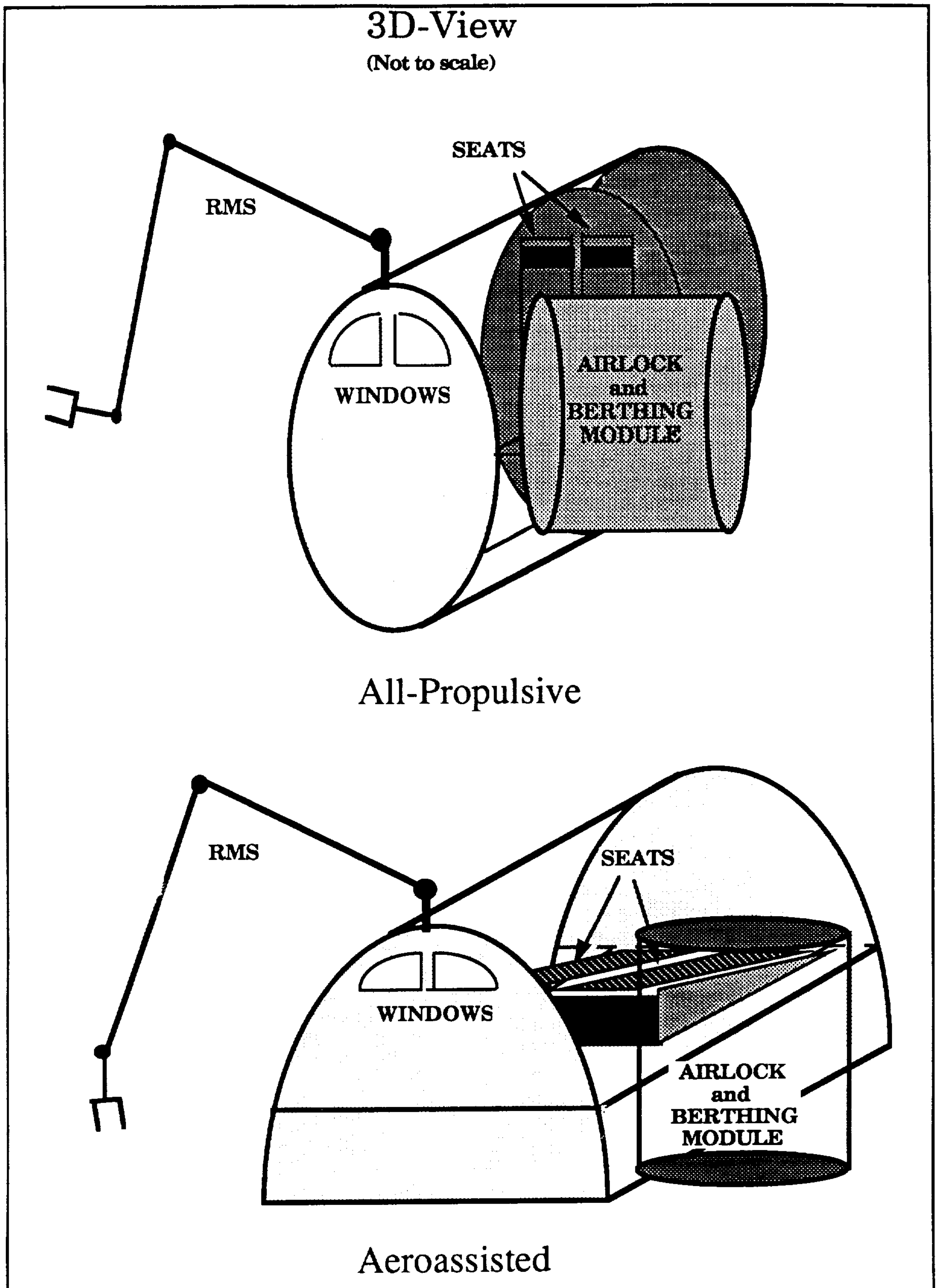
3D-View
(Not to scale)
SEATS
RMS
WINDOWS
AIRLOCK
and
BERTHING
MODULE
All-Propulsive
RMS
SEATS
WINDOWS
AIRLOCK
and
BERTHING
MODULE
Aeroassisted

Figure 7.7

7.5 Extravehicular Activity (EVA)

7.5.1 Mission Requirements

A major reason that the STV will be manned is the flexibility that the human presence provides for any mission. A large part of this flexibility is derived from the fact that astronauts can perform delicate and complex tasks outside of the *Argo*'s environment by means of an EVA. The function of EVA within the scope of the STV's mission can be divided into two major categories: (1) repair and (2) servicing.

Repair

The first function that EVA has is the repair of satellites that are at GEO. Two advantages are gained by repairing satellites at GEO. The first advantage is that by repairing a satellite at GEO, the cost of bringing the satellite down to the Space Station to be repaired is eliminated. This is especially advantageous if only a minor repair is necessary. The second advantage is that if it is impractical to bring a satellite to *Freedom* at LEO, the cost of replacing the satellite can be saved if we can repair the satellite at GEO. Along these same lines, by repairing a satellite, its life can be extended, thus deferring the cost of replacement. This leads into the second category, servicing.

Servicing

The second function of EVA from the *Argo* will probably not be realized immediately. Unmanned scientific and communication platforms are planned for placement in GEO during the lifetime of the *Argo*. EVA servicing of these platforms will allow for more flexibility in the experiments performed. It will also extend the effective lifetime of these platforms by allowing for the replacement of consumables and for general maintenance and upkeep. The same is true for later generation satellites. By designing a satellite to be serviceable, its lifetime can be extended, thereby reducing replacement costs. With servicing, it is conceivable that after a satellite is launched, it could remain indefinitely in operation with the routine replacement of attitude control propellants and repair of failed components.

7.5.2 EVA Suit

EVA suits must be able to satisfy the physiological needs of a human in space, while at the same time remain practical for use. In meeting these two criteria a suit must specifically (1) maintain an internal pressure in which the astronaut can live and (2) remain flexible enough to perform work tasks efficiently. The EVA suits currently being used on the Space Shuttle are made entirely out of fabric and fit these criteria well. They are known as "soft suits". The problem these suits have, however, is that the internal pressure of the suits must be kept rather low. If this pressure is too high these suits lose their flexibility. Because the suits operate at such a low pressure, about 28 kPa, astronauts must pre-breathe pure oxygen for extended periods to prevent the "bends". Pre-breathe time can be shortened by lowering the pressure differential between cabin and suit pressure. It is impractical and unsafe to lower cabin pressure to a level of 28 kPa. Therefore, it is desirable to have an EVA suit that will operate at higher pressures and yet remain flexible for the astronauts to perform delicate tasks.

The Space Station and *Argo* will both use what are known as "hard suits". These are suits that have rigid mechanical joints and are made of a fairly rigid material. These suits are currently under development by NASA and are expected to have almost the same mass as the soft suits. The design that *Argo* will employ is a suit that can operate at 55 kPa. This will eliminate the need for a pre-breathe period, as the cabin is designed to be able to depressurize to 55 kPa. Hard suits also have another advantage in that they will provide more shielding for the astronaut and the electronics of the suit from radiation. With the higher levels of radiation experienced at GEO, suit shielding becomes very important. Even with this higher degree of protection, additional radiation shielding will be needed to allow for extended EVA's at GEO. *Argo* will be supplied with one suit for each astronaut.

The hard suits will contain the heat removal, carbon dioxide removal, and basic environmental control/life support equipment necessary for extended EVA.

7.5.3 Manned Maneuvering Unit (MMU)

The *Argo* will carry a manned maneuvering unit, a device that is already being used by the Space Shuttle. The MMU allows an astronaut full mobility and attitude control away from the spacecraft. It is propelled by cold gas nitrogen jets. Its ability to actually maneuver right up to an object will make it particularly useful for *Argo* missions such as repair, servicing or retrieval of a satellite. The range of the MMU currently being used is about 1 km. Later designs should have an extended range.

Along with the MMU, a device called the MMU Servicer will be carried by *Argo*. The MMU Servicer is a tele-robot, currently being designed, that will mate with the MMU and be able to go out and service or do minor repairs on an object. We expect that the MMU Servicer will be able to perform routine and minor EVA activities, thus allowing the crew to perform more intricate tasks.

7.5.4 EVA Procedure

The most important part of current EVA procedures, pre-breathing, has been eliminated with the use of hard suits by *Argo*'s crew. On *Argo* the cabin will be lowered to a pressure of 55 kPa while in transit from LEO to GEO. This makes the pre-breathe unnecessary. Once on station, EVA can begin almost immediately, if desired. While working on EVA, servicing can be made easier by designing all the nuts and bolts of satellites and platforms to be a uniform size. This allows servicing using a single size socket wrench and a general simplification of EVA tools. This system has already been implemented with the Hubble Space Telescope, with all of its nuts and bolts using a 0.172 cm standard diameter [ref. 7.12].

7.6 Power and Mass Distribution

7.6.1 Mass Distribution

A major constraint in the design of our STV is mass limitation. The mass of the crew systems and the radiation shielding have to be within an acceptable limit to minimize operation costs. The mass breakdown of the Life Support and Human Factor systems for the CSTV and the ASTV follow and are the same except where noted.

Subsystem	Mass (kg)
Atmospheric Control:	
Oxygen	21.8
Nitrogen and tanks	77.6
Atmosphere	10.0
Lithium hydroxide	26.4
Fans	6.0
Charcoal	0.6
Pressure regulators	3.0
Emergency pressurization	40.0
Heat exchanger/water separator	10.0
Interior Structure:	
Galley	10.0
Interior lights	10.0
Crew seating	40.0
Health maintenance	5.0
Toilet	75.0
Hand wash	2.0
Water plumbing/storage	3.0
Water pump packages (3)	30.0
Radiation Shielding	550.0 (CSTV) 650.0 (ASTV)
Storm Shelter (part of airlock)	575.0
Airlock	291.0

Food and Water:	
Food (with 7 day emer. supply)	20.0
Emergency water	8.0
Equipment and Personnel:	
Crew (2)	160.0
Tools	10.0
EVA suits (2)	364.0
MMU	160.0
MMU Servicer	60.0
RMS	350.0

Totals: CSTV mass = 2918 kg ASTV mass = 3018 kg

7.6.2 Power Requirement

The power requirement for all of our systems will be met with fuel cells. The power necessary for Air Revitalization System is 0.225 kW. This includes fans, pressure regulators, and the heat exchanger-water separator. The power requirement for interior equipment is 1.325 kW, which includes lighting, commode, water pumps and galley equipment. When the *Argo* is performing an EVA mission, an additional 0.5 kW of power must be available for the EVA equipment .

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Chapter 8

Logistics and Support

8.0 Summary

Logistics and Support is a unique group in Project Argo in that, unlike other groups, we do not design any one component or subsystem of the STV, but rather we think of the "logical accessories" to the STV that must be present for the entire program to work. Because the STV will be based at the Space Station *Freedom* (SSF), most of our concerns lie in the interaction of the Argo program with the Space Station. The specific areas under our consideration are: the impact of the STV on the Space Station, the delivery of the *Argo* and the fuel for it to the Space Station, docking of the STV at the Space Station, refueling and maintenance of the STV at the Station, and finally the storage and maintenance of satellites for repair at the Space Station.

When considering the impact of Project Argo on the SSF, we are mainly referring to any structural changes that must be made to the Space Station as a result of the presence of the STV. One such structural modification is the addition of an extra node to the habitat system to better facilitate the docking of the STV. Another structural impact is the building of hangars into which both the aeroassisted and the all-propulsive STV's will dock.

Before the STV can begin to operate in space, it must be delivered to the Station from Earth. This will be done using the Heavy Lift Launch Vehicle (HLLV) to transport the unfueled and unmanned STV to LEO. From there, the OMV from the SSF will be used to tow the STV to the Space Station and insert it into its hangar.

Once the STV arrives at the Space Station, it must dock so that the crew can enter the habitat. This concern is not a problem in the initial delivery of the *Argo* to the SSF because it will not contain passengers, but in subsequent manned missions of the *Argo*, transferring crew members to the Space Station will be an issue. To best facilitate this task, we will dock both designs directly to the habitat using the robot arm on the Space Station, thus allowing the crew to enter the living compartment of the Space Station without performing a spacewalk.

A major concern for our group is the refueling needs for the STV at the Space Station. We plan to directly refuel the STV while it is docked in its hangar at the SSF. The fuel tank will be delivered to the Space Station in a manner similar to that used to bring up the STV. The OMV will place the permanent "Depot" tank on the middle spar of the SSF and this large refueling tank will occasionally be resupplied with fuel by less expensive "Transport" tanks brought to the Space Station by the HLLV.

The *Argo* will need maintenance and repairs during its lifetime at the Space Station, so we have addressed this issue as well. As a matter of fact, the decision to have a hangar for the STV was strongly supported by the need for a surrounding structure to enable successful repairs on the

STV. Computer and human checkout before and after repairs is essential to the proper maintenance of the STV.

Finally, since our mission definition calls for the repair of satellites at the SSF if needed, we have provided for facilities to accommodate this task. There will be special repair hangars attached to the framework of the SSF to hold any returned satellites. The mobile robot arms on the Space Station will transport the satellites to these hangars.

Looking at the different areas under our jurisdiction in the total design effort, one can clearly see that our group is atypical of the others. Though we do not concentrate on the design of any one aspect of the actual STV itself, except the docking interface, we are intrinsically concerned with the "common sense" and supporting aspects of Project Argo which will make the entire program feasible. Thus our name, Logistics and Support.

8.1 Impact on Space Station

The design of the Space Station on which we are basing all of our work is that of the dual keel design. This design is shown in Figure 8.1.

On this design for the space station, there are many possible locations to dock the STV. The decision was made to have the STV dock directly to the habitat of the Space Station. The main reason behind this decision was to avoid the necessity of the astronauts conducting an EVA from the STV to the Space Station between every mission. Referring back to Figure 8.1, the STV can then dock either to the right or to the left of the habitat, under the main boom of the Space Station. The idea of docking on the back end of the habitat was rejected because of the fact that the STV will use berthing to dock. The berthing arm is located along the front of the boom, and therefore docking to the front of the habitat will be much easier. This docking procedure will be discussed in Section 8.3.

The next constraint for docking is the fact that the Space Shuttle also has to dock to the front of the habitat. The existing plan is that the Shuttle will dock to the front of one of the nodes. Therefore, to keep as much space as possible between the Shuttle and the STV, we plan to dock the STV onto the side of the other node. This leaves approximately 12.6 meters between the habitat and the boom of the Space Station for the STV. The STV cannot overlap the boom because it will interfere with the motion of the Mobile Servicing System (MSS) that travels along the dual keels. In our design, we have placed the STV docking on the right node, which leaves the left node open for the Shuttle.

By docking the STV on the right node, the possibility of using this node as a backup docking port for the Space Shuttle has been eliminated. To provide for the possibility of such an emergency situation occurring, it is recommended that another node be added to the left side of the left node to fill this purpose. But, this is not a major consideration since future plans show that more modules and another node will be added to the left side of the habitat. This future node will act as the Space Shuttle backup docking port.

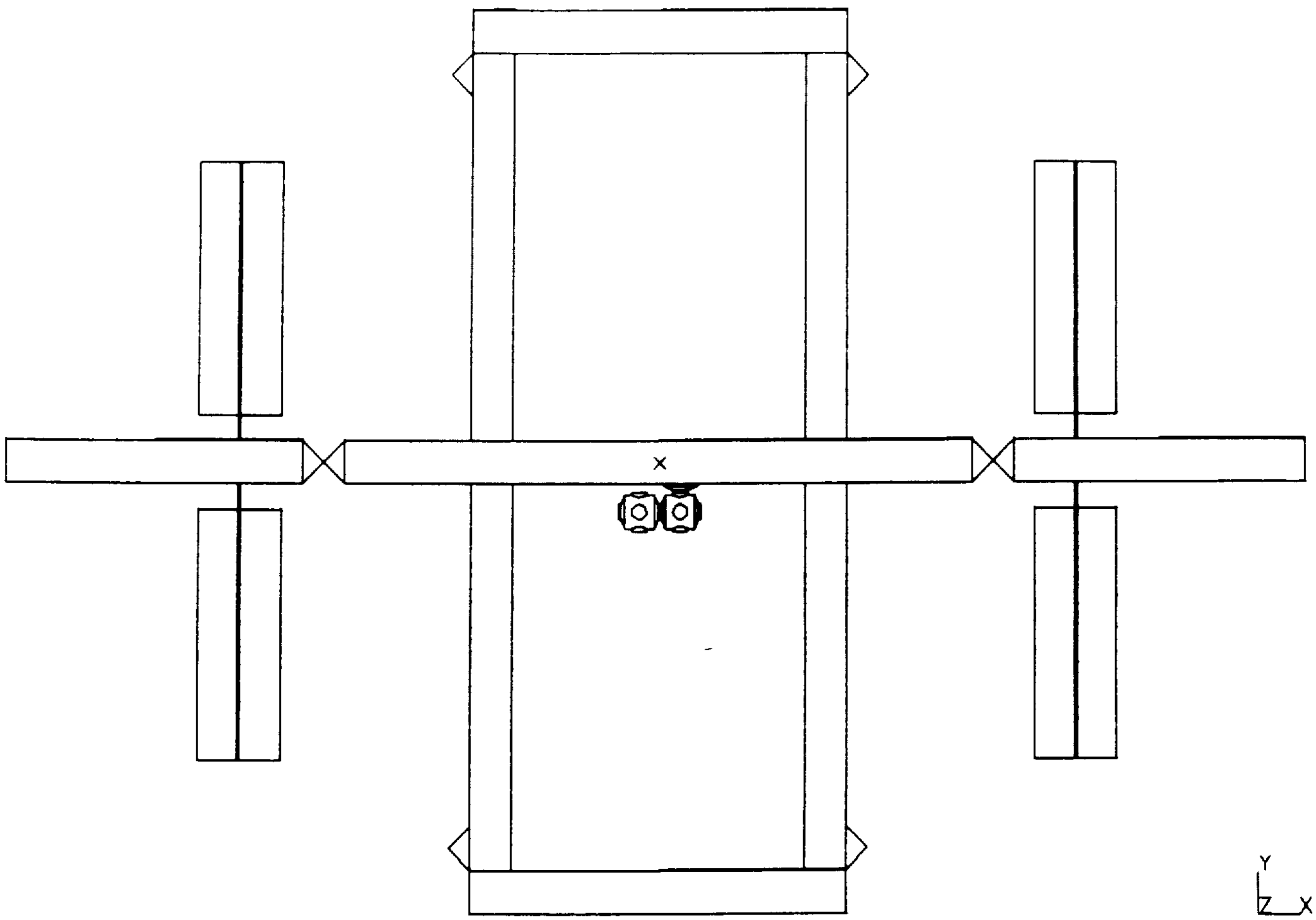

Figure 8.1 Dual Keel Design of Space Station

While stationed at LEO, the STV will be subjected to a series of environmental hazards such as repeated thermal cycles from approximately 95 degrees Celcius to -130 degrees Celcius, ultra violet radiation, ultra high vacuum, and collisions with micrometeoroids and other space debris. Although the STV's structure will provide some protection from these hazards, the life of the STV can be greatly extended by providing a hangar. This hangar will also have all the necessary provisions for the STV maintenance: lighting, work platforms, and a storage area for spares and equipment. It will also provide basic thermal and radiation protection for the EVA maintenance crew.

A pressurized hangar will not be needed since the replaceable units of the STV will be designed to be easily replaced by astronauts in pressure suits. These units are called space removable units (SRU's) and are further explained in Section 8.5.2. Although the SRU's must have mounting provisions and quick-disconnect electrical and fluid connections, the penalties of a pressurized hangar are much more significant. For example, a large amount of electrical energy would be needed to recover the pressurized atmosphere every time the hangar is opened or else the atmosphere would be lost. In addition, there is a risk of exposing residual propellants from the engine or other STV components to a combustive environment. Thus, we selected the non-pressurized hangar for Project Argo.

8.1.1 The Hangar Walls

Because of the STV's large surface area and its long duration in orbit, micrometeoroid and orbital debris hazards are of major concern. Previous studies show that both protection from these hazards and light mass are simultaneously optimized by using a multi-layer structure. Table 8.1 shows some of the different combinations of layers we considered for our hangar and their relevant properties. From this data we can see that Aluminum sheets decrease the probability of penetration but they also increase the mass and the cost of the wall. Thus, we decided to use 90 layers of multi-layer insulation (MLI) in the walls of our hangar. This provides a high probability of no penetration while keeping the wall's mass and thickness fairly low.

Hangar wall type	Wall thickness (cm)	Wall mass (kg)	Wall cost (M$)	Probability of no penetration
30 layers MLI	.0066	170	1.55	.9547
60 layers MLI	.0132	341	3.12	.9725
90 layers MLI	.0198	511	4.67	.9784
.016 Aluminum sheet + 30 layers MLI	.0472	1220	11.15	.9886
.0374 Aluminum sheet + 30 layers MLI	.1016	2623	23.96	.9924
.0774 Aluminum sheet + 30 layers MLI	.2032	5246	47.93	.9935

Table 8.1 Hangar Wall Design Comparison [ref. 8.1]

While the MLI as a whole prevents micrometeoroid and debris penetration, the individual layers are responsible for the protection from other environmental hazards. The materials chosen to accomplish these tasks are aluminized Teflon for solar radiation shielding, and aluminized Kapton for radiation and thermal insulation.

The thickness of MLI required for environmental protection is also shown in Table 8.1. The dimensions of the walls will be affected by the necessary maintenance provisions that will be stored on the walls. These include fixed lighting, astronaut foot restraints, inspection cameras, door mechanisms, and functional test and propellant umbilicals. The sizes of the walls for the different hangars are described in more detail in Sections 8.1.3 and 8.1.4.

8.1.2 Other Common Features in the Hangar Design Concept

To facilitate the description of the hangars and avoid being redundant we will mention some other characteristics that both the all-propulsive hangar and the aeroassisted hangar share before entering their individual descriptions.

Both hangars provide a movable work platform 20 cm thick and 1 m wide. These platforms are equipped with boot supports, lighting, and inspection cameras. The platform in the CSTV hangar will move from the top to the bottom of the vehicle giving access to any area that needs maintenance while the platform in the ASTV hangar will give access to any area on the outer shell of the aerobrake for the replacement of tiles. The movement of each platform will be controlled by a mechanical device operated from an astronaut on the platform. They will be locked into place by latches in the walls of the hangar. These latches will be located on ten equally spaced stations (about 2 m between stations) along the CSTV hangar walls and on the junctions of the armadillo door sections for the ASTV (providing fourteen platform stations).

Each hangar has a storage compartment for spares and maintenance tools. This compartment will only house those space removable units which need frequent repair as determined by failure rate tests. The rest of the components will be brought from Earth by the Shuttle as needed.

As explained at the beginning of this section, the hangar must fit between the Space Station's trusses and care should be taken not to interfere with the docking of the Space Shuttle. Because of these spatial constraints, the hangar's shape will be made to fit the shape of the STV.

The internal structure of the hangars must provide at least 2 meters of space between the the STV and the walls for EVA and spare mobility. This 2 meter clearance was determined by the dimensions of the engines, the largest unit that will be replaced in the Station.

Both hangars can be easily launched from Earth using either the Space Shuttle or the HLLV. Both hangars will require some EVA assembly by astronauts.

8.1.3 Hangar Configuration for the CSTV

The hangar for the all-propulsive STV is shown in Figure 8.2. Four doors will be opened on three of its longitudinal sides for the docking of the STV. For the design of the these doors, we decided to use extendable/retractable protective blankets (ERPB) which operate similar to deployable solar arrays [ref 8.10]. The doors can be mechanically extended and retracted as needed from controls inside the Space Station. The remaining five longitudinal walls will contain fixed lighting, functional test umbilicals, propellant hoses, and six rails for the movable work platform.

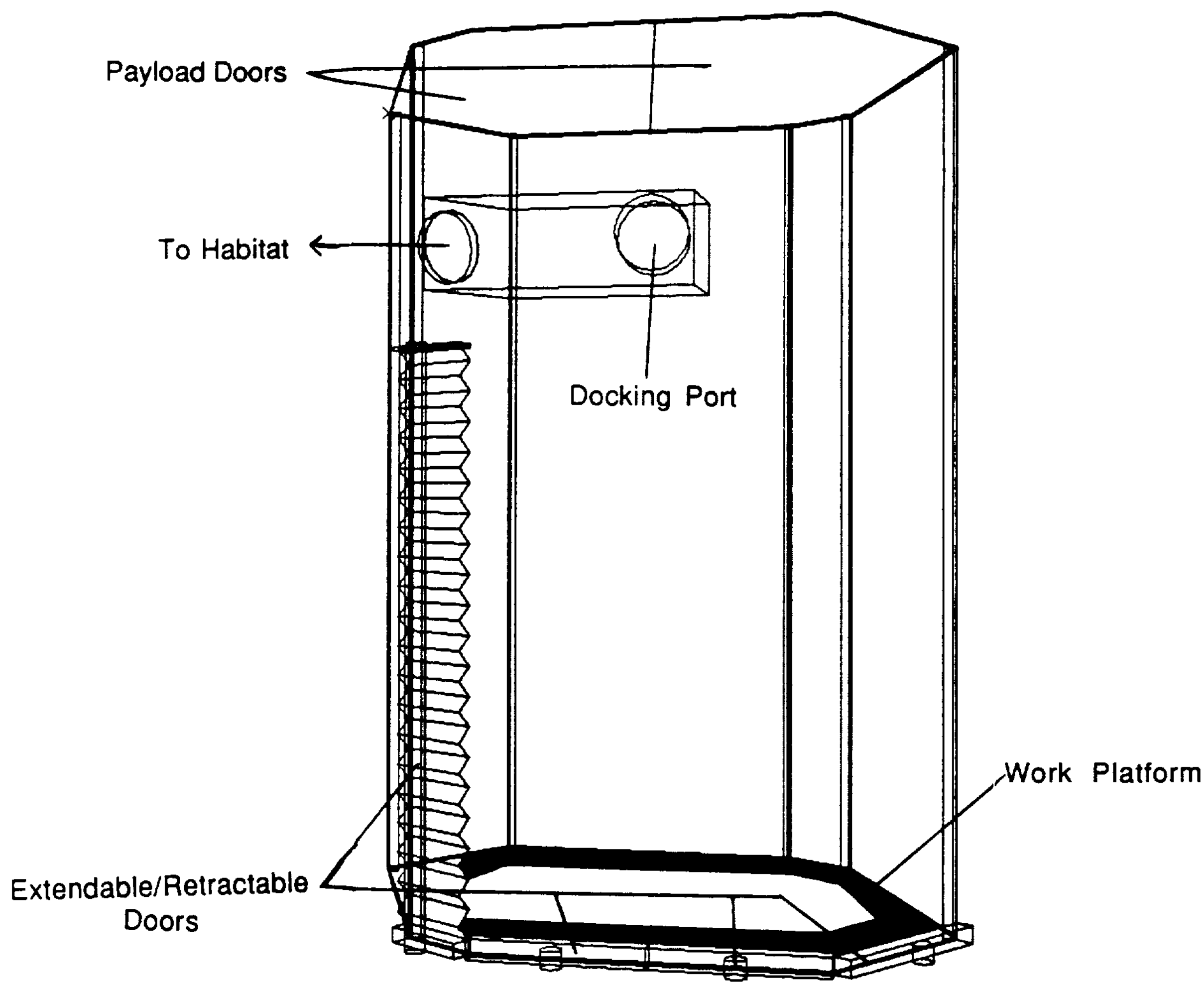

Figure 8.2 CSTV Hangar

The top of the hangar will serve as a door for payload integration. The opening and closing of this door will be controlled from inside the Space Station. The operation of the door is shown in Figure 8.3. The door will remain opened while the STV and payload unit is being docked or undocked. Any payload wider than this door will have to be integrated outside the hangar using the Station's MRMS and MMS, and the STV RMS.

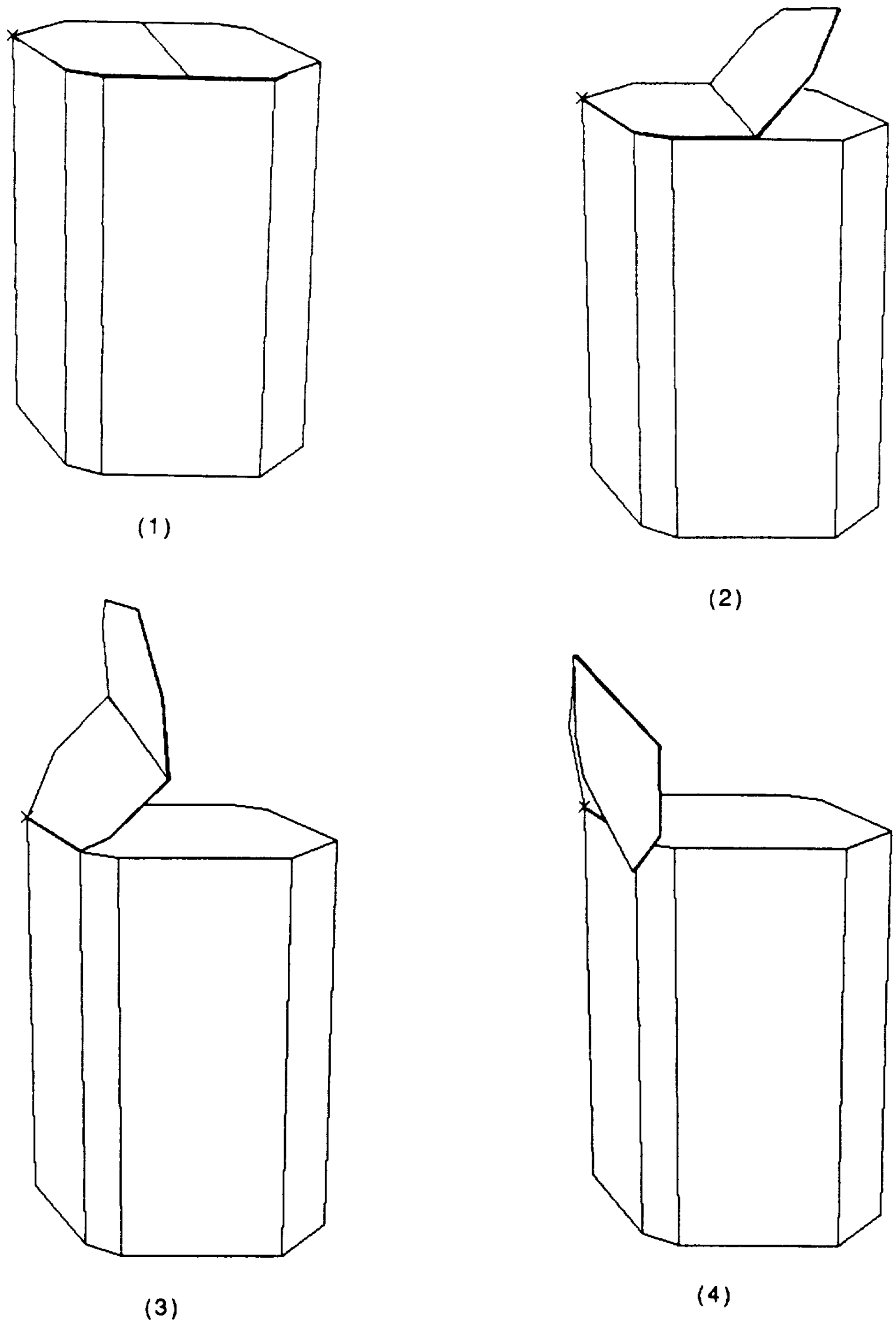

Figure 8.3 CSTV Hanger Payload Door Operation

The storage compartment of this hangar is located near the top, and the work platform will be stored at the bottom of the hangar while it is not in use.

The position and orientation of the CSTV hangar on the Space Station is shown in Figure 8.4.

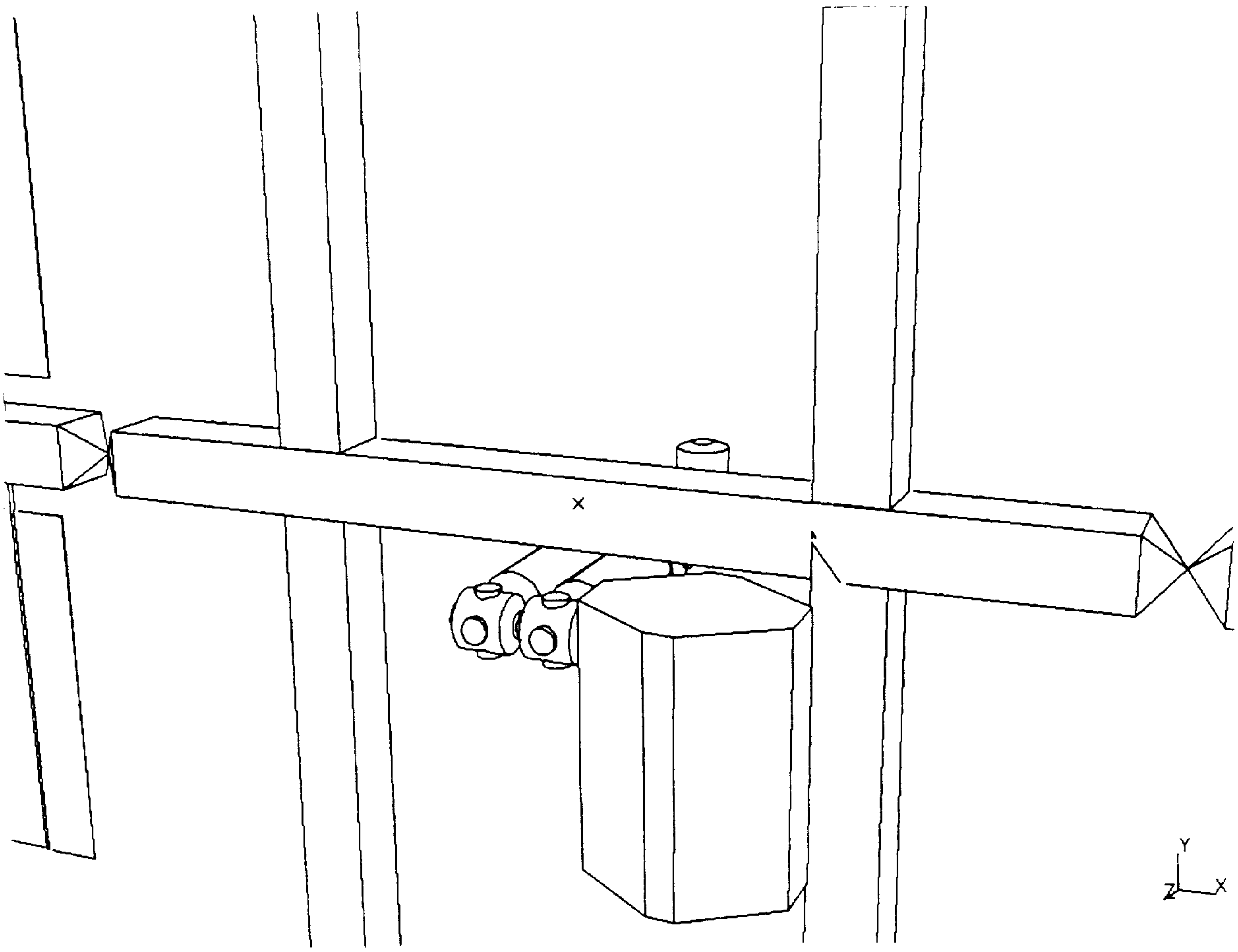

Figure 8.4 CSTV Hangar on Space Station

8.1.4 Hangar Configuration for the ASTV: The Armadillo Hangar Design (AHD)

The Hangar Configuration for the ASTV is shown in Figure 8.5. This hangar consists of a circular wall 20 m in diameter and 10 cm wide which will be connected to the right node of the Space Station. This wall will have permanent lighting, inspection cameras, a tunnel connecting the Space Station docking port to the STV, and a sliding door, similiar to the doors on the CSTV hangar, for payload integration. The storage compartment will be located adjacent to this wall.

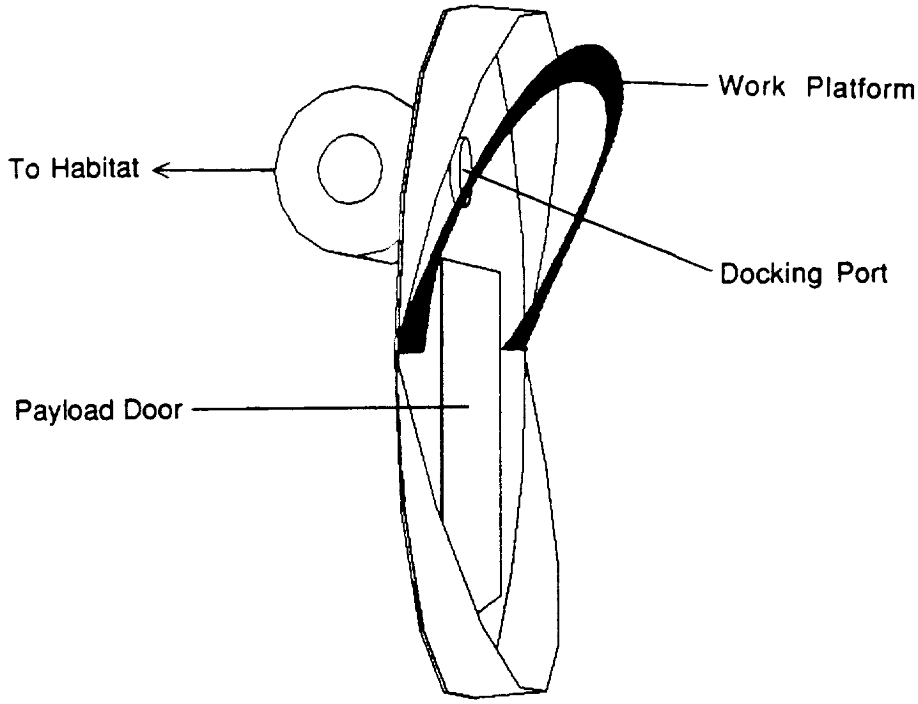

Figure 8.5 ASTV Hangar

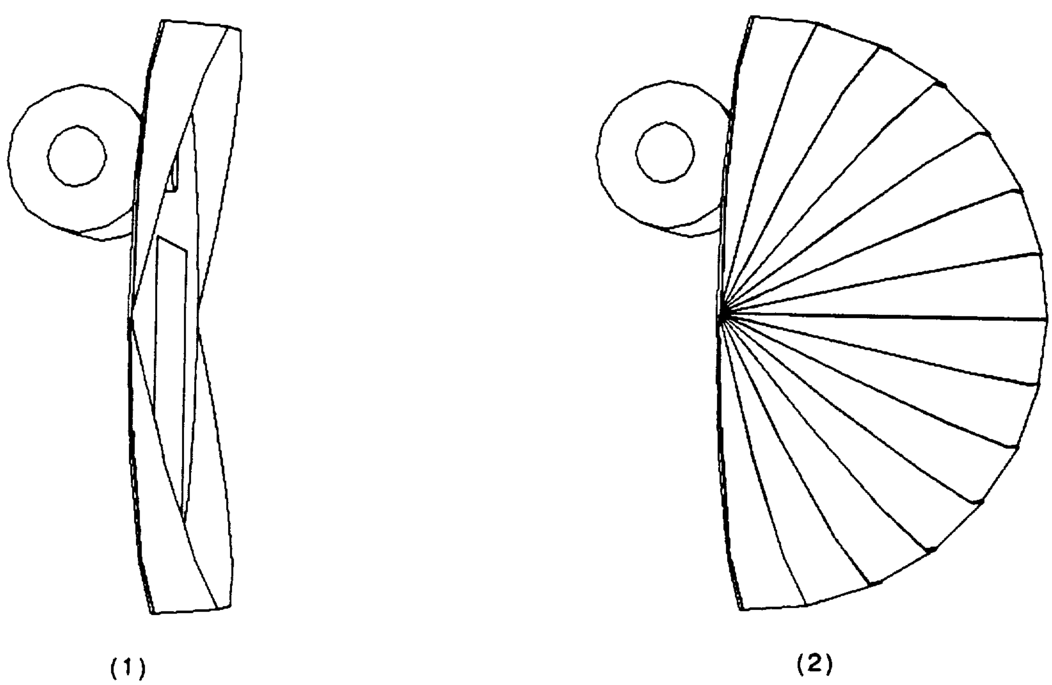

Figure 8.6 ASTV Hangar Door Operation

The design and operation of the doors of this hangar are shown in Figure 8.6. Each door consists of seven sections attached by two hinges to the main circular wall and by latches to each other. When expanded the two doors form a spherical structure which resembles an armadillo's armor, hence the name of Armadillo Hangar Design. Both the doors and the work platform will be operated by mechanical devices on the hinges controlled from inside the Space Station.

The position and orientation of the ASTV hangar on the Space Station is shown in Figure 8.7.

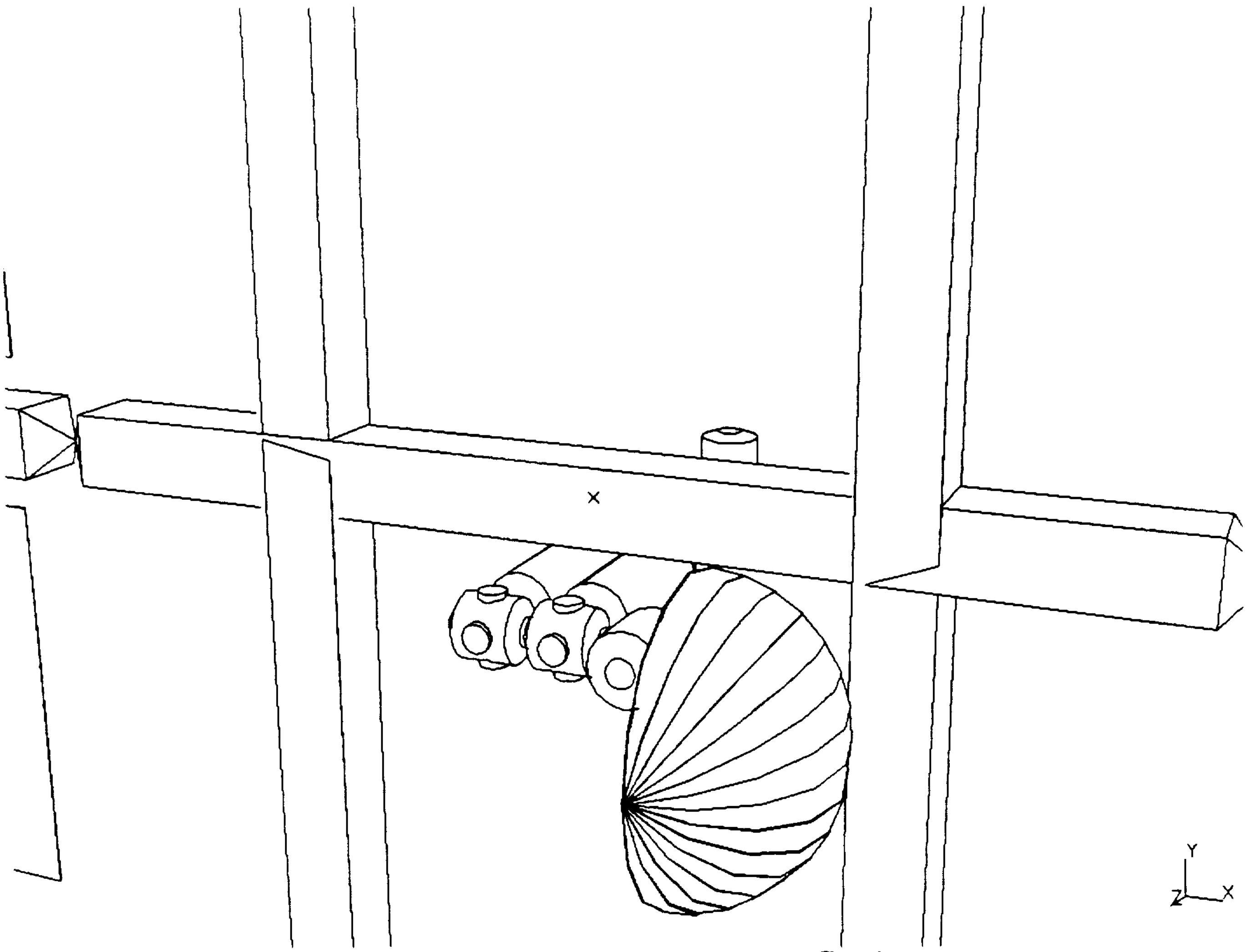

Figure 8.7 ASTV Hangar on Space Station

Another main impact on the Space Station is the placement of the fuel tanks needed to fuel the STV. This impact will be discussed in Section 8.4.

8.2 Delivery of STV to the Space Station from Earth

Presently, the only *existing* vehicle that can be considered to deliver the STV to the Space Station is the Space Shuttle. But, with a maximum lift weight of 29,000 kg and a cargo bay with dimensions of 18.29 meters in length and 4.57 meters in diameter, the Space Shuttle will not be sufficient for the job. Therefore, a Heavy Lift Launch Vehicle (HLLV) will be needed to perform the delivery. Ronald Toelle of NASA has presented a design of an HLLV with the following payload characteristics:

Mass:	136,000 kg
Diameter:	15 m
Length:	60 m

Many of the other designs for an HLLV presented have approximately the same payload characteristics as above, therefore we will go with Toelle's design [ref. 8.2]. The projected date for the first launch of the HLLV is 1995, which is before the STV projected launch.

The HLLV will place the STV in an orbit approximately 50 km below the Space Station. Since the STV will be launched without fuel, the OMV will grapple the rear RMS Grapple Fixture and bring the STV close to the Space Station. The MRMS will then grapple the front RMS Grapple Fixture and berth the STV to the station. The berthing procedure is described in the next section.

8.3 Docking of the STV at the Space Station

8.3.1 Berthing versus Docking

We have investigated two methods of bringing the STV docking interface into contact with the mating Space Station docking interface. The first method is berthing. Berthing entails using the Mobile Remote Manipulator System (MRMS) based on the Space Station to grapple the STV and bring it into contact with the Space Station docking interface at a very small velocity. The second method is soft docking (or docking). Docking involves the use of STV RCS thrusters to translate, via low-Z maneuvers, directly into contact with the docking interface of the Space Station [ref. 8.3].

We have chosen berthing as our primary method of mating with the Space Station. Berthing has several advantages over soft docking. A few of these are:

1) Berthing velocities are typically an order of magnitude smaller than docking velocities. This decreases both the stresses involved in the docking process and the disturbances of space station experiments, antennae, etc. [ref.8.4].

2) Berthing positional errors are also much smaller than those incurred during docking [ref. 8.4].

3) If the low-Z RCS thrusters fail while making an approach to the Space Station in an attempt to soft dock, the STV will have to use an extensive blast in the direction of the Space Station in order to avoid collision. This is undesirable because of the corrosive nature of the exhaust gases. This emergency situation is not possible when using berthing [ref.8.3].

4) There is a significant RCS propellant savings by using the berthing method.

8.3.2 Structures and Mechanisms

Docking / Berthing Interface

The docking/berthing interface we employ is an androgynous interface which is compatible and identical to the docking/berthing interfaces on both the Space Shuttle and Space Station. Therefore it is possible to dock at any of several ports on the Space Station when necessary. The configuration shown in Figure 8.8 is similar to the one we employ. The inclusion of several connections located directly in the face of the interface allows for the replenishment of the STV consumables and utilities while docked at the Space Station. Connectors are provided for electrical power, fiber optics data busses, air intake and exhaust, drinking water, waste water, oxygen supply, and nitrogen supply. All of these utilities will be taken directly from the Space Station supply [ref. 8.3].

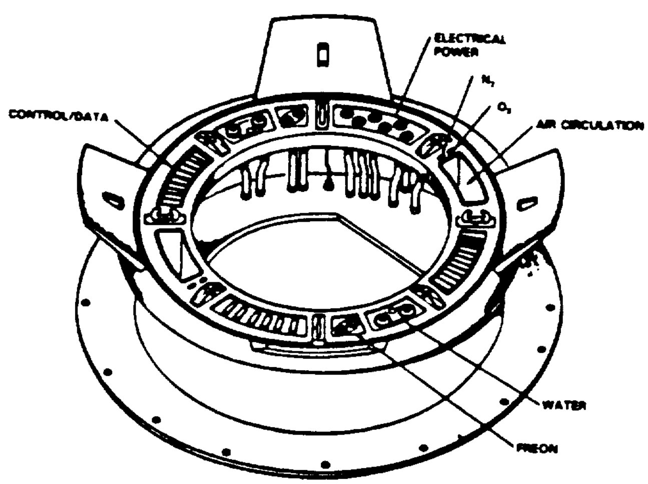

Figure 8.8 Docking/Berthing Interface [ref. 8.3]

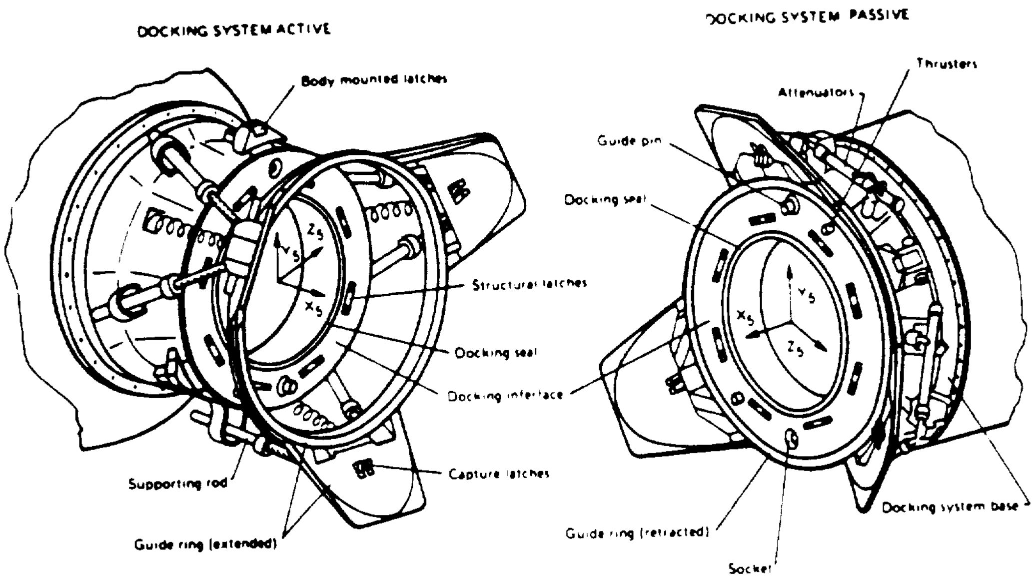

Figure 8.9 Universal Docking Mechanism [ref. 8.5]

Docking / Berthing Mechanism

The docking/berthing mechanism used on the Space Station, Space Shuttle and STV is similar to the Universal Docking Mechanism (Figure 8.9) used in the Apollo-Soyuz Test Project (ASTP). The docking/berthing mechanism is designed primarily for use in berthing, but, as the name implies, can also be used in docking. The dimensions of the Docking/Berthing Mechanism and Interface are as follows [ref. 8.3]:

Outer diameter of base ring	2.25 m
Tunnel diameter	1.27 m
Hatch diameter	1.27 m
Depth (guide extended)	0.38 m
Depth (guide retracted)	0.23 m

Docking / Berthing Hatch

The docking/berthing hatch and the airlock hatch will both be D-shaped, as shown in Figure 8.10, with a diameter of 1.27 meters. This is the conventional hatch used on both the Space Shuttle and the Space Station and is large enough to allow passage of an astronaut in an EMU [ref. 8.3].

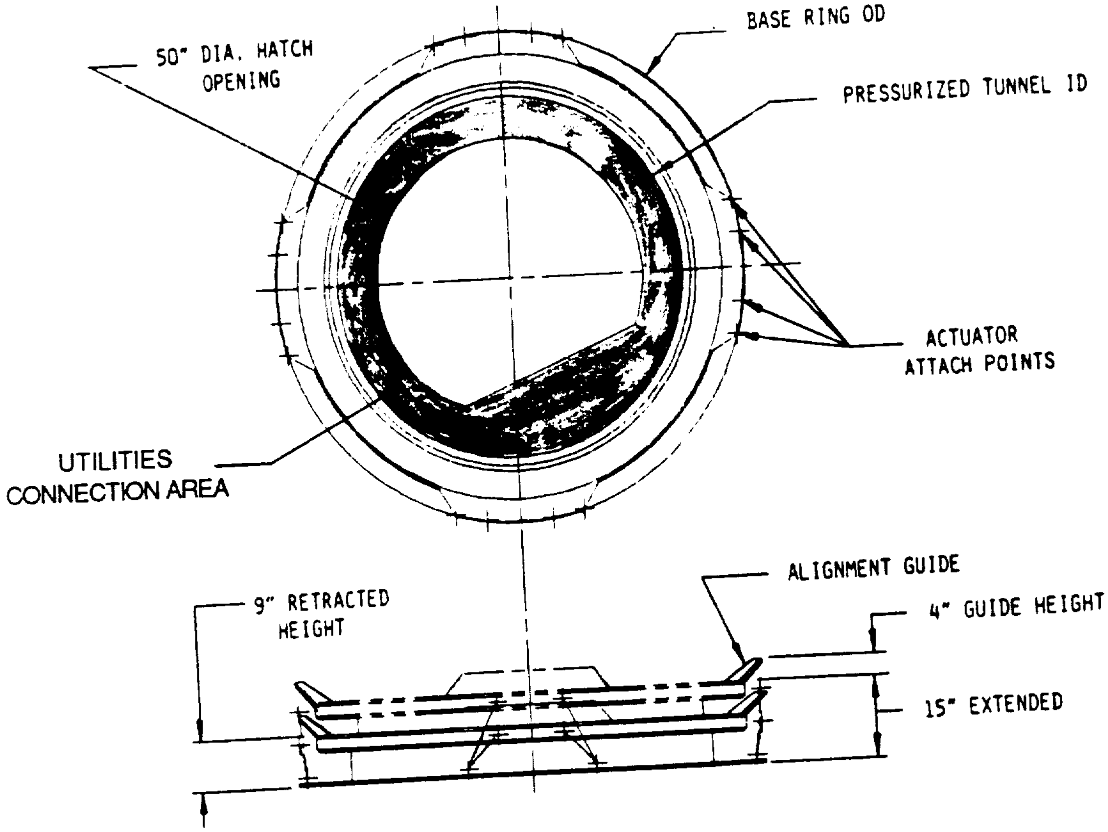

Figure 8.10 Docking/Berthing Hatch [ref. 8.3]

Refueling Probe

A refueling probe will be located next to the docking/berthing interface on the STV. This probe will automatically connect with its mate, similarly located on the Space Station, when the STV berths. This will allow the STV to be refueled while at the Space Station without the need of an EVA or the MRMS.

The Mobile Remote Manipulator System (MRMS)

The MRMS used to berth the STV will be located on the center truss of the Space Station. The MRMS is a variation of the RMS currently used on the Space Shuttle. The Space Station MRMS will be able to manipulate the STV at small velocities and will be able to track and grapple the STV once it is in berthing range, using attached sensors and television cameras as guides. Use of the MRMS during berthing will allow only small velocity and positional errors [ref. 8.3].

Guidance and Tracking Mechanisms

Up to four types of sensors can be employed during the rendezvous and berthing or docking [ref. 8.6]. They are:

1) Laser proximity sensors
2) Closed-circuit television
3) Crew Optical Alignment System (COAS)
4) Rendezvous Radar

Each port on the Space Station will include a laser sensor system which provides precise measurements of the range, rate, angular position, and attitude of the approaching spacecraft. Small, passive retro-reflectors are required on docking spacecraft in order to determine attitude data [ref. 8.3].

8.3.3 Operations

Rendezvous and Capture

Once the STV is within berthing range (i.e. within reach of the MRMS or approximately 18 m), the MRMS, controlled from within the space station, will be used to grapple the front RMS Grapple Fixture of the STV. This procedure will be aided by television cameras mounted on the MRMS arm. The MRMS will then apply joint braking torques to cancel the residual relative velocity of the STV [ref. 8.3].

Berthing

Once the STV has been grappled and braked, the MRMS will bring the STV into contact with the Space Station at a very small velocity. The laser proximity sensors located on the Space Station docking port will enable accurate positioning.

The alignment guide height of .10 meters can tolerate lateral positional errors of approximately ±.08 meters in combination with yaw and pitch misalignments of ± 5°. Roll misalignment of ± 5° can also be allowed. The optimal docking/berthing velocity is ≤ 0.03 m/s. However, the docking/berthing ports can withstand a dock at 0.3 m/s [ref. 8.3].

Once the STV is securely docked, the hangar doors can be closed.

Separation from the Space Station

Separation begins with a small separation rate (~ .06 m/s) using the MRMS to gain distance from the port. The direction of separation will be directly away from the docking port until clear of any obstructions and then directly away from the Space Station to maximize distance. After coasting approximately 15 minutes, the STV will be out of the assumed explosion range and will be able to perform a larger separation maneuver. At this point, the STV is considered to be separated [ref. 8.3].

Contingency Operations

If, for any reason, the Space Station MRMS is inoperable, the STV will be able to use its RMS to grapple the Space Station and control the berthing operations.

If, for any reason, both the Space Station MRMS and the STV RMS are inoperable, the STV will be able to dock using low-Z maneuvers at a docking/berthing interface which is clear of obstructions.

If, for any reason, the STV docking/berthing interface or mechanism, or the docking/berthing hatch is inoperable, the Space Station MRMS will grapple the STV and the crew will be able to perform an EVA to transfer to the Space Station.

8.4 Storage of Fuel at the Space Station and Refueling of STV

8.4.1 Design of Fuel Tanks

As the STV, STS, OMV, and SSF all use cryogenic propulsive systems, an orbital cryogenic liquid storage facility will be an essential part of the success of the U.S. space program. The basic requirement for such a storage facility is that it must be able to contain a large supply of the necessary cryogenics safely for an extended period of time in the harsh conditions of outer space. It must be able to fuel the STV quickly and efficiently, and the facility itself must be able to be refilled by ground launched transport tanks.

The primary concern of any cryogenic fluid storage system which must operate in a reduced gravity environment is that of fluid/vapor separation. Because of heat transfer caused by external heating and pumping, some of the super-cooled liquid being stored can turn to vapor. In a gravity field this does not pose a problem as the vapor will rise and separate from the liquid. However, in a reduced gravity situation, this separation does not occur. This vapor-liquid mixture is difficult to pump, thus complicating fluid transfer. To solve this problem, the concept of capillary acquisition is used. This method makes use of the liquid retaining properties of fine mesh to absorb the fluid , thus separating it from the vapor. This concept will be discussed in greater detail in Section 8.4.3.

Another concern regarding fluid storage is that of boiloff. The vapor which has formed in the storage tank must be removed, thus raising the question as to what to do with it. For our purposes we have considered two different systems for dealing with boiloff: Purely Passive or Total Reliquefaction [ref 8.7].

Both systems have a 115,000 kg of propellant capacity and route hydrogen boiloff through vapor-cooled shields on both the hydrogen and oxygen tanks. However, Concept 1, as shown in Figure 8.11, stores hydrogen and oxygen boiloff in high pressure accumulators, whereas Concept 2, as shown in Figure 8.12, reliquifies all boiloff and returns it to the tanks, thus recycling the vapor.

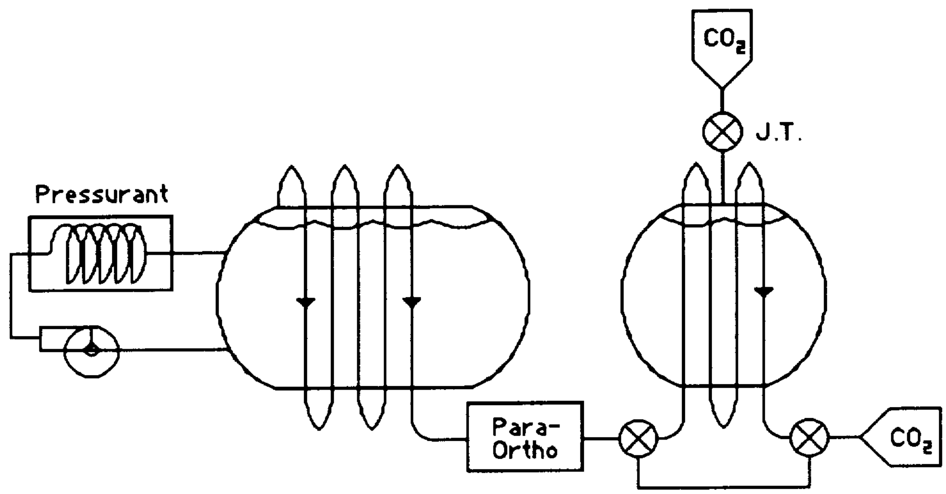

Figure 8.11 Concept 1: Purely Passive System [ref. 8.11]

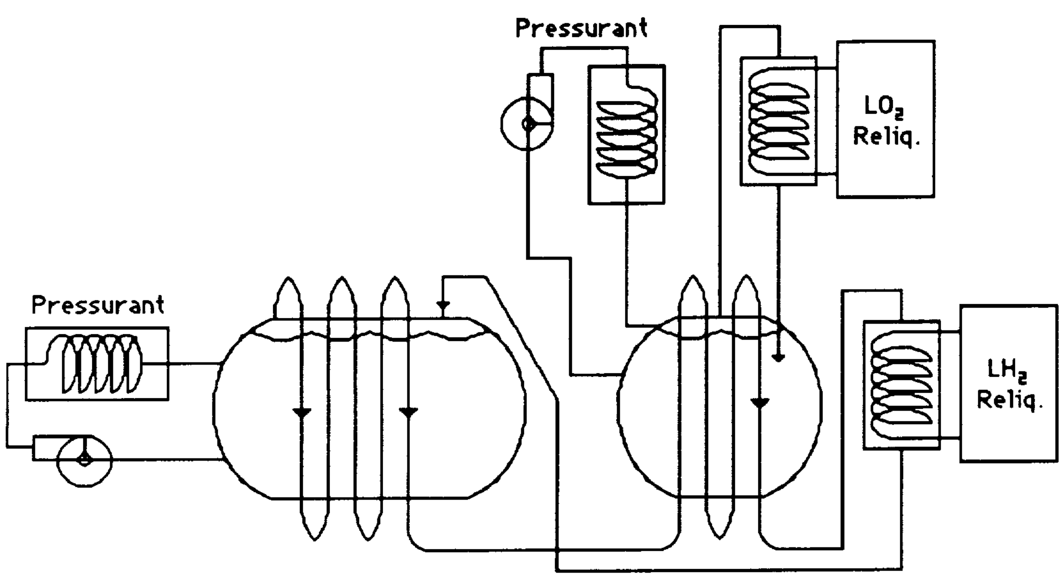

Figure 8.12 Concept 2: Total Reliquefaction [ref 8.11]

Both concepts use capillary acquisition to transfer fluid and are capable of efficiently fueling the STV, as well as being refueled by a ground launched transfer tank. Also, both concepts make use of multi-layer insulation (MLI) for thermal protection in space and are capable of storing fluid for an extended period of time. Concept 1 has a capacity for 90 days worth of hydrogen boiloff. This boiloff could be used to power the SSF attitude control thrusters. Oxygen boiloff can be used for breathing air onboard the SSF.

The two concepts were evaluated and compared on such factors as safety, reliability, cost, and power requirement; the results of which allowed us to choose between the two designs.

Concept Design Selection

<u>Safety</u>: Concept 2 does not require storage of boiloff and on this basis has a slight safety advantage.

<u>Reliability</u>: Concept 2 has refrigerators that are being developed to provide five to seven years of maintenance-free operation. Concept 1 has a high pressure boiloff subsystem with a 20.7 MPa multistage compressor train that is expected to have similar reliability and operating life to that of the refrigerators.

<u>Cost</u>: Concept 2 has a lower initial operating cost than Concept 1 primarily because it does not require development of the large high pressure boiloff accumulators that are needed to collect propellant boiloff in Concept 1.

<u>Power Requirement</u>: Concept 2 requires nearly ten times the electrical power of Concept 1. Concept 1 requires 146 kW-hr/mo while Concept 2 uses 1,430 kW-hr/mo [ref 8.7]. This is currently more power than the Space Station is designed to be able to produce.

While safety, reliability, and cost factors for concepts 1 & 2 are not significantly advantageous in one design over the other, power requirement is the deciding factor. The electrical power requirements of Concept 2 makes Concept 1, the Purely Passive facility, the design choice for the fuel tanks. Figure 8.13 shows the design of the storage tank.

The size of the storage tank is limited only by the lifting capacity of the HLLV, which will be used to launch the tank. As shown previously, the maximum payload weight that the HLLV can carry to the Space Station is 136,000 kg. Based on the fuel tank design by Schuster, Bennett, et. al., the weight for our storage tank was found to be 12,600 kg [ref 8.7]. Knowing the HLLV limit and the weight of the storage tank, the capacity of the tank has been set at 115,000 kg.

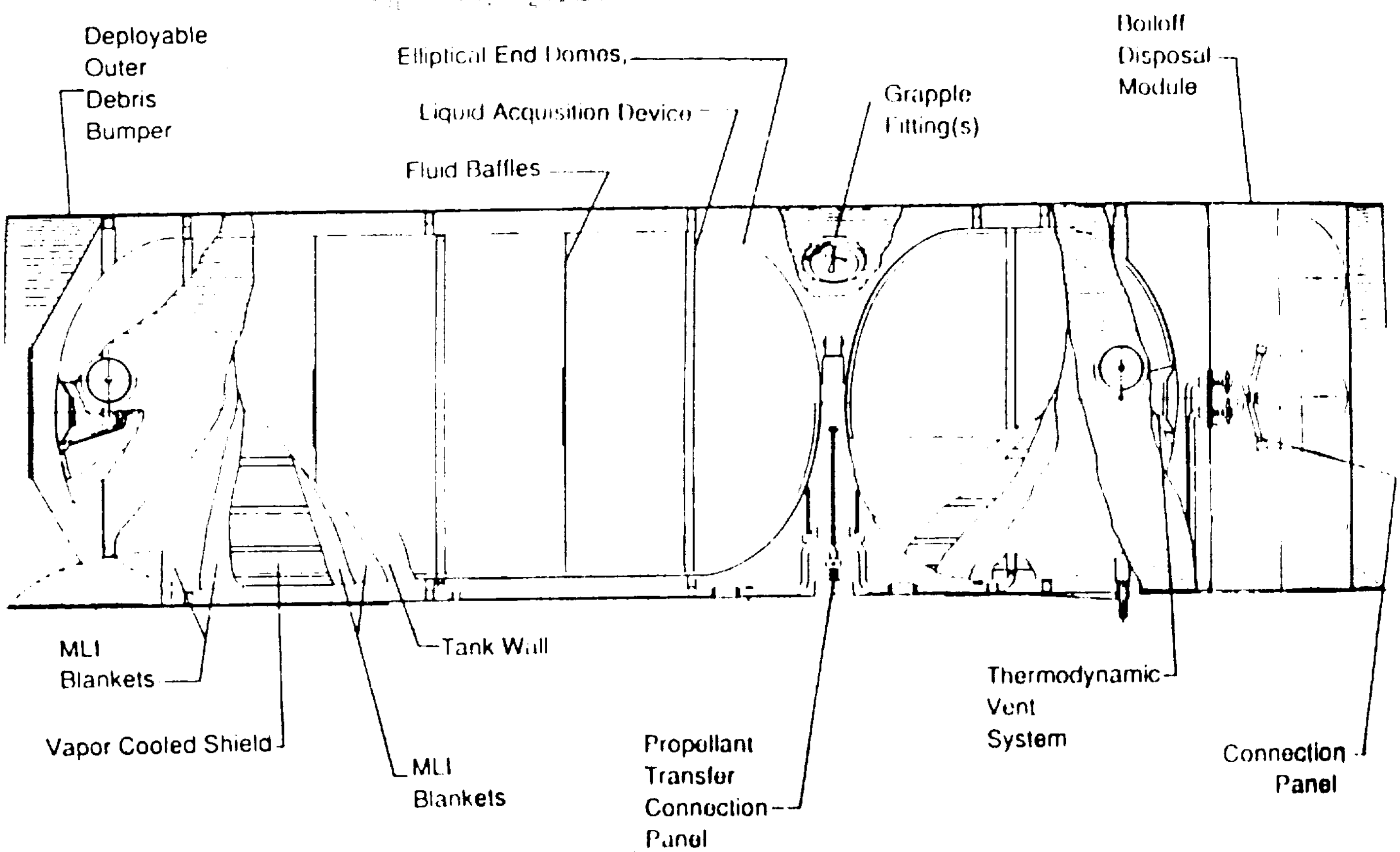

Figure 8.13 Storage Tank Design [ref. 8.11]

If the tank was delivered to the Space Station empty, then the fuel tank could be larger, but we feel that 115,000 kg of fuel is sufficient for permanent storage at the Space Station. With this mass constraint and knowing that the mass ratio of oxygen to hydrogen needed for the fuel of the STV is 6.0, the volumes of the oxygen and hydrogen tanks in the storage tank are found to be 86.47 cubic meters and 234.66 cubic meters, respectively. Since we are planning on docking the storage tank on top of the middle truss, we have set a limit on the diameter of the storage tank at five meters, which is equal to the width of the truss. This limit was chosen so that the tank is not overlapping the truss. The diameter of the internal oxygen and hydrogen tanks were then set at 4.5 meters, which allows for the outer layer and the plumbing of the storage tank. With these dimensions, the oxygen and hydrogen tanks are found to be 5.44 meters and 14.75 meters long, respectively. Providing for space between tanks as well as the boiloff disposal module, the length of the storage tank will be 24.2 meters.

Since the storage tank will be a permanent part of the Space Station, it must be protected from radiation and debris. The outer layers of the storage tank will serve the same purpose as the hangar does for the STV and will consist of layers of various types of materials to protect the inner workings of the tank. Based on Table 8.1, the same type of protection will be used for the storage tank as for the hangar. The storage tank will have 120 layers of MLI, compared to 90 layers for the hangar, as an added safety factor for the fuel. The outermost layers of Teflon and Kapton will provide vapor-cooled shielding of the tank [ref 8.1].

From the above statements, the storage tank characteristics are:

Capacity:	115,000 kg
Dry Weight:	12,600 kg
Size:	24.19 m x 5 m
Electrical Power:	146 kW-hr/mo
Fluid Transfer Rate:	1,515 kg/hr

When the storage tank is empty, a transport fuel tank will be needed to refuel it. The transport tank will carry 115,000 kg of fuel and will also need to use capillary acquisition to transfer the fluid propellant. But it will not need to have the thermal insulation or other equipment which would be required for long term storage of cryogenic fluids, and therefore, will be lighter than the storage tank. The transport tank design is similar to the design of the storage tank as shown in Figure 8.13, except for the fact that it will not need a boiloff disposal module.

8.4.2 Delivery of Fuel to the Space Station

The storage fuel tank has been designed with the constraint that it will be launched fully fueled by the HLLV. The delivery process will be similar to that of the STV, with the HLLV placing the storage tank in an orbit approximately 50 km below the Space Station. The process of bringing the storage tank from this lower orbit to the Space Station is slightly more difficult than that of the STV itself because of its much larger weight. With the help of Norman Brown at NASA Marshall Space Flight Center, it was found that the OMV can deliver the 136,000 kg tank to the Space Station if the following launch procedure limitations are made:

1) The orbit of the tank must be no more than 50 km below the Space Station.
2) The orbit must be in the same plane as the Space Station so that the transfer to the Station does not involve a plane change.

If these limitations are made, the OMV can perform the transfer, using approximately 2000 kg of its available 3000 kg of fuel. The extra fuel can provide for a slight plane change if the orbits are not matched, but the size of the plane change that is possible with the extra fuel is minimal: approximately 0.2°.

Once at the Station, the OMV will maneuver the storage tank above the main truss, where it will dock the fuel tank into its cradle with the help of the MRMS. Spring activated bolts in the arms of the cradle will then be activated to hold the tank permanently in place.

Once the storage tank is locked onto the Space Station, the fuel lines have to be connected. The main fuel line will run from the propellant transfer connection panel (see Figure 8.13) to the hangar. There will also be a secondary fuel line running from this line for the possibility of using the storage tank to refuel other spacecraft besides the STV. The last two lines which need connecting are the boiloff lines. The oxygen boiloff line will run into the Space Stations oxygen tanks to be used as breathing air by the astronauts. The hydrogen boiloff line will run to all four attitude control thrusters on the Space Station to be used to adjust the Station's orbit.

When the storage tank needs refueling, a transport fuel tank will be launched from Earth aboard the HLLV with the same launch limitations as for the storage tank. The OMV will rendezvous with it and dock it to the top of the storage tank. Figure 8.14 shows the positions of the storage tank and the transport tank on the Space Station.

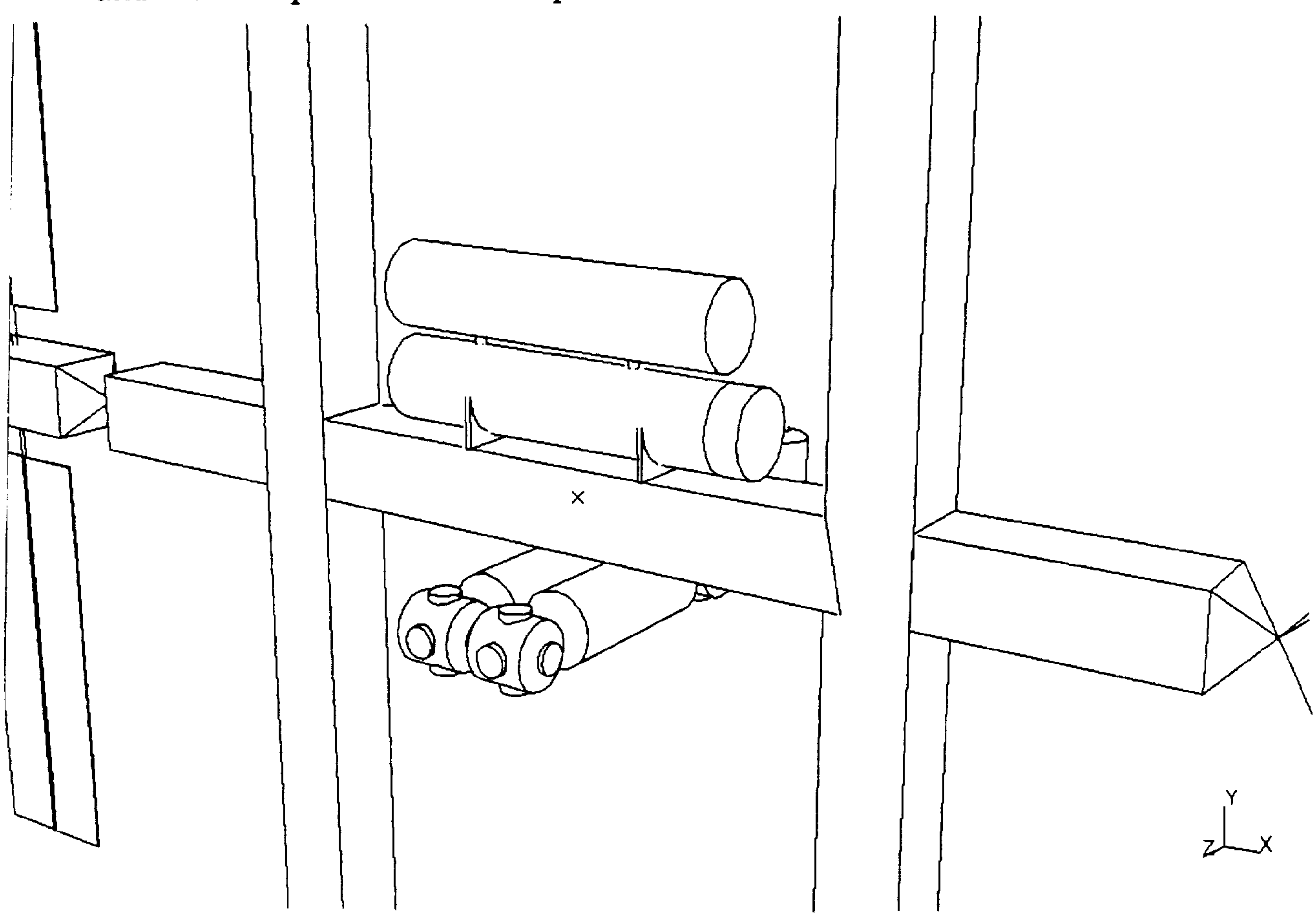

Figure 8.14 Fuel Tanks on Space Station

The process of docking the transport tank onto the storage tank will cause a connection to be made between the two tanks which will allow the fuel to be pumped from the transport tank to the storage tank. When the transport tank is empty, the OMV will disconnect it and insert it into a trajectory that will cause it to burn up in the Earth's atmosphere.

8.4.3 Refueling Process

As stated in Section 8.4.1, the primary concern of any cryogenic fluid storage system which must operate in a reduced gravity environment is that of fluid/vapor separation. To solve this problem, the concept of capillary acquisition is used. Within the fuel tanks are several liquid acquisition devices which are made up of fine mesh screens which absorb the liquid fuel, much like a paper towel, and contain this fluid. The fluid is then pumped out of the tank and into its destination tank (in this case the STV). Stored vapor boil-off is used to create a 15-20 kPa. pressure differential in the storage tank to allow pumping of the fluid in the reduced gravity environment.

The oxygen and the hydrogen will both be pumped through the propellant transfer connection panel to the hangar and into the STV. As stated earlier, there will also be an alternative route whereby the storage tank can be used to refuel other vehicles besides the STV.

The limiting factor in fluid transfer rates is not the pump size but rather the liquid acquisition devices themselves. If the fluid is pumped out of the liquid acquisition devices too rapidly, vapor might be sucked into the screens resulting in a liquid-vapor mixture again being present. Upon talking to Earv Sumner at NASA Lewis Research Center, it was found that, for the tank design which we will be using, the maximum fluid transfer rate is approximately 1515 kg/hr. Figure 8.15 presents a graph of refueling times for the STV from the storage tank.

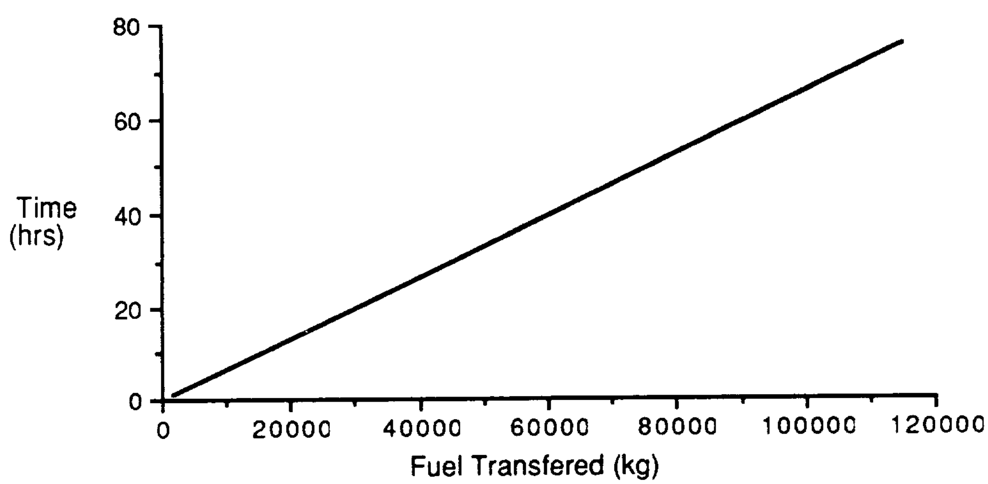

Figure 8.15 Time Required for Fuel Transfer

This same method of fluid transfer will also be used to refill the SSF depot tank from the ground launched transport tank. The fuel transfer rate will be the same for this operation as for the STV refueling process. The following figures show pumping operations for both processes.

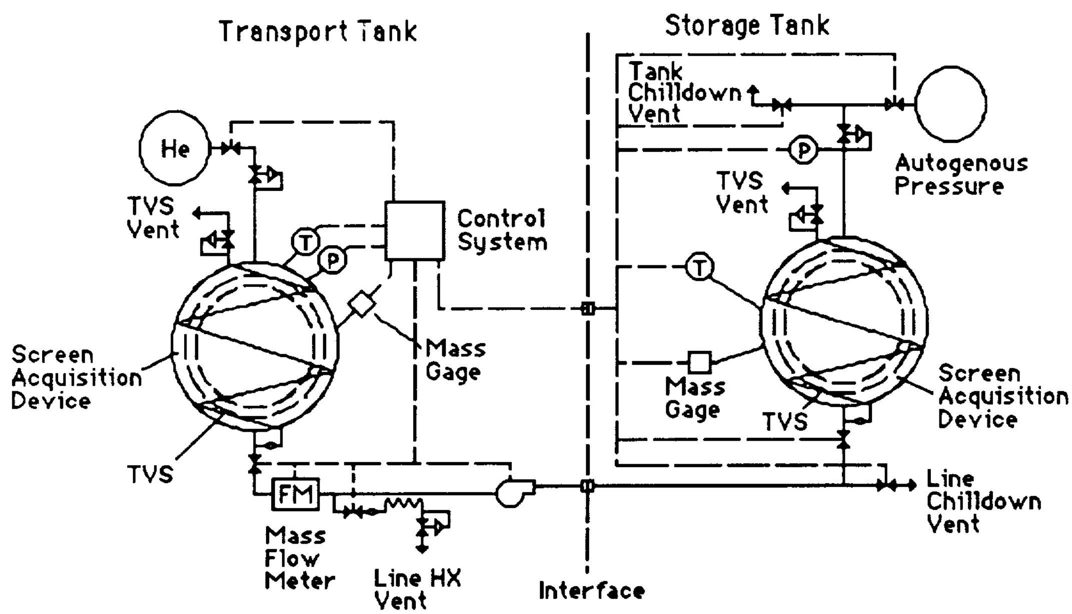

Figure 8.16 Fluid Transfer/Resupply--Transport Tank to Storage Tank [ref 8.12]

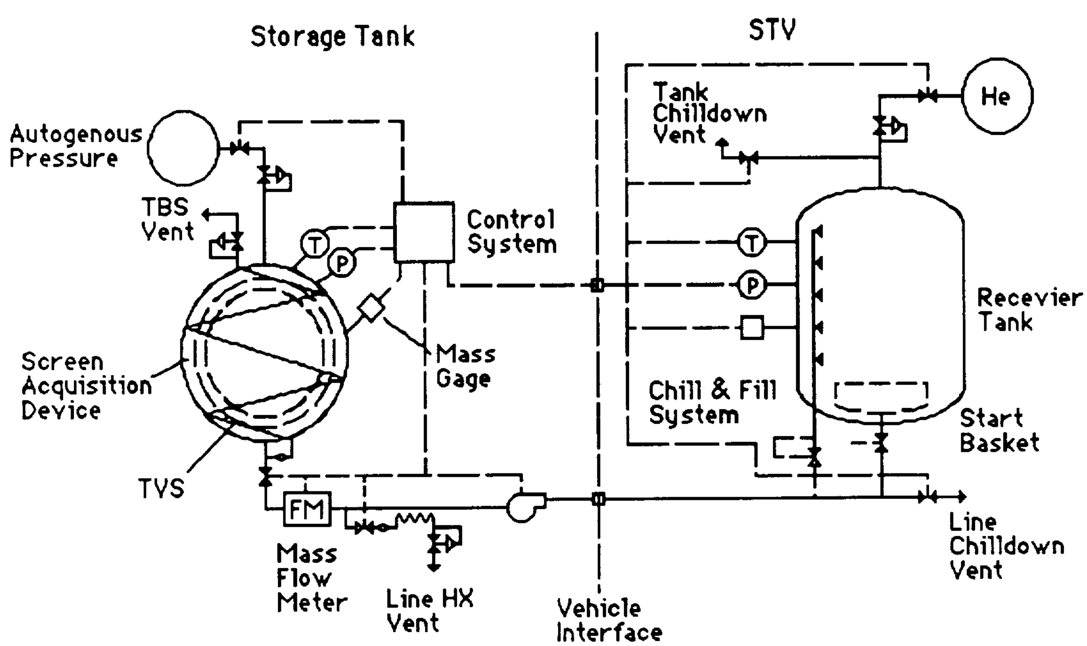

Figure 8.17 Fluid Transfer/Resupply--Storage Tank to STV [ref 8.12]

8.5 Maintenance and Checkout

In order for the STV to operate without having to return to Earth after every mission, on-orbit maintenance in the Space Station must be provided. The maintenance activities were determined to include such operations as handling, assembling, servicing, repair, inspection and checkout. Since the shapes and some of the components of the ASTV and the CSTV are different, the types of repairs required for each STV will differ. However, the necessary maintenance procedures can be generalized under one concept. This section discusses this general maintenance and checkout concept which remains the same for both vehicles.

8.5.1 Scheduled and Unscheduled Maintenance

Some of the maintenance activities can be planned for in advance; others, however, depend on random circumstances and cannot be determined until the end of each mission. Therefore, the maintenance concept is subdivided into two main categories: scheduled and unscheduled.

Scheduled maintenance encompasses the entire systematic maintenance scenario, including servicing and preventative actions required to retain an operational capability. These preventative actions involve inspection, failure detection, and some time-related remove-and-replace tasks, such as engine changeout. Scheduled maintenance generally deals with:

1) items that have wear-out characteristics less than the total STV design life
2) expendable hardware elements
3) those components which require regular servicing.

Some of the scheduled maintenance activities for the STV are listed in Table 8.2.

Activity	Frequency
Replace main engines	Every 31 missions for the ASTV; every 26 missions for the CSTV
Waste removal and disposal	Every mission

Table 8.2 Scheduled Maintenance Activities [ref 8.9]

Unscheduled maintenance refers to the unplanned corrective actions required to restore the STV to an operational level as the result of vehicle failure. In other words, it is the repair of components that fail on a random basis or due to an unscheduled event (e.g. an accident).

The need for unscheduled maintenance can be addressed by performing an STV reliability analysis to determine if any components fail during flight and, if so, what consequences result if the failures are not corrected. This reliability analysis requires knowledge of the failure rates of the different STV components, which can be approximated by using a data base composed of existing documents dealing with failure rates of similar components.

8.5.2 Three-Level Maintenance Structure

Section 8.1 describes the different hangars which will serve as the servicing sites for the ASTV and CSTV and will be equipped with all the necessary maintenance provisions. Since the hangars will not be pressurized, the STV components will have to be grouped into packages capable of being handled by astronauts in pressure suits. We will refer to these packages as space removable units (SRU's). Maintenance will then consist of the removal and replacement of the damaged SRU's, which will then be taken either to the pressurized modules of the Space Station or to Earth for repairs. Thus, the STV maintenance process can best be described using a three-level structure: STV local maintenance; Space Station maintenance of SRU's; and return-to-Earth maintenance. However, the actual operations are further categorized as scheduled and unscheduled activities.

Level I maintenance consists of the scheduled and unscheduled activities that occur on the vehicle while it is in the hangar. Maintenance operations begin upon arrival of the vehicle into the the dock. The dock berthing interfaces are engaged and their integrity verified. The hangar doors are shut to cover the STV. Propellant leak checks are performed on the vehicle and on the propellant transfer system. Visual inspection is performed on the vehicle with a television camera and monitoring system. At the same time the vehicle computer-controlled fault detection system is scrutinized for fault identifications and the results are recorded for maintenance planning. Faults are verified by performing an operational test of the system. The fault is then isolated to the SRU by activating the built-in-test capability. This built-in-test equipment is important because it minimizes the STV-to-Station interface and Station equipment diagnostic requirements. This process is explained better in Section 8.5.3. After all the faults have been identified, unscheduled maintenance tasks are integrated into a complete scheduled and unscheduled maintenance plan. The STV components will be replaced using EVA operations.

The SRU's that fit into the Station maintenance facility airlock, and are determined to be free of contaminants, are repaired, or attempted to be repaired within the Station's shirtsleeve environment. This is Level II maintenance. The units that cannot be repaired in the Station are returned to Earth for Level III maintenance.

Level I maintenance can be accomplished in an average of fifteen hours [ref. 8.9]. The time required to perform Levels II and III maintenance depends on the type of component that needs repair and on the degree of damage.

8.5.3 The Checkout Concept

Checkout is defined to include the ability to assess the condition of the system, the detection of faults, and isolation of faults to the appropriate space removable unit. It is needed mainly after new units have been installed in the STV, before every mission, and in the evaluation of data after a flight has been completed.

The checkout method chosen uses a ground-based control for initiating the checkout and data analysis. In this concept, condition assessment, fault detection, and isolation data are obtained using vehicle-mounted built-in-test equipment to stimulate and simulate the STV components. The data from the vehicle is sent to Earth were it is analyzed resulting in the identification of the faulty SRU's and the indication of the system status. The required maintenance actions are then transmitted back to the maintenance crew located at the Space Station. This concept was chosen over a Space Station-based control because the latter would increase the number of personnel and the hardware and software required for checkout in the Station. We also looked at mounting all the equipment necessary for fault detection and isolation in the vehicle, but this would have increased both the complexity of the avionics in the vehicle and its mass.

The particular areas of concern in the ground-based concept are the bandwidth of the signal and the methods required to ensure the integrity of the data path between the Space Station and Earth, but these concerns are minimal when compared to the effects of placing the equipment on the Space Station or the STV. The estimated time for this checkout process is from two to three hours.

8.6 Satellite Storage and Maintenance at the Space Station

In the event of a satellite being brought back to the Space Station from GEO for repairs by the STV, facilities for its protection and repair must be provided. Hangars similar to the one presented previously for the all-propulsive STV will be provided for these satellites. The hangars will be designed as shown in the following figure:

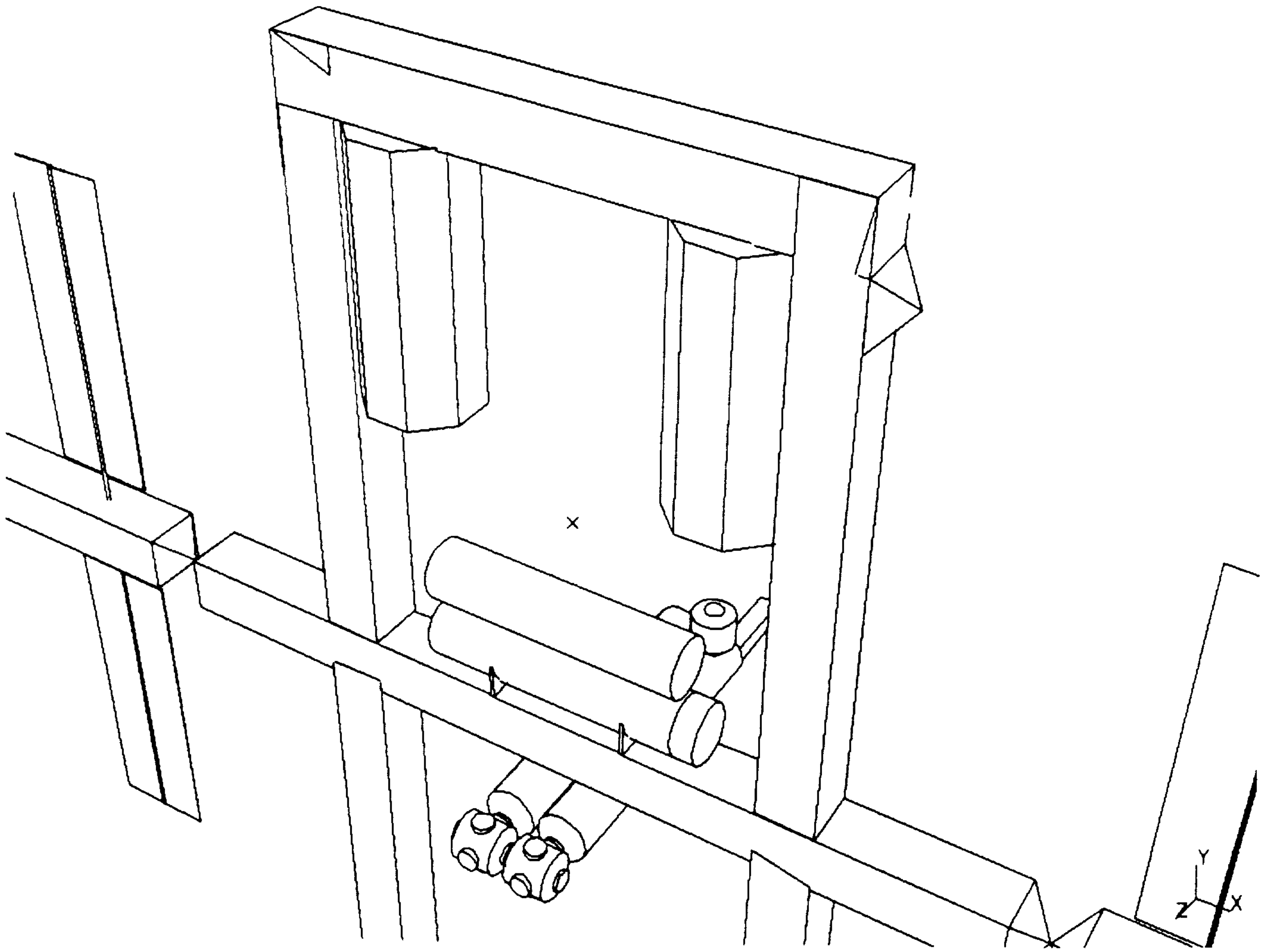

Figure 8.18 Satellite Storage Hangars

The shape of the hangar will be octagonal with a height of 21.3 meters and a diameter of 9.1 meters, with each face having a width of 3.7 meters. The front three faces of the hangar will be extendable solar array blankets, similar to those used in the all-propulsive STV hangar design. The back five faces will be permanent, and therefore loading and unloading of satellites will occur from the front. The hangars will include top and bottom berthing stations for the satellites [ref. 8.10].

As shown in Figure 8.18, The hangars will be positioned at the top of the space station, below the top truss. This placement provides for easy access for the MRMS.

Once the STV is docked, the RMS removes the satellite from the STV's cargo bay. There are then two options for transferring the satellite to a hangar. The first option is to have the OMV lock onto the satellite and bring it near the hangar, docking in the OMV grapple. The Mobile Servicing System (MSS) would then transfer the satellite from the OMV to the hangar. The second option is to have the RMS transfer the satellite directly to the MSS at the point where the main boom and the duel keel meet. The MSS can then travel along the top keel with the satellite to the hangar and place it inside. Once the satellite is in the hangar, the hangar doors can be closed and the satellite is now protected. The satellite can be repaired inside the hangar by astronauts delivered to the hangar by the MSS. The tools needed will be either on the MSS or in the hangar itself.

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Chapter 9

Systems Analysis

9.0 Summary

The role of the Systems Analysis group is twofold: first, to examine the differences between the all-propulsive and aeroassisted STV from a qualitative standpoint; second, to detail a cost analysis that takes into account as many factors as possible. Thus, our group must integrate the philosophical qualitative side with the cost based quantitative side. The following table gives a brief overview of both STV's.

ADVANTAGES FOR ALL-PROPULSIVE AND AEROASSISTED STV'S

All-Propulsive	Aeroassisted
• Expandable on greater scale - Greater modularity - Larger payload capability - Special Mission • Known technology - Less DDT&E - Shorter construction time • Less risk - Lower g-loading - Lower heating values • Less structural fatigue - Longer lifecycle • No aerobrake maintenance • Less dry mass	• No return-to-LEO recirculization burn - Fuel Savings - Less fuel tank volume & mass - Less engine use

Table 9.1

On the cost side, the initial development, production, and launch cost of the CSTV and ASTV is, respectively, $933 and $1,359 million. Note that the CSTV cost is less by $426 million, as much more research must be done to make the ASTV a workable technology. However, the operational costs of the CSTV and ASTV are, respectively, $2,115 and $1,485 million per year. The ASTV saves over 21,000 kilograms propellant per mission. With a launch cost of $1,500 per kilogram [ref. 9.1, 9.2] and assuming 15 missions per year, the aeroassist vehicle saves over $470 million per year. With a vehicle life of approximately eight years, a total of almost $4 billion can be saved using the aeroassisted vehicle. Many of the cost figures generated for this report were interpolated or extrapolated from 1985-86 Boeing and General Dynamics OTV design study reports [ref. 9.2, 9.3].

Vehicle mass is examined to show the major differences between systems. The two most significant mass differences are the aerobrake (1630 kg) on the ASTV and the larger tanks needed (230 kg difference) on the CSTV. Note that Human Factors comprises at least 38% of both vehicle dry masses because they include all crew and crew related systems (i.e. EVA suits, RMS, MMU, airlock, etc.). The mass comparison is followed by a chronological history of the mass growth for the vehicles throughout the project and associated rationale for design changes.

On the qualitative side, maintenance, technology, risk factors, and expansibility are examined. Maintenance includes upkeep and lifecycle, taking into account both necessary hardware (and its transport to Space Station) and man-hours needed for repair. For example, systems such as the thermal protection system covering the aerobrake on the ASTV make maintenance more extensive. The technology base of the all-propulsive STV is more clear cut and sound than that of the virtually unproven aeroassisted technology. As a direct result, the risk to payload, vehicle, and crew are more significant in the ASTV. One such risk exists in the area of stability and control of the vehicle while in the atmosphere. The ability to accurately model atmospheric variations in density is critical to mission success. The CSTV is more expandable than the ASTV, as it is not as restricted by mass, shape, or most importantly, volume. The modular design allows for greater flexibility, whether expansion or modification involves adding more tanks and propellant or moving something beyond the 15 meter radius of the aerobrake. However, this constraint does not make aeroassisted technology inferior to the more conventional chemical configuration; it is just not as well known. Perhaps continued development of aeroassisted technology will allow for greater long term expansion into the solar system and beyond.

The two major conclusions we arrived at are as follows: the ability for virtually unlimited future expansion gives the all-propulsive vehicle a more beneficial long term portrait, while the fuel savings, which directly translates to money savings, is an overriding factor in favor of the aeroassisted vehicle. It is difficult if not impossible to arrive at a definite conclusion as to which design is "better." Perhaps the design, construction, and use of both vehicles is in order.

9.1 Cost of All-Propulsive and Aeroassisted Designs

Although cost is a design factor that frequently makes engineers cringe, it is frequently the deciding factor as to whether or not the design is implemented. Our cost analysis accounts for general costs of each vehicle as well as quantifying as many qualitative arguments as possible. Two types of cost must be considered, initial (one-time) and operational (recurring). These costs have been categorized and are shown in Table 9.2. Much of the information in the following table was interpolated or extrapolated from Boeing and General Dynamics OTV design studies from 1985-86 [ref. 9.3, 9.4].

COST COMPARISON
(in Millions of 1989 Dollars)

INITIAL COSTS

	All-Propulsive	Aeroassisted
DDT&E [ref. 9.4]	600	1,000
Production [ref. 9.4]	320	345
Launch/Checkout	13	14
TOTAL	933	1,359

ANNUAL OPERATIONAL COSTS (based on 15 missions per year)

	All-Propulsive	Aeroassisted
Fuel	1,912	1,275
Maintenance	138	145
Mission Control [ref. 9.3]	65	65
TOTAL (per year)	2,115	1,485

Table 9.2

9.1.1 Initial Cost

Initial cost is comprised of three major divisions: DDT&E (design, development, testing, and engineering), production, and first launch and checkout. At this level, the all-propulsive vehicle has an advantage since the technology already exists and is proven; the DDT&E will cost millions of dollars less to develop. Also, production cost for the all-propulsive vehicle is slightly less because it does not need the elaborate thermal protection system (TPS) tiles that is a basic requirement of the aeroassisted design.

INITIAL COSTS: Production Breakdown
(in Millions of 1989 Dollars)
[ref. 9.4]

CSTV		ASTV	
Propulsion	100	Propulsion	100
Power and Comm.	70	Power and Comm.	70
Structures	45	Structures	50
Thermal Control	25	Thermal Control	25
Assembly	80	TPS [ref.9.5]	10
		Assembly	90
TOTAL CSTV - $ 320 M		TOTAL ASTV - $ 345 M	

Table 9.3

9.1.2 Operational Cost

Operational cost is also comprised of three major divisions: fuel, maintenance, and mission control. The two most significant categories are fuel and maintenance, both of which have been detailed in this chapter. On the fuel side, the aeroassisted vehicle has an overwhelming advantage over the all-propulsive design. Because the aeroassisted vehicle uses aerodynamic drag to slow itself down and insert itself into low Earth orbit instead of the final burn required by the all-propulsive design, a large mass of fuel can be saved. At a launch cost of $1,500 per kilogram of fuel [ref. 9.1, 9.2], this expense adds up quickly. On the maintenance side, the CSTV has a significant advantage over the ASTV, which is subject to more thermal and structural stress than the CSTV. The reason is due to the stresses endured throughout the atmospheric flight portion of the mission, which is encountered only with the aeroassisted vehicle. The aeroassisted vehicle will experience loads of over two g's during each mission, and will experience the large temperature extremes associated with the aerodynamic frictional heating involved with aeroassisted vehicle operation. On the other hand, the CSTV will never experience loads over 0.5 g, and the thermal stresses of space existence will be the same for the CSTV as those experienced by the ASTV when it is in space.

Fuel and Fuel Savings

Fuel is the major cost and mass component of any chemically propelled STV. The amount of fuel necessary to complete a mission is solely dependent on the mass of the vehicle, including its payload for a given ΔV. Both aeroassisted and all-propulsive vehicles use their engines during flight to GEO. The major difference is found in their return to LEO. During this part of a mission, the all-propulsive vehicle will continue to burn its engines, thus using more fuel. The aeroassisted vehicle, however, will use atmospheric drag instead of another burn, thereby requiring less fuel to complete the same mission.

During an aeroassisted return to LEO, the return payload mass affects the fuel savings more than the initial mass. Figure 9.1 illustrates this effect. For a constant initial payload, the fuel savings increases as the payload returned to LEO increases. A maximum fuel savings of thirty-four percent occurs when there is zero initial payload and 5,000 kilograms of return payload mass. The minimum fuel savings occurs when the initial payload is 10,000 kilograms and the return payload is zero. Even under these conditions, the aeroassisted vehicle still cuts fuel consumption by sixteen percent. For a nominal mission transporting 10,000 kilograms to GEO and returning 5,000 kilograms to LEO, the minimum fuel required for an all-propulsive vehicle would be 77,000 kilograms. The same mission conducted with an aeroassisted vehicle would require less than 56,000 kilograms. The result is a fuel savings of more than twenty-six percent.

The greatest operational cost of either STV is its fuel cost. Fuel cost includes the basic manufacturing cost of the liquid hydrogen and liquid oxygen as well as the cost to transport the fuel to the Space Station. Transporting fuel from Earth to the Space Station costs approximately $1500 per kilogram. Therefore, the aeroassisted vehicle saves between 13 and 33 million dollars in fuel transportation cost per mission. For an aeroassisted nominal mission, 84 million dollars is spent in fuel transportation costs versus 117 million dollars spent on the same mission using an all-propulsive vehicle. The corresponding savings is 33 million dollars. Fuel manufacturing cost is relatively small compared to fuel transportation cost. The cost of manufacturing liquid hydrogen is 3.56 dollars per kilogram, while manufacturing liquid oxygen costs 5.5 cents per kilogram [ref. 9.6]. The total fuel mass is made up of 85.7 percent of liquid hydrogen and 14.3 percent of liquid oxygen. Therefore, an aeroassisted vehicle would spend between 90 thousand dollars and 172 thousand dollars on manufacturing costs per mission. The all-propulsive vehicle spends between 117 thousand dollars and 239 thousand dollars per mission. For a nominal mission the aeroassisted vehicle would spend 172 thousand dollars in manufacturing costs, while an all-propulsive vehicle spends 239 thousand dollars. Therefore, an aeroassisted vehicle would save 67 thousand dollars in manufacturing costs and 33 million dollars in transportation costs as compared to an all-propulsive vehicle.

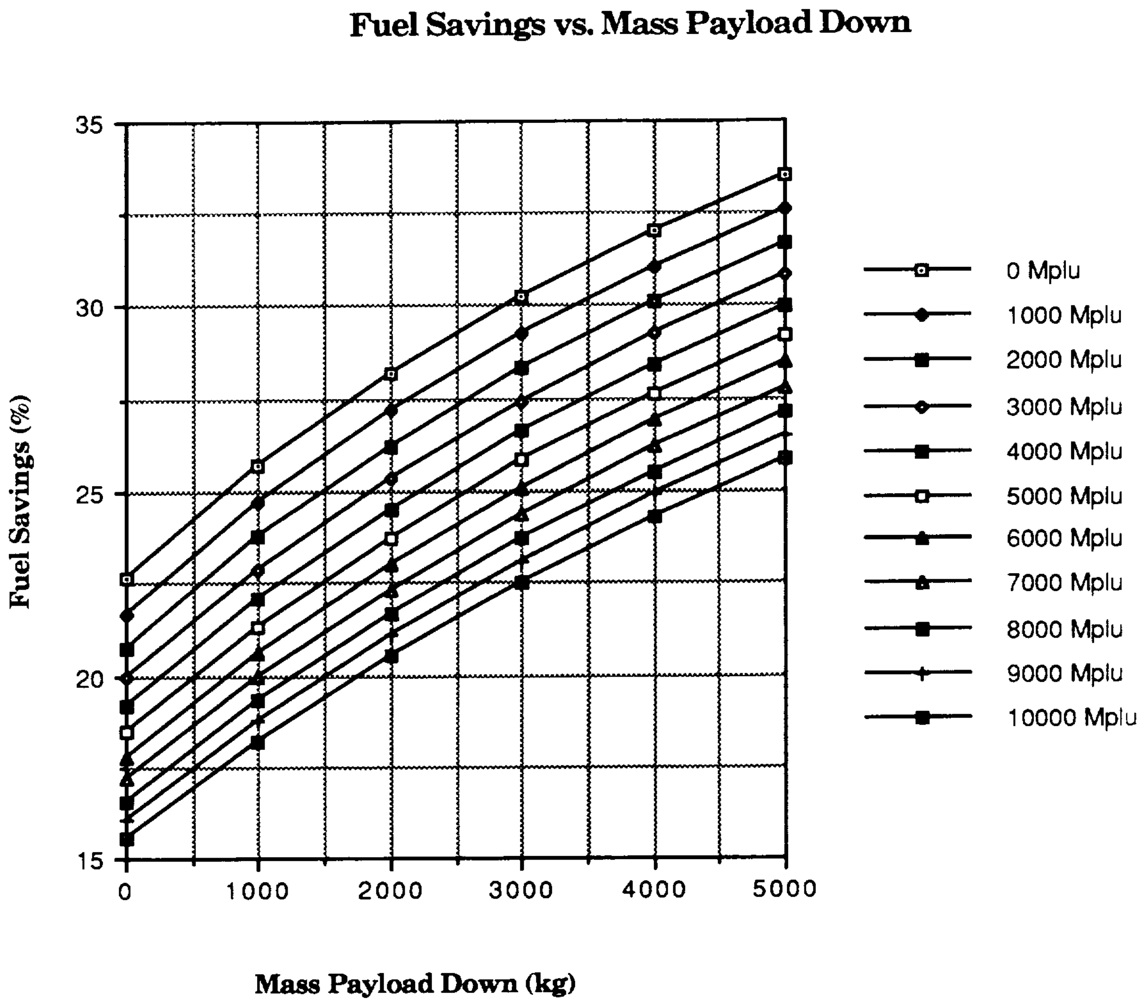

Figure 9.1

Maintenance

There are two types of maintenance for the STV, general and replacement. General maintenance covers necessary maintenance for upkeep while replacement maintenance covers the overhaul of the entire vehicle or specific components of the vehicle after a certain number of missions have been completed.

Because the all-propulsive vehicle is a more modular design than the integrated ASTV, general maintenance is easier for the CSTV. The ASTV will also require more extensive inspection and higher levels of structural integrity due to the demands placed on the vehicle during the atmospheric portion of the aeroassisted mission. Based on EVA costing $100,000 per man hour [ref. 9.7], the added EVA time that results will cause routine annual maintenance costs to be about 15 million dollars more for the ASTV relative to the CSTV (see Table 9.4). The upkeep and periodic replacement of the TPS will cause a further relative maintenance cost for the ASTV of about 13 million dollars per year. However, the ASTV will save a significant amount in engine replacement costs because of the longer burn times required by the CSTV for similar missions. The differencc :s about 21 million dollars per year.

Replacement maintenance includes everything associated with replacing any component of either vehicle. This is perhaps the most expensive of any maintenance since it will usually require a large amount of EVA time. The major area of replacement maintenance for the CSTV is replacement of the main engines and all associated hardware, such as lines, valves, and turbopumps. This process will require a great deal of EVA time that will further add to replacement costs. Smaller areas needing replacement will be the micrometeoroid shield and other minor structures.

The ASTV will not need the main engines replaced as often as the CSTV because the ASTV engine burn time is less for a given mission. Although the more modular design of the CSTV will allow for easier replacement of the engines, the amount of time required for the replacement of the ASTV engines should not be much larger. Consideration of the need for engine replacement ahead of time will minimize the amount of EVA time that this maintenance requires. Over the life of the vehicle, the less frequent replacement of engines in the ASTV will result in a substantial savings. However, the ASTV will need more frequent structural replacement due to the higher stresses of flying through the atmosphere. It is expected that any structural repair performed on the ASTV will require a large amount of EVA time. Another major area of maintenance for the ASTV is replacement of the aerobrake, which includes replacing the tiles and all of the structure that supports the tiles. Subsequently, a large number of EVA hours will be needed to replace the aerobrake.

OPERATIONAL COSTS: Maintenance Breakdown
(in Millions of 1989 Dollars)

	CSTV	ASTV
Routine	45	60
Engine Replacement	81	60
TPS Replacement	-	13
Other Non-Routine Repair	12	12
TOTAL (per year)	138	145

Table 9.4

The stresses placed upon hardware (structure, propulsion, communications, avionics, crew systems, etc.) of the aeroassisted vehicle are greater than the all-propulsive vehicle since atmospheric reentry gives a higher g-loading. These higher stresses lead to a direct increase in replacement or lifecycle cost. Thus, fatigue becomes a factor that must either be eliminated by sturdier designs or more frequent replacements (lifecycle decrease). However, one advantage of the aeroassisted vehicle is that because the engines are not being used in the final burn, engine lifecycle increases at nearly a 3:2 ratio.

9.2 Comparison of All-Propulsive and Aeroassisted Designs

This section discusses the major design differences between the vehicles. First, a mass breakdown is developed showing the major areas of difference between the two vehicles followed by a chronological history of the mass growth for the vehicles throughout the project and associated rationale for design changes.

Second, a more qualitative approach looks at other subjects such as technology, risk, and future growth. Because these factors are difficult if not impossible to quantify, the following discussion expresses some of these problems and concerns that cannot be overlooked.

9.2.1 Mass Comparison and History

Mass is a factor that affects nearly all levels of both design and operation. A mere 100 kilograms saved from any system can translate into hundreds of kilograms of fuel saved per mission. Because of the trickle-down effect of fuel savings, this simple mass savings could save tens of millions of dollars over the lifecycle of the vehicle. Two important things to be looked at are the mass breakdowns of the two vehicles to determine where differences lie, and the mass growth history as a function of time (throughout the semester) to show the design considerations behind the mass differences.

Mass Breakdown

Figure 9.2 shows the general mass breakdown for each system between both vehicles. The specific component masses were either explicitly given (e.g. engine mass) or estimated (e.g. some structural masses). The Human Factors mass total comprises such a large part of the total mass because it includes radiation shielding (general and storm shelter) and every system and mass related to external crew operations (EVA suits, RMS, MMU, airlock, etc.). These two subgroupings comprise 80% of the Human Factors mass total with general living environment, atmosphere, and crew totalling the remaining 20%. The 100 kilogram difference between the two designs lies in the general crew compartment radiation shielding; the ASTV configuration requires more shielding mass because of its larger surface area. Both Structures and Power and Communications have nearly identical mass necessities for both vehicles. The 225 kilogram difference in the Propulsion Hardware mass total stems from a need for larger fuel tanks for the CSTV. The aerobrake mass total includes not just the tiles and necessary backing structure, but also the structural mass (i.e. from stringers) necessary for a 15 meter diameter aerobrake.

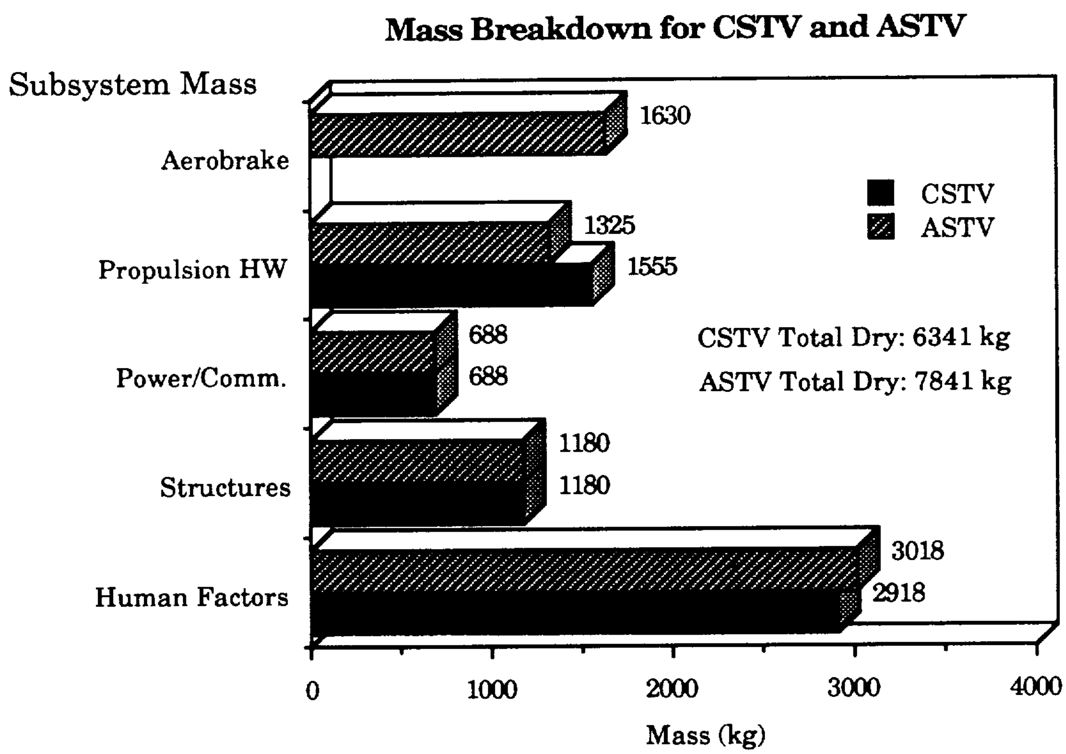

Figure 9.2

Mass Growth

The following graph shows the historical growth of the two vehicles throughout the project. Originally, both had the same mass except for the aerobrake (initially approximated at 1000 kilograms). As the term continued, masses ranged from 6180 to 8850 kilograms for the CSTV and from 6980 to 10,200 kilograms for the ASTV. This variance was a result of two factors: first, systems adding mass for more and more necessary subsystems increased general mass, and second, integration and elimination of overlapping subsystems between systems reduced general mass.

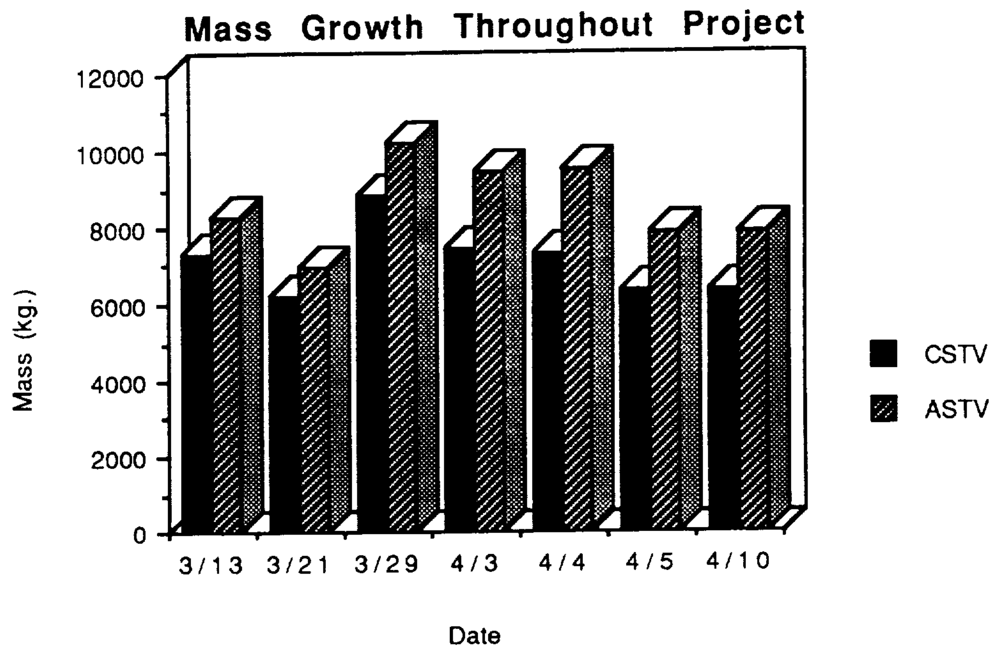

Figure 9.3

The following discussion highlights the areas of significant mass change for each date. The changes common to both vehicles are listed first, followed by the total mass of each vehicle and then the particular mass changes to each vehicle. After each system is identified, a +/- format follows showing whether the change was a mass addition or subtraction, a short description of the change, and the mass difference (*not* subsystem total). Note that the first iteration (3/13) has no +/- changes.

3/13 First true breakdown estimations.

CSTV: 7277

- Human Factors (HF): Radiation storm shelter in rear of crew cabin.
  RMS estimated very low [~50 kg.].
- Power, Comm., Guidance (PCG): Included fuel for fuel cells.
- Propulsion Hardware (PHW): Tank mass underestimated (approximated from previous studies).

ASTV: 8277

- Aerobrake (AB): Rough guess [~1000 kg.].

3/21 Vehicle masses begin to be differentiated beyond aerobrake mass

- HF - Storm shelter eliminated; escape to GEO thought possible [-600 kg.].
- Structure (STR) - Crew module mass being counted twice [-600 kg.].

CSTV: 6177

- PHW + Tank mass increase [+200 kg.].

ASTV: 6977

3/29 First full group review (no real mass restrictions as of yet).

- HF + Crew consumables (food and water) increased for 7 day emergency [+50 kg.].
  + True RMS mass found [+300 kg.].
- STR + Crew module mass for structure reset to original mass [+600 kg.].
- PCG + Radiator mass [+100 kg.].
  + More fuel needed [+100 kg.].

CSTV: 8842

- PHW + Tank mass increase due to overall mass increase [+800 kg.].

ASTV: 10194

- AB + First true mass breakdown [+630 kg.].
- PHW + Tank mass increase due to overall mass increase [+550 kg.].

4/3 First cuts-reason: logistically too expensive to operate high mass.

- HF - Storm shelter integrated with airlock [-100 kg.].
- STR - Crew module mass being counted twice [-600 kg.].
- PCG + Fuel cell mass increase (misunderstanding) [+180 kg.]

CSTV: 7439

- PHW - Tank mass decrease (new type) [-600 kg.].
  + RCS subsystem first true breakdown [+75 kg.].

ASTV: 9469

- AB + Structure/stringer mass for aerobrake previously unaccounted for [+400 kg.].
- PHW - Tank mass decrease (new type) [-200 kg.].
  + same (RCS subsystem).

4/4 Clarifications by entire group.

- HF - Crew perishables reduced (still 7 day emer.) [-60 kg.].
- STR - Total mass decrease (more composites) [-150 kg.].
  + Auxilliary tank added (crew systems, fuel cells, RCS) [+100 kg.].
- PHW (+) Additional cold gas RCS system considered [+500 kg.].

CSTV: 7315
ASTV: 9530

4/5 Second cuts-reason: to get ASTV mass down so that two missions per HLLV fuel launch (i.e. 57,000 kg. fuel/ mission).

- HF - Atmosphere and perishables reduced to *four day* emergency [-30 kg.].
  - General radiation shielding due to config. less [-50 kg.].

- STR + Crew module interior (bulkheads, etc.) added [+100 kg.].
- PCG + Power requirement increase-->larger fuel cells [+25 kg.].
  - Thermal control overestimated [-50 kg.].
- PHW - Thinner tanks, less pressure [-150 kg.]
  - Some plumbing/actuator mass found redundant [-150 kg.].
  - RCS engine mass decrease [-120 kg.].
  - Contingency eliminated (redundant) [-150 kg.].

CSTV: 6237
ASTV: 7828

- HF + General radiation shielding due to config. more [+50 kg.].
- AB - Thinner tiles needed [-300 kg.].
  - Less structure/stringer mass needed [-120 kg.].
  - Less adhesive mass needed [-100 kg.].
  - Less honeycomb structure needed [-250 kg.].
  + Increase in TPS component (skin) [+300 kg.].

3/10 Final Mass Day

- HF - Interior handles/restraints redundant [-10 kg.].
- STR + New calc. for thrust structure [+45 kg.].
- PCG - Reserve and fuel tankage redundant [-20 kg.].

CSTV: 6341 kg.
ASTV: 7841 kg.

9.2.2 Technology

Since the first flight into space, all spacecraft designs have utilized an all-propulsive concept. The particular system proposed for use in our vehicle possesses many similarities to systems currently in use. As a direct result we have a great wealth of DDT&E data from previous flights available to assist us in determining the feasibility of such a system. Aside from the actual engines themselves, all components have been successfully used in one form or another on past space missions. To ensure that the safety and performance of the engines meet or exceed standards, much testing will be necessary. From a system point of view, however, the integration of testing and existing technology could result in production in a relatively short period of time.

The production of an aeroassisted vehicle, on the other hand, would require a great deal more testing. Aerobraking is a relatively new concept in space travel, and no previous experimentation has been conducted to actually put the theory to work. The first aeroassisted flight experiment (AFE) is in its final planning stages. Thus, current technology is not significantly advanced to afford us the luxury of using a proven system. A major concern of aerobraking is the extreme temperatures experienced by the outer surface of the spacecraft. An effective thermal protection system based on the design parameters of the tiles currently employed on the Space Shuttle would need to be developed. Besides thermal protection, our vehicle would also encounter the additional problems of stability and control [ref. 9.8]. Development and testing of these new systems would be made difficult due to unpredictability of the upper atmosphere. The new technology necessary to develop this type of control system would not only increase its cost, but also delay initial production. Our study shows about a forty percent larger development cost for the aeroassisted technology development might be necessary. This larger investment in return should result in a capability to build an acceptably reliable ASTV that can handle the problems of thermal heating and extremely precise stability and control.

9.2.3 Risk associated with each vehicle

Both of the designs, the all-propulsive and the aeroassisted, have a certain amount of risk, as would be found with any space vehicle. Associated with each mission are risks to the payload, vehicle, and crew. The payload is considered the most expendable of the three because it does not have a direct effect on the crew's life in most situations. On the other hand, the vehicle is not expendable and it should be protected in all situations. Obviously the crew is not at all expendable. One objective of a good design is to make sure that the crew is protected in any situation that might occur, and also to be sure that the crew can return safely to the Space Station. Any system that sustains the crew must be triplely redundant: it must have the capacity to survive two system failures and still function normally. This procedure is vital to maintaining the crew's life. The all-propulsive vehicle has only these risks that are present for any space mission. However, the aeroassisted vehicle has three more risks inherent in the mission.

Since the ASTV uses the atmosphere to decelerate, it must account for all of the variations of the atmosphere. Successful operation of the ASTV depends on its ability to predict atmospheric density and cope with possible irregularities. If the ASTV encounters densities less than expected, it has to fall deeper into the atmosphere to gain the drag required to decelerate; and if the ASTV encounters densities higher than expected, it has to take a more shallow flight path to avoid too much drag and the subsequent inability to return to LEO that could result. With the uncertainty found in our present model of the upper atmosphere, these density irregularities pose quite a threat. If the vehicle encounters a patch of atmosphere that is much more dense than expected, it will slow down more than desired, and the ASTV might not have enough power to rise out of the atmosphere with the use of its engines. The orbit of the ASTV would then continue to decay in the atmosphere, with the vehicle finally burning up in the atmosphere or crashing into the Earth. These unexpected atmospheric variations will be difficult to correct for successfully, even with a lifting aerobrake. Another risk associated with the ASTV is stability and control; it will be very difficult to stabilize the vehicle in the atmosphere, and control problems will be present as well. A third risk that the ASTV will encounter is increased cyclic fatigue and failure. The ASTV undergoes thermal and structural stresses on each mission to a greater extent than the all-propulsive vehicle. There is a greater risk to the vehicle, payload, and crew with the ASTV than there is with the all-propulsive vehicle.

9.2.4 Potential Growth

Expandability and Future Growth of the All-Propulsive Vehicle

Another important issue of the designs is expandability and the potential for future growth. The all-propulsive vehicle, since it is more modular than the aeroassisted vehicle, is much easier to expand. A minimum of added structural mass is needed to carry additional payload, which in turn requires fewer man hours for installation. Also, the additional fuel tanks may be installed with a minimum of structural mass penalty. Another benefit of the all-propulsive vehicle is that the payload is not restricted by size, shape, or volume. In contrast, the aeroassisted vehicle must take into account that no structure may be outside the 25 degree flow impingement cone. Therefore the volume of the payload is limited. The only way to avoid this limitation is to increase the size of the aerobrake. Since the aeroassisted vehicle is an integrated design, a major amount of redesign and replacement of supporting structure would be required.

The all-propulsive vehicle also has the ability to expand in areas other than payload. An example might be an extra or larger crew module. If a mission required additional astronauts or scientists, the insertion of another crew module next to the existing one would be relatively easy; and it would not reduce payload capabilities, even with the addition of more fuel tanks. To do the same for the aeroassisted vehicle would be difficult because payload capabilities would be automatically reduced with the addition of any kind of structure.

Other considerations for future growth are special missions. There are a wide variety of possible missions that both designs can perform. A lunar mission is one example, either to orbit the moon or deliver supplies and personnel to a lunar base. In order to make the CSTV capable of lunar missions, some modification of the vehicle would be required. The most important modification necessary for this mission would be the addition of an extra set of fuel tanks. This procedure will be facilitated by the CSTV's modular design, which is designed to accommodate such an expansion. Another interesting mission might be a trip to Mars' moons. Landing on the Martian moons compares favorably from a ΔV standpoint (but would require much more time) with landing on Earth's moon [ref. 9.9]. This mission, of course, would require more extensive modification to the basic

CSTV. In addition to extra fuel tanks, the replacement of the crew module with a much larger module would probably be necessary. The communications and power systems would need upgrading as well. But these expansions are made easier as a result of the vehicle's modularity. Reasons to go to the moons of Mars include the facts that Phobos and Deimos are volatile rich bodies where propellants such as hydrogen can be mined. Or delivery of supplies and personnel to an established Martian base might be possible, perhaps with an unmanned STV. A third mission might be nuclear and toxic waste disposal. A payload of nuclear and toxic waste could be launched from the STV at GEO either out of the solar system or toward the sun and disposed of safely. With the environmental issues facing the world today, this mission scenario could help alleviate some environmental problems. A fourth mission might be to reduce the amount of space debris in GEO from rockets, which is becoming more of a problem with every launch of a new satellite [ref. 9.10]. The STV could pick up large pieces of debris with the RMS and either bring them back to the Space Station, or perhaps send them through the atmosphere to burn up.

In all of the missions considered here, the important thing to remember is that the all-propulsive vehicle, unlike the aeroassisted vehicle, is not limited by size or volume and can carry a wider variety of payloads. The all-propulsive vehicle is also not as limited in mass. Larger fuel capacities are readily attained as larger tanks can easily be added because of the more modular design of the CSTV. It would be harder for the aeroassisted vehicle to add larger tanks because of its integrated design.

Expandability and Future Growth of the Aeroassisted Vehicle

There is no question that aeroassisted technology has basically not been tried and thus remains unproven. As a result, the all-propulsive STV seems to allow for greater expandability. However, let us look beyond the immediate scope of this project.

Although the technology growth necessary for our aeroassisted STV to function at low risk might carry heavy initial cost, it would more than pay for itself in the future. If the technology can be developed for aerobraking in Earth's atmosphere, we will be one step closer to applying it to long-range missions to other moons and planets that also have an atmosphere. Because all-propulsive technology might not be able to effectively accommodate long-range missions to other planets, this could limit or even prohibit exploration of such moons and planets in the near future. In other words, the necessary research for the aeroassisted STV can serve as a giant stepping stone for future space exploration. If aerobraking technology is not followed up sometime soon, valuable time and experience in this technology will be lost.

9.3 References

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