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Submitted to:
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Degobah Satellite Systems

Degobah Satellite Systems
AERCAM Project

Autonomous Extravehicular Robotic Camera

University of Texas At Austin

Degobah Satellite Systems

AERCAM Design Project

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EXECUTIVE SUMMARY

Introduction

Degobah Satellite Systems (DSS), in cooperation with the University Space Research Association (USRA), NASA - Johnson Space Center (JSC), and the University of Texas, has completed the preliminary design of a satellite system to provide inexpensive on-demand video images of all or any portion of Space Station Freedom (SSF). DSS has narrowed the scope of the project to complement the work done by Mr. Dennis Wells at Johnson Space Center. This three month project has resulted in completion of the preliminary design of AERCAM, the Autonomous Extravehicular Robotic Camera, detailed in this design report.

This report begins by providing information on the project background, describing the mission objectives, constraints, and assumptions. Preliminary designs for the primary concept and satellite subsystems are then discussed in detail. Included in the technical portion of the report are detailed descriptions of an advanced imaging system and docking and safing systems that ensure compatibility with the SSF. The report concludes by describing management procedures and project costs.

Design Objectives

Video access to any structural component and ORU (Orbital Replacement Unit) of the Station is an essential aid for crew operation and maintenance of Space Station Freedom. Versatile viewing capabilities also offer unique opportunities for enhancing the public image of the SSF program and its vital role in space exploration.

The video system currently baselined for use on SSF is composed of up to 14 cameras positioned in fixed locations throughout the Station truss. The cameras offer limited viewing angles and require extensive data and support cables. Though some of the cameras can be moved, this operation requires support from the Space Station Remote Manipulator System (SSRMS) and extravehicular activity (EVA) crew members. The system is therefore cumbersome, limited in its mobility, and complicated to service or modify.

The DSS design team has, consequently, undertaken the design of a system more appropriate to space applications and based on advances in a number of technical areas. A few major goals and constraints were established at the beginning of the project to guide the design. The primary task of the AERCAM system, as determined by DSS, is to provide on-demand detailed video access to all or any portion of SSF. In order to accomplish this task, a number of important design objectives were identified. These objectives were ultimately met by the final AERCAM design. The following is a brief list of those objectives:

Satellite Operations:

- Provide observation of the entire Station from orbit
- Provide observation of small components in great detail
- Provide the ability to translate and hold position relative to SSF at various distances
- Provide the ability to revolve around SSF in an apparent sub-orbit for wide angle viewing
- Allow for both autonomous and user-controlled maneuvering and operation

Safety:

- Integrate a highly automated safing and proximity detection system
- Meet SSF safety and outgasing requirements for proximity operations
- Maintain continuous Station avoidance capability
- Ensure negligible damage to the SSF, through some method of energy attenuation, in case of collision

Satellite Bus:

- Maintain an inexpensive, long life satellite system
- Design the satellite to be retrievable, refuelable, and serviceable
- Design simple, modular components for servicing
- Remain compatible with SSF communication and computer protocols

Imaging System:

- Provide capability for both high resolution and a wide field-of-view
- Provide zoom, tilt, and pan capabilities for the camera
- Allow for viewing in all lighting environments

The DSS design team made three major assumptions, each of which implies that AERCAM will be a dedicated element of the SSF baseline. The first assumption is that there will be dedicated human support for control of the satellite. The second assumption is that limited scarring of the Station will be allowed and some SSF resources (propellant and power) will be available to AERCAM. Finally, the AERCAM will require some communication and data handling support from SSF.

Concept Selection

The DSS design team studied four design concepts before selecting a final design. The concepts were evaluated heavily on their safety, reliability, simplicity, and the viewing detail they could provide.

A self-reboosting satellite was considered a possibility for a long-life satellite that would not require modifying the Station truss for a satellite support structure. Although the system would require very little maintenance and retrieval, the mass would have to be greater than 200 lb., and the satellite would require a stand-off orbit greater than a kilometer, reducing the resolution and versatility of the imaging system.

A disposable satellite was also considered. For this concept, an inexpensive satellite would orbit the Station in a manner similar to the self-reboosting satellite, and then re-enter the atmosphere by means of a drag balloon after one or two SSF reboosts (~200 days). Hardware that is inexpensive enough not to prohibit disposal would not be of a high enough quality to provide adequate imaging or maneuvering systems, however; and storage and deployment of replacement units was also a problem.

To avoid scarring of the Station truss, while allowing for a re-usable satellite, DSS studied using an independent docking platform designed to orbit behind the Station on the SSF velocity vector. The added costs of an external docking platform, which would need periodic maintenance, and the added complexity of multiple autonomous vehicles interacting with the docking platform resulted in the rejection of this concept.

DSS ultimately decided to base the preliminary design on a satellite which would nominally dock to the Space Station truss structure, either autonomously or with human assistance, for periodic servicing. This satellite could be smaller due to reduced propellant requirements, and more weight and power could be dedicated to improving imaging and maneuvering systems. This concept was also evaluated to be the most reliable.

The Concept

The AERCAM has been designed as a refuelable, free-flying spacecraft, based from a docking/servicing platform on the SSF truss and housing an advanced imaging system. The spacecraft bus weighs about 130 lbs. and is 0.7 cubic meters in volume. It is powered by NiH_2 batteries, which provide 40 Watts of power for up to 4 hours of operation at peak usage. Portions of the exterior panels of the satellite are removable to provide access to a number of modular components, including the imaging system. The spacecraft features a docking mechanism compatible with the Space Station arm end effectors and a cold-gas propulsion system using compressed waste gas from the Station. The spacecraft is shown in Figure 1, including a front view, a view of the docking mechanism on the right and a view of the interface to the SSF arm end effector on the left.

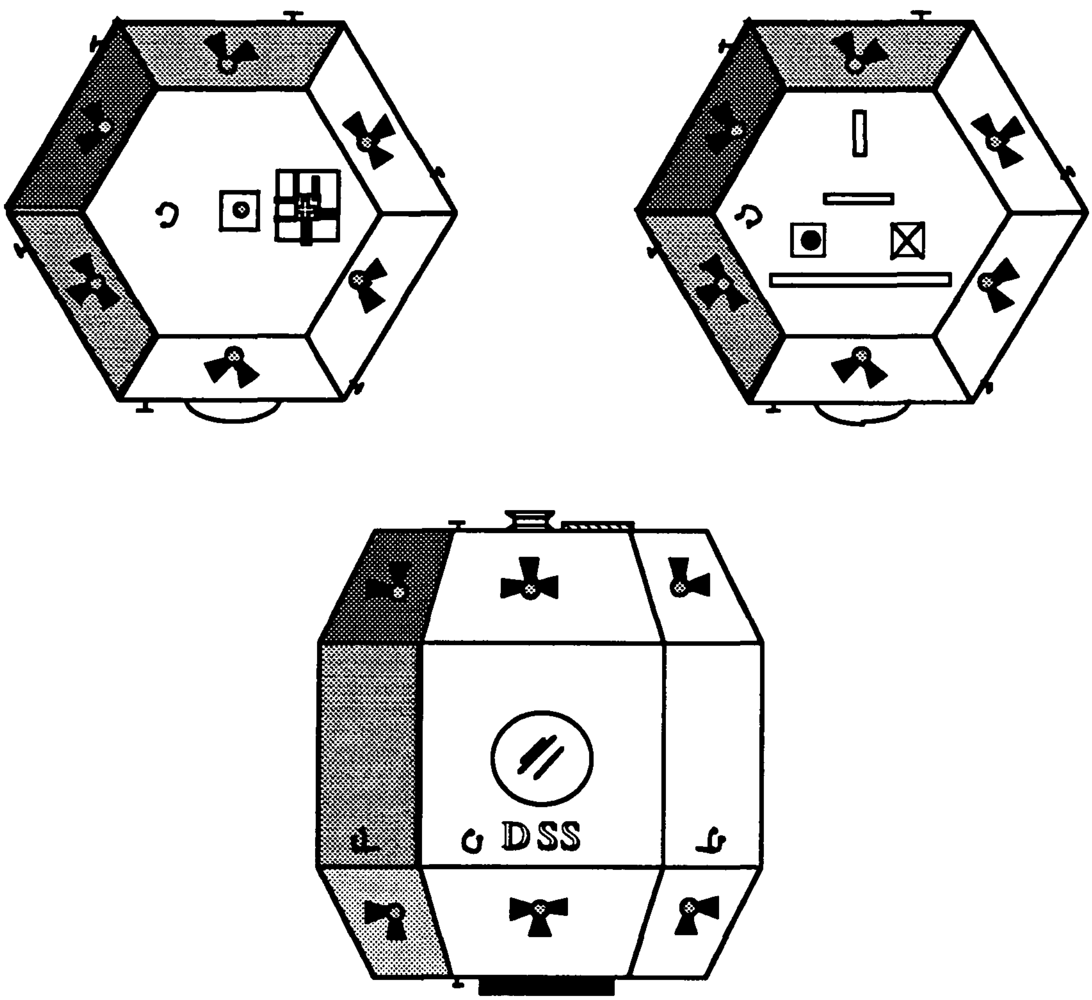

Figure 1: Satellite Concept

Though configured for remote control by a user, AERCAM is fully capable of autonomous orbiting and safing. It features three fault tolerant components and an array of sensors and artificial intelligence for proximity sensing and contingency safing operations. The bus is also equipped with a passive energy attenuation system to prevent damage to the Station in the event of a collision.

The imaging system for AERCAM is a self-contained infrared and visual camera package, housed within a removable, spherical encasement. It provides 40° panning in all directions and zooming. Coupled with a number of flight modes ranging between 10m and 500m from the Station, the imaging system can provide a resolution as detailed as 1mm/pixel and can provide a field of view which includes the entire Station.

Operational Concept

AERCAM is nominally docked to the Space Station truss, on a dedicated docking platform, until it is needed by a crew member. The docking structure is a standard attach platform used for SSF ORU's, modified to include a power and propellant interface to refuel and recharge the satellite while it is docked. This platform is shown in Figure 2.

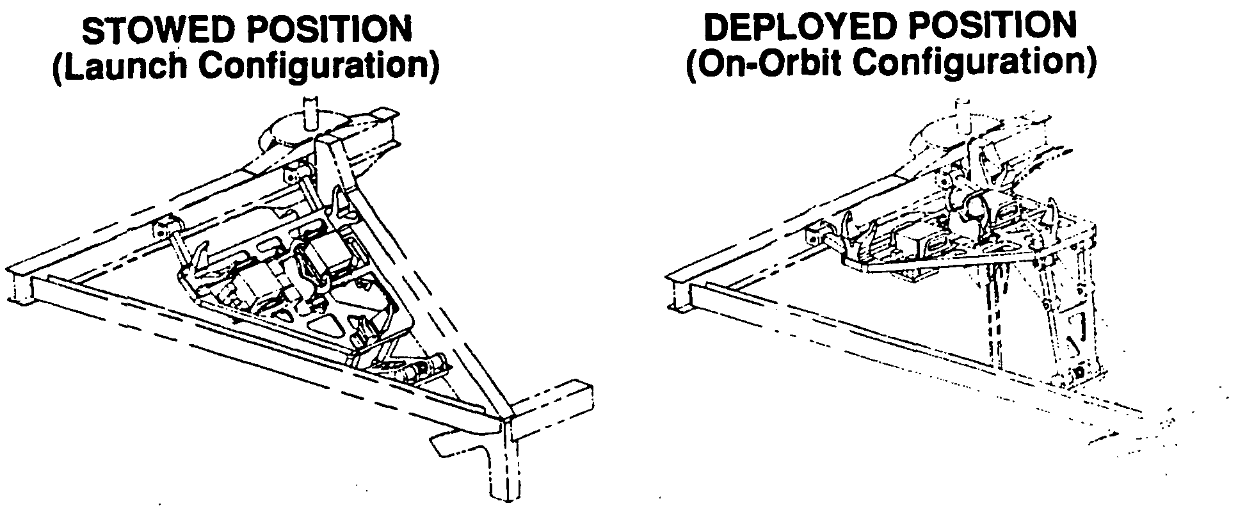

Figure 2: Standard Attach Platform

Once AERCAM is launched from its platform, it is capable of a number of flight modes. The satellite can be controlled by a hand controller and command console onboard the Station. The Station command console can control the operation of the imaging system, maneuvering system, and the internal functions of the satellite. The satellite is also capable autonomous flight modes. It can hold its position at any location with respect to the Station, including holds on the velocity vector and the radial vector. Nominally, the satellite is designed to translate and position hold ten meters from the Station, allowing for an adequate safety zone. In addition, it can maintain a sub-orbit about the SSF at one half a kilometer from the Station. Though capable of user-controlled operation, the satellite calculates its own trajectories and propulsion and attitude requirements, and it is actively aware of a dynamic environment, allowing for operation independent of constant human control.

Imaging System

The imaging system is housed in an independent, spherical encasement, capable of panning up to 40° in all directions. The system is shown in top and side views in Figure 3. The imaging system housing has two locking gimbals, coupled with two small actuators and gimbal stops, to provide the required pointing at a rate of 10° per second. The housing weighs approximately 10 lb., occupies 1.77 $ft.^3$, and the system requires approximately 15 Watts during peak operation. The system is modular and easily accessed through a removable panel and detachable latches at the gimbals.

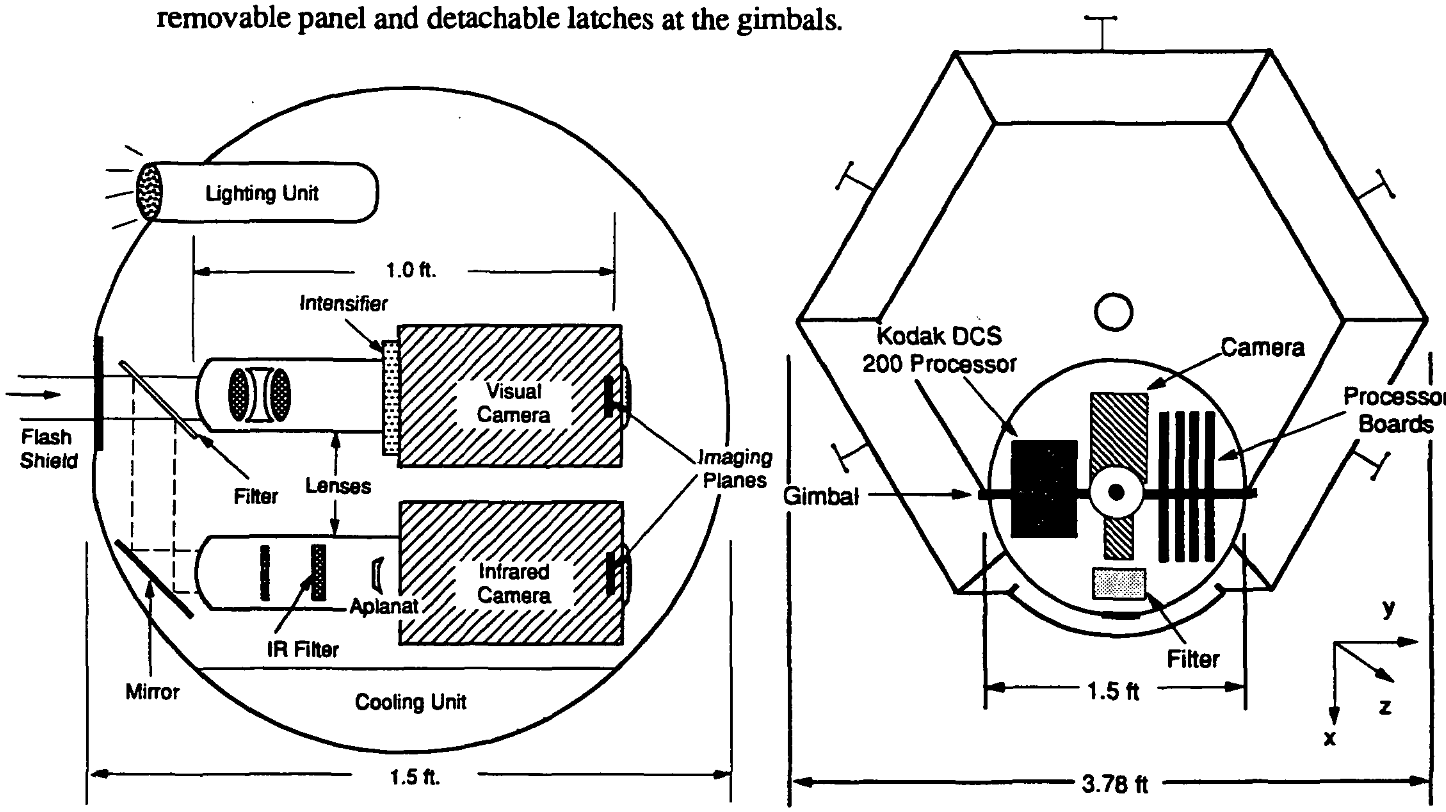

Figure 3: Imaging System

The imaging system features a small light, providing illumination for nearby viewing in dark conditions. It also features a flash shield, reactive in 2 milliseconds, designed to protect the camera components from intense radiation.

The cameras are based on digital charged coupled devices (CCD's) and produce digital images using a matrix of 2048 by 2048 sensors. A series of mirrors and filters separate the incoming electromagnetic radiation into two streams, one directed to a visual camera and the other to an infrared (IR) camera. The visual camera responds over a spectrum from 0.45 μm to 1.1 μm, and the IR camera responds over a spectrum from 1.1 μm to 15 μm.

The imaging system is designed to provide a resolution of 1 mm / pixel at a distance of ten meters from the Station, which corresponds to a field of view 2.048 meters. The imaging system can also provide a field of view as wide a 128 meters at 500 meters from the Station, which corresponds to a resolution of 62.5 mm / pixel.

Images are produced at a rate of 3 frames per second, generating at least 4.5 Megabits (Mbps) of data each second. This data is processed and compressed using a computing system derived from the Kodak DCS 200 and processors used for the Brilliant Pebbles system.

The performance of the AERCAM imaging system is summarized in Table 1, below.

Table 1: AERCAM Performance

Spectral Response	0.45 μm to 15 μm
Maximum Resolution	1 mm / pixel
Maximum Field of View	128 meters
Frame Size	2048 pixels X 2048 pixels
Frames Per Second	3 (nominal)
Maximum Data Compression	12 to 1

Communication System

In order to provide adequate communications support, the satellite is capable of transmitting 100 Mbps of telemetered data and of transmitting and receiving 10 Kbps of commanding, health, and status data. In order to reduce the amount of data telemetered from the imaging system, AERCAM offers a number of data reduction and compression techniques, including Vector Quantization - a patented compression technique that can reduce data in a lossless manner by up to a factor of 12.

The AERCAM communication system is based on a space-to-space communication system originally designed for Space Station Freedom. It employs twelve antennas on the satellite and ten antenna arrays on the Space Station. Each antenna is a circular loop antenna, one wavelength in diameter. The satellite antennas draw 0.5 Watts of power for transmission, and the Station antennas draw less than 2 Watts. The carrier frequency ranges between 14.0 GHz and 14.9 GHz. Based on a transmission path length of 1 kilometer, the AERCAM link budget predicts a signal-to-noise ratio of 25 and a transmission margin of 18.5 dBWatts, both excellent characteristics for a communication system.

Satellite Bus

The satellite bus is a 20 sided hexagonal structure, shown in Figure 1, composed of an internal support structure and 20 external panels, 8 of which are removable. The structure is designed for modularity and ease of access to the internal components. The propulsion system includes a 0.5 cubic foot tank, containing compressed carbon dioxide. It provides up to 150 ft/sec of ΔV. Twenty-four NiH_2 cells supply 40 Watts of power, and can operate up to 4 hours at peak usage. Twenty Watts are allocated to the imaging system for nominal operation, and the computer system and propulsion/attitude control systems are always guaranteed 20 Watts, respectively, for emergency operations.

One end of the satellite is equipped with a docking mechanism compatible with the standard attach platform, and the opposite end of the satellite is equipped with a standard SSF grapple fixture, both shown in Figure 2, to allow for grappling and maneuvering using the Special Purpose Dexterous Manipulator. The docking mechanism provides interfaces with a power fixture and propellant valve to allow recharging and refueling of the satellite while it is docked.

Safety is one of the most important elements in the AERCAM mission. In order to maintain autonomous proximity detection and collision avoidance, the spacecraft is equipped with an intelligent logic system, capable of adapting to a dynamic world model. To support this system and to generate the dynamic world model, the satellite is equipped with sensor clusters on each face of the satellite. These clusters provide both ranging and relative velocity information for the satellite.

The following critical components of the satellite have been designed for three fault tolerance: computer system, thruster/propulsion system, and power system. Key elements in the communication system have also been designed for fault tolerance.

Should the satellite collide with the Station, the bus has been equipped with an energy attenuation device. The method of energy attenuation which DSS chose to implement requires that the vertices of the satellite be fitted with compartmentalized "bumpers," filled with an inert gas. These bumpers represent a passive system, and therefore will always be operational, despite other failures within the spacecraft.

The insurance of safety is perhaps the most significant hurdle to overcome for acceptance of a satellite of this nature. Therefore DSS has been committed to identifying the most reliable methods of operation, control, and safing. We believe that the satellite's design will offer reliability and safety measures which will be acceptable to the SSF program.

Recommendations

Due to the limited duration of the project and the scope of work which the DSS design team undertook, some areas of design have been left for future work. Some crucial areas of future design work are listed below:

- Design of proximity sensor arrays
- Design of a remote user control station
- Selection of an energy attenuating material for the spacecraft skin
- Analysis for low impact scarring of the SSF truss to support servicing and communications
- Testing and simulation for contingency operations
- Miniaturization of spacecraft components

Conclusion - The AERCAM Advantages

Advances in automation, robotics, and microtechnology, as well as the modern proficiency for designing inexpensive, reliable systems with multiple fault tolerance offer a unique opportunity to take a large step forward in space imaging and servicing technology. The current baseline video system for SSF uses the same design philosophy used for video surveillance on Earth - multiple, fixed cameras linked through a large system. While this design philosophy is a proven one, Degobah Satellite Systems believes that a new technology, more appropriate to space application, is not only feasible, it is also more versatile and powerful and represents a stepping stone to automated robotic maintenance in space.

The AERCAM is a more mobile system than the fixed video system. It can be moved more rapidly and does not require crew or SSRMS support to change locations. Its viewing is not limited by physical obstacles, and access to all or any portion of the Station is available on very short notice. It offers much greater versatility in its viewing options, and can even be attached to the SSRMS as additional mode of viewing.

AERCAM requires only a single platform for docking and servicing, and two (as opposed to 14) units must ever be modified or serviced. It also offers superior imaging through higher resolution, wider spectral response, and a larger field of view.

AERCAM's application can be extended far beyond the SSF project. The camera, even based from the Space Station, can be used to provide imaging for nearby spacecraft and to assist in any docking or proximity operations of multiple spacecraft. Further, an easily modified or replaced docking panel and the advanced proximity detection and maneuvering capabilities of the satellite make it adaptable to use on future orbiting facilities.

Finally, if AERCAM were implemented, it would represent the first major step toward using fully automated robotics for space maintenance. The imaging, proximity detection and maneuvering, and intelligent logic systems are the major design hurdles in developing advanced space robotic devices. AERCAM's "hands-off" viewing operations offer an excellent opportunity to safely test these three crucial systems before progressing to actively interacting with Station hardware. It is then a small step to equip one or more of AERCAM's unused external panels with robotic arms and end effectors for use as an autonomous maintenance system. Such robotic systems will be invaluable in terms of reducing crew work time and increasing crew safety.

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Acronyms

A/D	Analog/Digital
AERCAM	Autonomous Extravehicular Robotic Camera
ADSS	AERCAM Docking and Servicing System
CCD	Charged Coupled Device
CCZ	Command and Control Zone
CDR	Conceptual Design Review
CDZ	Collision Detection Zone
CO_2	Carbon Dioxide
CWPROP	Clohessy-Wiltshire Propagation
DOD	Depth of Discharge
DRAM	Dynamic Random Access Memory
DSS	Degobah Satellite Systems
EVA	Extra Vehicular Activity
FOV	Field of View
GN&C	Guidance Navigation and Control
GPC	General Purpose Computer
IR	Infrared
JPEG	Industry Standard Data Compression Technique
JSC	Johnson Space Center
LVLH	Local Vertical Local Horizontal
MAMA	Maneuvering Autonomous Maternal Assistor
MB	Megabyte
MSS	Mobile Service System
N_2	Nitrogen
NASA	National Aeronautic and Space Administration
NiCd	Nickel Cadmium
NiH_2	Nickel Hydrogen
ORU	Orbital Replacement Unit
PDR	Preliminary Design Review
RF	Radio Frequency
RFP	Request For Proposal
RMS	Remote Manipulator System
SAFER	Simplified Aid for EVA Rescue

SCSI	Standard Connecting Serial Interface
SPDM	Special Purpose Dexterous Manipulator
SRS	Supplemental Reboost System
SSF	Space Station Freedom
SSRMS	Space Station Remote Manipulator System
TDRSS	Tracking and Data Relay Satellite System
USRA	University Space Research Association
UV	Ultraviolet
VQ	Vector Quantization
WGA	Waste Gas Assembly

1.0 Project Overview

This document has been prepared by Degobah Satellite Systems (DSS) in response to the Request For Proposal (RFP) #ASE274L for design of a Space Station Freedom (SSF) observation system. It represents the completion of the preliminary design phase for the Autonomous Extravehicular Robotic Camera (AERCAM), developed by DSS in conjunction with the University Space Research Association (USRA), NASA-Johnson Space Center, and the University of Texas at Austin.

The DSS design team's work with the AERCAM project is a partial continuation of the work done by Mr. Dennis Wells at the Johnson Space Center prior to 1992. DSS has completed the initial design effort for the AERCAM project - an effort which included determining the configuration and orbits of the satellites, designing the satellite bus, designing and selecting a camera system, and specifying docking and maintenance scenarios.

This report includes an overview of the project work, the actual design of the AERCAM satellite, and a discussion of the management activities required to complete the project. A recommendation for future work is also included to aid subsequent projects associated with AERCAM.

1.1 Introduction: The Advantages of AERCAM

Video monitoring of the Space Station Freedom (SSF) offers a number of opportunities to assist the crew and ground in operating and maintaining the Station and to enhance the public image of the manned space program. Most significantly, high precision spectral and infrared video monitoring will assist the crew and ground in identifying problems on external elements of the Station and in performing nominal extravehicular operations. It will aid in both the precision and expediency of their work and perhaps significantly reduce the crew's extravehicular activity (EVA) work load. In addition, bringing images of an operating space station to households throughout the U.S. and establishing programs through which schools or other organizations may selectively monitor portions of the station can inform the public, in a captivating manner, of the program which their tax dollars support. This video imaging also allows NASA an opportunity to further demonstrate the importance of the Space Station and its role in the manned exploration and utilization of space.

The role of AERCAM is to provide a video camera platform for Station observation. Currently, the baselined video system for the Space Station Freedom is composed of 14 fixed cameras. Although some of the cameras can be moved, the available viewing locations are limited, and they require EVA and Remote Manipulator System (RMS) support for mobility. The AERCAM concept offers a number of advantages which recommend it over this video system.

To begin, the AERCAM satellite is far more mobile. It can be deployed from its docking platform to any location much more rapidly than any of the fixed cameras can be moved. Unlike the fixed camera system, AERCAM is not limited in its viewing access. It is even capable of viewing the entire Space Station and Station elements outside of the solar arrays. AERCAM's mobility also means that it will not be hindered by bad viewing angles or physical obstacles. Although the AERCAM can easily be attached to the Station RMS as an additional viewing mode, it can move independent of RMS and EVA crew support - a significant improvement over the baselined system.

Maintenance for the AERCAM satellite is also simpler than that required for the baselined video system. Both AERCAM satellites can be docked on a single platform using existing Station hardware. Any repair or modification planned for the cameras would only require work on two units as opposed to fourteen, and the modular design of the satellite makes accessing individual components an easy task. In addition, the satellite is small enough to fit in a Station module's airlock, for servicing without hindrance by cumbersome space suits.

AERCAM's application can also be extended beyond use directly on the Station. The satellites' variable flight modes offer the ability to view operations and spacecraft nearby the SSF.

Finally, the AERCAM camera and computer systems allow more versatile viewing options. They offer superior imaging, through higher resolution, a wider spectral response, and larger field of view.

1.2 Scope & Limitations

The DSS design team's work on AERCAM encompassed only the preliminary design phase of the project. The project focused primarily on the design of the AERCAM satellite and support structure required on the SSF truss. The DSS design team worked heavily with NASA to insure that the satellite design would be compatible with the Space Station, and investigated using existing Station hardware wherever possible.

Due to the limitations of a semester-long design effort, the DSS team limited the scope of its work in a number of the technical areas of the project. Thermal control and materials for satellite structure were not addressed. Energy attenuation, satellite servicing, and some aspects of the camera design were only addressed at the conceptual level.

The project sponsor had already completed significant work in other areas, as well. The DSS team felt that the time required to come up to speed in these areas was prohibitive and that focusing on other work would be more effective and beneficial to the customer. The following areas of design were considered beyond the scope of this project and were either input from Mr. Dennis Wells at NASA or left for future work:

1. Design of the user control station
2. Automated logic and artificial intelligence
3. Command authority
4. Proximity sensor arrays
5. Propulsion system concept

1.3 Design Objectives

AERCAM's primary mission objective is to provide on-demand detailed video access to all or any portion of SSF. In order for this concept to be viable, an inexpensive long life satellite system had to be developed. This goal was achieved by using off the shelf hardware and simple, modular designs.

AERCAM had to be a lightweight, retrievable, and refuelable satellite. This fact, coupled with modular design for easy servicing, would increase the life-span dramatically.

Always being available to crew members for rapid deployment was necessary in order to make the AERCAM more versatile to SSF personal. AERCAM, therefore, required a docking and servicing platform where refueling and recharging can take place so that a fully operational unit is available when needed.

The viewing capabilities of AERCAM were required to include full Station observation from a sub-orbit around SSF and high resolution images from close range. The satellite also had to be able to resolve a one millimeter crack from no less than a five meter distance from SSF. To acquire this close range video, the satellite had to be able to translate, rotate, and hold position relative to SSF.

The imaging system also needed to be capable of viewing the Station in all lighting conditions. The camera was, finally, required to provide tilt, pan, and zoom capabilities to increase the versatility of AERCAM.

Autonomous operation of AERCAM was desired to reduce crew workload. Ideally, the satellite computer system would be able to plot the necessary trajectories and propellant usage to allow it to translate to a specified location for video monitoring, and remain there until needed again. The satellite, therefore, had to provide a number of autonomous flight modes, including position holding and proximity operations at 0.5 km from the SSF.

The final and perhaps most important objective was to design a highly automated safety and proximity sensing system. AERCAM is required to know where it is, what is around it, and how everything is moving relative to it.

1.4 Constraints & Assumptions

In addition to the primary and secondary objectives, the design of AERCAM was limited and directed by a number of constraints and assumptions. The constraints and assumptions which are listed in this report have been developed as a result of the preliminary design effort and information provided by the customer. They are broad in scope, and do not represent an exhaustive or detailed list.

Constraints are design requirements that are driven by factors beyond the control of the DSS design team. If constraints are compromised, then the viability of the project and the ability to satisfy the requirements of the customer may be seriously jeopardized.

Assumptions are requirements on elements which are beyond the scope of the project design work or beyond the design authority of DSS. Assumptions are integral parts of the AERCAM design, and the AERCAM will rely on the support that can only be provided if they are met. In the event that assumptions are not met, significant elements of the AERCAM design would be invalid or unfeasible.

1.4.1 Design Constraints

Since the AERCAM will be operating relatively near to SSF, it was required to meet a number of design constraints to satisfy SSF safety requirements for proximity operations. Among these constraints are plume impingement limits, maximum relative velocities, and provisions for thruster interlocking. In addition, the satellite had to be capable of continuous autonomous Station avoidance through the use of sensors, internal logic, and multiple fault tolerance of system hardware.

In the event of an unavoidable collision with the Space Station, the AERCAM had to ensure that negligible damage would be done to the Station. This constraint necessitated the development of a collision energy attenuation device capable of dissipating the energy produced by the satellite's collision with the Station.

The AERCAM design team also had to take into account the fact that the Station's structures and operations will introduce physical and electromagnetic obstructions to communications between the satellite and its receiving antenna. This affected the design of the communications system both on the satellite and the Station.

Finally, since the AERCAM will be requiring communication and computer support from SSF, the AERCAM was designed to accommodate the limitations of the SSF systems and to be compatible with the SSF protocols.

1.4.2 Design Assumptions

The AERCAM design relies on three major technical assumptions. The first assumption is that there will be dedicated human support for control of the satellite. It has further been determined that remote control of the satellite is desirable for the users. Therefore, it is assumed that dedicated human control of the satellite from a remote location will be available. Although this assumption is integral to the satellite's safety and operation close to the Station, the satellite is still designed to be capable of limited autonomous positioning and orbit holding.

The second important assumption is that limited scarring of the Station will be allowed and some SSF resources will be available to AERCAM. This is necessary to accommodate the user control station and the docking/servicing platform. Resource allocation may include computer processing support and provisions for supplying power and propellant, either from SSF waste gases or from external tanks dedicated to AERCAM.

The final major assumption is that SSF will provide communication and data handling assistance for the AERCAM. This may be provided using existing Station hardware or by using the Station as a platform for housing additional dedicated AERCAM hardware.

Each of these assumptions implies that AERCAM will be a dedicated element of the SSF baseline. AERCAM cannot, in its current design configuration, operate independently of the Station. Therefore, it would not be appropriate or feasible to develop the AERCAM as an independent payload. This also implies that the AERCAM design will be subject to the same safety and reliability requirements as the rest of the Space Station hardware, software, and operations elements.

2.0 Mission Overview

The overview of the mission describes the four primary design alternatives developed by Degobah Satellite Systems during brainstorming sessions early in the project development. All concepts were considered up to the first preliminary design stage, where one concept was chosen as a primary scenario. The primary design, along with the criteria used to determine this design were developed satisfied during the second preliminary design stage. This section overviews the four designs initially considered, and presents the primary design alternative chosen, as well as justification for this design.

2.1 Concepts Considered

This section describes the four concepts developed in response to the RFP, and details the preliminary design alternatives for the AERCAM project. In addition, the criteria and selection process for eliminating concepts and choosing the preliminary design are detailed. The original concepts are termed the self reboosting satellite, the disposable satellite, an external docking platform, and the station docking satellite. The final design is based on the station docking concept and is discussed along with the selection criteria.

2.1.1 Self Reboosting Satellite

The self reboosting concept allows enough propellant to be stored on board each satellite so that it may reboost either before, during, or after SSF reboosts. While paying the penalty of a large satellite with a mass greater than 200 pounds and a more complex system, this option does not require retrieval or servicing during Station reboost. However, some type of refueling would be necessary in order for the satellite to have an acceptable life span. A single, large lensing system would be needed for the camera in order to accommodate the stand-off orbit of one kilometer. This distance from the station would result in poor viewing. Finally, because of the size, complexity, and cost of a reboostable satellite, the number of satellites deployed would be reduced.

2.1.2 Disposable Satellite

Disposable satellites refer to systems that deploy a drag balloon or by some other method increase their ballistic coefficient significantly, causing them to fall out of orbit around the Earth, re-enter the atmosphere, and burn up. This design would demand a satellite that is simple and inexpensive enough that disposal would not be cost prohibitive.

For this design, the AERCAM would be disposed of after one or two SSF reboosts, would require significantly less fuel, would have a mass less than 100 pounds, and would also eliminate any proximity operations required for retrieval. The satellite would use a stand-off orbit similar to the reboostable concept, and would contain minimal computing and support hardware. This hardware would adversely affect the imaging system in areas such as inaccurate pointing and image processing.

Maneuvers within one kilometer of the Station would be avoided because the satellite would not have advanced safing systems. Propellant tank size would limit the life span of the satellite to one or two reboosts, and concerns such as exposure problems and component life span could be ignored.

Two problems that arise for the disposable satellite are the storage and deployment of replacement systems to be placed in orbit after the disposal of the old satellites. While this design does present some interesting options, the higher cost of numerous satellites eliminated the possibility of its implementation.

2.1.3 External Docking Platform

The external docking platform, or the Maneuvering Autonomous Maternal Assistor (MAMA), concept combines the self reboosting and station docking concepts by introducing a second vehicle. The MAMA vehicle would be a docking and refueling platform located behind SSF on the V-bar that would carry enough fuel for refueling the satellites and for reboosting with all AERCAM units on board while SSF reboosts. This vehicle would operate autonomously, with satellite docking commands being controlled by MAMA. This system would allow for hands off operation by the crew of SSF and safely place refueling operations out of range in case of any mishap.

Problems with this design are the added costs of a second vehicle, which needs to be maintained periodically, and the added complexity of multiple autonomous vehicles interacting. The MAMA would also require periodic refueling. The reliability of the system is sacrificed in order to reduce human support time. The external docking platform would be very expensive and unreliable due to its complexity.

2.1.4 Station Docking Satellite

Station docking implies that the satellite would dock to the Station either autonomously or with human assistance. The docking could take place either on the outer surface of SSF or AERCAM could be maneuvered into a pressurized module to be serviced and refueled. This option allows for a much smaller satellite due to the reduced amount of propellant necessary, and therefore increases the number of satellites to be implemented without significant cost impacts. Station docking would present many favorable options in the areas of mass and refueling.

A few negative issues arise with the Station docking concept. One problem is the clearance from SSF for proximity operations. Any station docking scenario requires maneuvering inside SSF's Command and Control Zone (CCZ). The CCZ, which is defined as 20 miles to the front and back and five miles deep out of the orbit plane, is the space in which SSF controls all spacecraft. Another concern is the amount of crew time required for retrieval of the AERCAM. This concern can be eliminated by nominally parking AERCAM on the truss of the Station. Only when crew members need AERCAM will it be called in to action; therefore, crew time to deploy and retrieve the satellite may be overlooked.

2.2 Preliminary Design Alternative

The preliminary design is centered around the station docking concept. This design concept was chosen from the others through a comparison of important criteria, including the viewing detail, the safety of SSF, the reliability, the variety of flight modes, the simplicity, the cost, the lifetime, and the autonomy of the satellite. Table 2.2-1 shows the primary design decision matrix. The decision matrix clearly details the station docking concept's superiority in meeting the mission objectives for the AERCAM. The station docking satellite excels in viewing detail, lifetime, flight mode variety, and reliability. One major advantage of this satellite is the use of autonomous operation. The preliminary design is based on the station docking concept because of its overall superiority.

Table 2.2-1: Primary Design Decision Matrix

	Mechanical Simplicity	Autonomy	Safety	Reliability	Lifetime	Cost	Variety of Flight Modes	Viewing Detail	TOTAL
Weight	3	3	4	4	2	2	3	5	
Self Reboosting Satellite	9 $\boxed{3}$	12 $\boxed{4}$	12 $\boxed{3}$	16 $\boxed{4}$	8 $\boxed{4}$	8 $\boxed{4}$	3 $\boxed{1}$	5 $\boxed{1}$	73
Disposable Satellite	15 $\boxed{5}$	15 $\boxed{5}$	20 $\boxed{5}$	10 $\boxed{2}$	2 $\boxed{1}$	2 $\boxed{1}$	6 $\boxed{2}$	10 $\boxed{2}$	80
Station Docking Satellite	9 $\boxed{3}$	6 $\boxed{2}$	12 $\boxed{3}$	20 $\boxed{4}$	10 $\boxed{5}$	6 $\boxed{3}$	15 $\boxed{5}$	25 $\boxed{5}$	103
External Docking Platform	3 $\boxed{1}$	12 $\boxed{4}$	16 $\boxed{4}$	5 $\boxed{1}$	6 $\boxed{3}$	4 $\boxed{2}$	15 $\boxed{5}$	20 $\boxed{4}$	81

WORST ← → BEST

1 2 3 4 5

The description of the preliminary design will include discussion on the overall concept, orbiting and maneuvering, safety, and the imaging system. Docked on the Station when not in use, two satellites are used as mobile, free-flying, remotely controlled platforms to observe the entire station or a component of the Station. The AERCAM supplements or replaces the fixed cameras currently designed for SSF. The AERCAM satellites have autonomous proximity detection and maintain autonomous position correction. The satellite is capable of three types of orbits: one half kilometer suborbit around SSF, position holding ten meters away from SSF, and translation with respect to SSF from a distance of ten meters.

The docking mechanism on the satellite is compatible with SSF robotic arm end effectors, and the satellite will remain docked to SSF until needed by the crew or the ground. For safety purposes the satellite has a collision detection zone in which continuous proximity sensing and reaction for collision avoidance is maintained, and utilizes a collision energy attenuation device to diminish damage to SSF in the event of a collision. Critical to the preliminary design, the imaging system uses two cameras: one for visual and one for infrared spectra. The imaging system nominally transmits data to SSF at rate of three frames per second. In the event of the shadowing or the eclipse of the sun, the imaging system provides a self contained lightsource. The satellite has a mass of 130 pounds, and provides 40 Watts of power. The satellite is propelled by a cold gas propulsion system which is a modified version of the SAFER propulsion system operating on CO_2. The satellite docks on a platform attached to the truss of SSF. Docking is performed by an SSF robotic arm .

3.0 Technical Design Areas

The following portion of the report presents each of six technical design areas that the DSS design team undertook for the AERCAM project. Also included is a discussion of the public relations benefits offered by AERCAM.

3.1 Imaging System

The success of the AERCAM mission is dependent upon the performance of its primary cameras. The imaging system includes the primary camera components and a computer processing unit. The six major camera components are the lighting system, lensing system, camera encasements, cooling unit, processing unit, and pointing actuators. The primary objective of the imaging system is to provide the ground and SSF with high resolution, digital images of SSF while minimizing power usage, size, cost, and crew support time. These images will be used to assist in maintenance of the Station, as well as to observe any structural and thermal anomalies that may occur to the exterior of the Station. The imaging system is equipped with a zoom lens to provide enlarged, high resolution images at a variable range of distances from the Station. The system has four major functions: panning, tilting, zooming, and pointing. Sample calculations of camera characteristics and resolution computations are shown in Appendix A. Trade studies of different cameras are shown in Appendix B.

3.1.1 Imaging System Requirements

There were seven design criteria that the imaging system was required to meet. A list of these requirements is shown in Table 3.1-1.

The stringent resolution requirements were set in order to see detailed components of the Station. In order to observe structural anomalies on the exterior of the station, the imaging system has to be able to resolve 1mm/pixel. This resolution is equivalent to providing an identifiable picture of an object 3 - 4 mm in length.

The 128 meter field of view requirement was based not only on the need to view the entire Station but also on the limits of the propulsion system. The AERCAM is designed to operate for short periods of time and carry a small amount of propellant. By limiting the maximum orbital distance to 500 meters, propellant and power usage is conserved. Therefore, the imaging system was designed to view the entire 128 meter long Station from 500 meters away.

Table 3.1-1: Imaging System Requirements

Viewing	1mm/pixel resolution at 10 meters 128 meter field of view at 500 meters
Power	< 15 Watts
Weight	< 10 lbs.
Spectral Response	0.3 μm - 0.75 μm (Visual) 1.0 μm - 15 μm (Infrared)
Frame Rate	3 fps
Panning	± 30° from center about Y & Z axes
Servicing	Removable / Modular System

The nominal power output of the satellite is 40 Watts. The imaging system has been limited to using about 1/3 of this power, i.e. 15 Watts, so that the other instruments on the satellite will have ample power to function.

One of the original objectives of the project was to design a small, low weight satellite. As a result, the imaging system is required to weigh less than 10 lbs. Also, a lighter imaging system allows more for fuel, hence extended the operational range of the satellite.

Since the satellite and the station are moving slowly in relation to each other, and since the available power is limited, the satellite was designed to nominally transmit at a rate of 3 fps. This rate is 10 times less than a continuous video signal, which is transmitted at the rate of 30 fps. , but will still provide sufficient viewing for safety and maintenance of the Station. By reducing the image resolution, the frame rate may be increased to provide near-video rates, however.

To simplify satellite attitude control and to minimize propellant usage, the imaging system was required to provide independent panning across 30° in all directions.

Finally, the system was required to be modular so that removal of the entire system for maintenance or replacement will be easy.

3.1.2 Imaging Architecture

The imaging system was designed to be small, versatile, and modular. The entire imaging system is housed inside of a 1.5 ft diameter sphere, shown in Figure 3.1-1, occupying a volume of 1.77 ft^3. The imaging sphere is both thermally and electromagnetically shielded to protect the camera and computer equipment. The sphere is attached to a dual spinning gimbaled platform which allows the entire imaging system to rotate about two axes: the pitch and the yaw. The sphere is constrained from rotating about the roll axis because it does not provide any additional viewing options, but would add to the complexity of the system. The sphere is attached to the gimbals by two latches, as shown in Figures 3.1-2 and 3.1-3. These latches allow easy removal of the imaging system through retractable pins for maintenance and replacement of camera components. Cables are run through the gimbals and latches to provide power and data transmission for the imaging system.

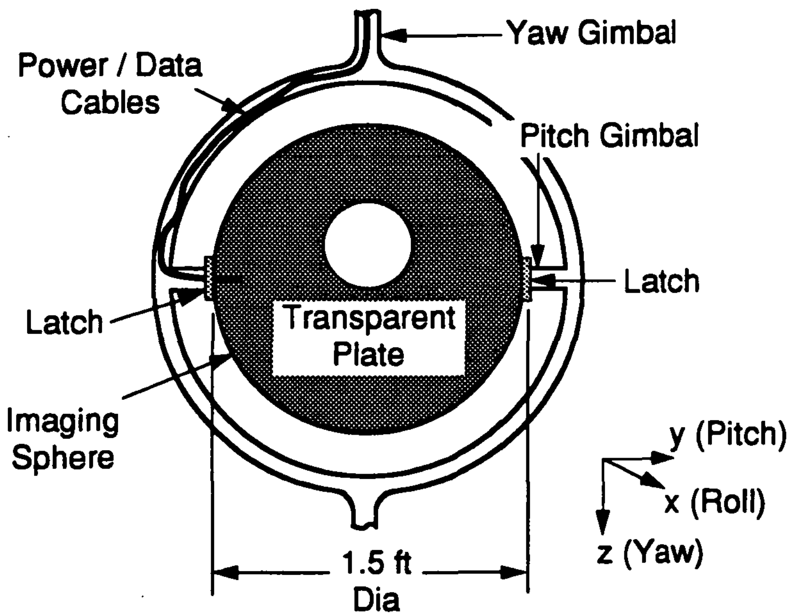


Figure 3.1-1: Imaging System Encasement

Providing maximum viewing of the Station at any distance requires the imaging system sphere to extend 2 inches past the exterior of the satellite. This means that one entire panel of the satellite is devoted to the imaging system, as shown in Figure 3.1-2. On this side of the satellite, the exterior includes a transparent bubble which extends 1 inch in front of the imaging sphere, allowing the sphere to rotate. This entire exterior panel is attached to the satellite by latches and sealed along the edges. These latches allow the panel to detach from the satellite so that the imaging system can be removed. The imaging sphere itself can then be removed using the gimbal latches, shown in detail in Figure 3.1-3, for repair or replacement. To avoid gimbal locking, gimbal stops are placed on the inside of the satellite as shown. These stops also prevent the sphere from rotating past its maximum rotation angle.

A lighter, less complex imaging system architecture, relying on multiple precision mirrors, has also been proposed. Although this system has only been developed to the conceptual stage, DSS feels it could potentially be the best design for the imaging system. This system is covered in more detail in Appendix C.

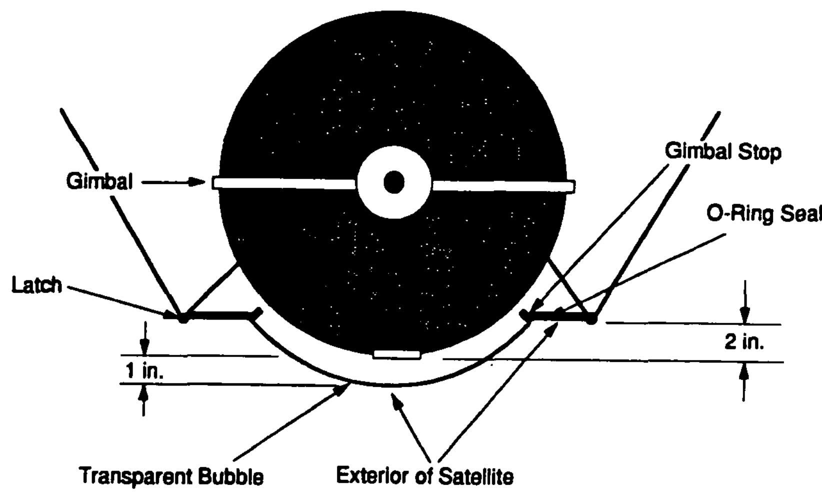


Figure 3.1-2: Imaging Sphere Location

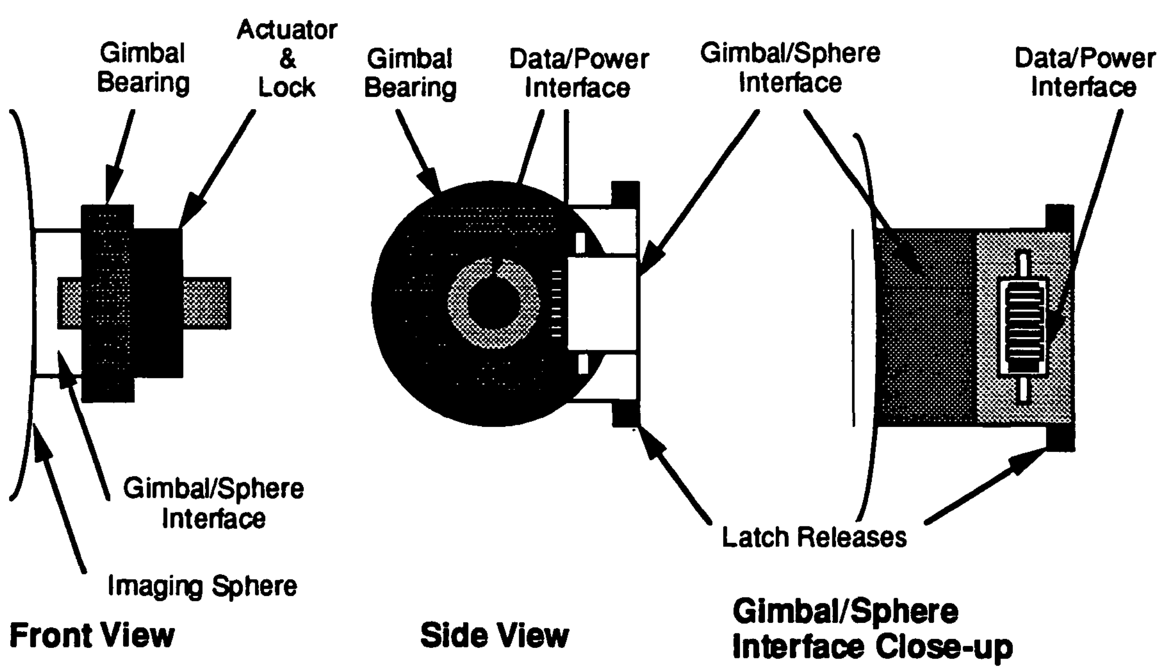


Figure 3.1-3: Imaging Sphere Latch

3.1.3 Major Functions

The imaging system has four major functions: panning, tilting, pointing, and zooming. These functions allow AERCAM to view a large span of the Station with high resolution from a fixed distance.

The imaging system is designed to pan and tilt $\pm 40°$ about both the pitch and yaw axes, as shown in Figure 3.1-4. This provides the satellite with the ability to view half the length of the station (64 meters) from 10 meters away. The panning angle is obtained by positioning the sphere 2 inches past the exterior of the satellite. The gimbal stops are positioned such that the $\pm 40°$ panning is not exceeded in either direction. This panning specification can be increased if more of the sphere is extended past the boundaries of the satellite.

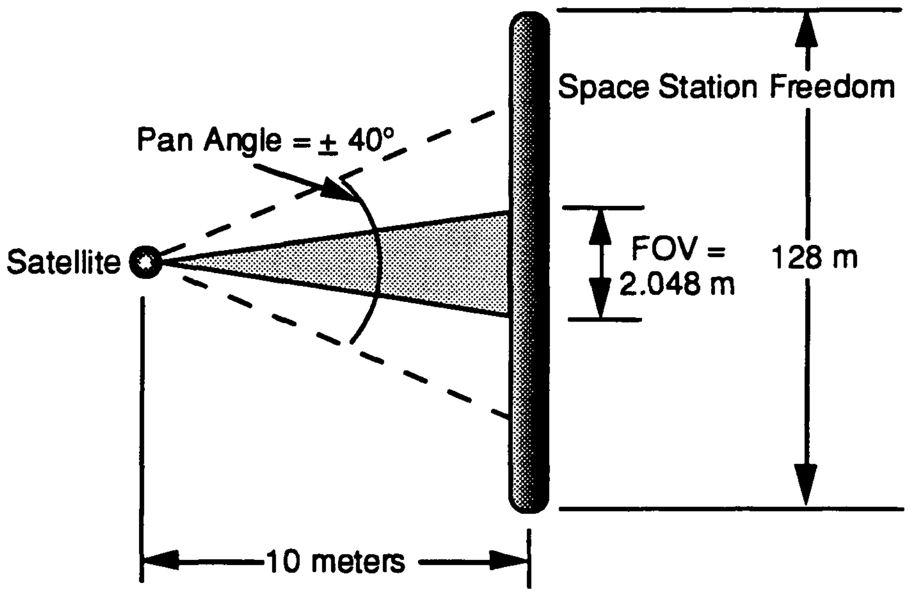


Figure 3.1-4: Panning Range and Field of View
(Not to Scale)

The imaging system is also designed to have a pointing accuracy of no less than 0.1°, which is equivalent to a variance of ± 1 centimeter for the 2.048 meter field of view. This is necessary due to the high resolution of the images that are recorded. The sphere will be pointed through the use of pointing actuators located on the inside of the sphere. These actuators cause the sphere to rotate about both axes while allowing the satellite to remain stationary.

The other major function of the imaging system is its ability to zoom. The lens used in the imaging system has a variable focal length. Nominally, the lens and imaging systems provide a field of view of 2.048 meters at a distance of 10 meters, as shown in Figure 3.1-4, and a field of view of 128 meters (i.e. the entire station) at a distance of 500 meters. These fields of view, coupled with the density of the sensor matrix, allow the camera to have a resolution of 1mm/pixel at 10 meters, and 62.5 mm/pixel at 500 meters. According to Mr. Rick Steambarge at Lockheed Engineering in Houston, a digital image requires 3 - 4 pixels to resolve the smallest object in the image; therefore, at a distance of 10 meters, the imaging system can resolve an object 3 - 4 mm in length. This high resolution provides the necessary accuracy to perhaps view a small crack forming on the exterior of SSF. It is important to note that if a higher resolution is desired with this imaging system, a closer orbit would need to be designed.

3.1.4 Major Components

The imaging system consists of 6 major components. These components are the lighting system, lensing system, camera encasements, cooling unit, processing unit, and pointing actuators; as shown in Figure 3.1-5. The figure was drawn using a 5:1 scale and hence accurately shows how the components fit inside the sphere.

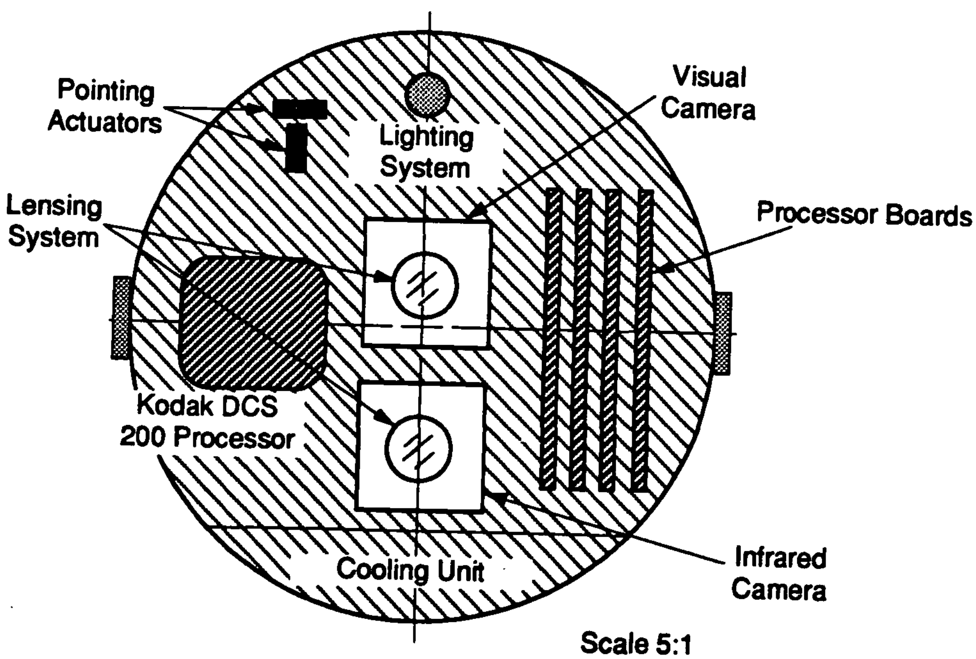


Figure 3.1-5: Imaging System Components

3.1.4.1 Lighting System

The first major component is the lighting system. The lighting system is an optional component for the satellite, and would be used to provide light to the darkened areas on the Station. It would be located in the upper part of the sphere, as shown in Figure 3.1-5, and away from the camera and lenses to avoid any distortions due to heat emissions. An adequate lighting system has yet to be designed; however, after talking with Dr. John Lundberg at the University of Texas, it has been determined that the lighting system would probably require 4 Watts of power, at the most, to provide an adequate light source.

3.1.4.2 Lensing System

The second major component is the lensing system. The lensing system consists of a flash shield, filter, mirror, and lenses for both cameras. Figure 3.1-6 shows a side view of the imaging system, and the placement of the lensing system along with some of the other major components. The entire lensing system weighs less than 2 lbs (Spindler, D-1). The image enters the sphere through a flash shield, which is a 3 inch diameter transparent circle covered with a reactive coating. This coating is similar to the material used in welder's glasses and nuclear blast glasses. The shield is a flat plate that turns opaque in 1/500th of a second when subjected to direct sunlight or to any other excessive lighting intensities. The plate is flat to minimize image distortion, and small to minimize thermal warping of the glass.

After passing through the flash shield, the data is transmitted to a long wavelength, pass interference filter (Elachi, 138). This filter reflects wavelengths greater than 1 μm (infrared spectrum) and transmits wavelengths less than 1 μm (visual spectrum). The filter is oriented at a 45° angle in front of the visual camera to optimize the image reflection and is coated using either a BK7 substrate or fused cilia (Spindler, T-19). These coatings are designed to transmit and reflect the desired wavelengths.

The infrared wavelengths ($\lambda > 1\mu m$) that are reflected from the filter are then transmitted down to a highly polished mirror, as was shown in Figure 3.1-6. The mirror is oriented at a 45° angle, but positioned in front of the infrared camera. The mirror has a Zinc Selluride or Geranium front coating and has a Silver back coating that reflects more than 96% of the infrared light up to a 30 μm wavelength. These coatings cause the mirror to reflect infrared light that is in the thermal infrared range, which will allow the imaging system to record any thermal anomalies. The operating temperature range of the mirror is between -40° C and 100° C (Spindler, T - 12).

After either transmitting through the filter or reflecting off the mirror, the data is then transmitted into the lens of the camera. The lenses are variable 100 mm - 135 mm focal length refractive lenses, which provide the zooming capability discussed earlier. The focal length of a lens defines the distance from the front of the lens to the focal point where the image is formed. For visual images, the 135 mm focal length provides a 2.048 meter field of view at 10 meters with an image magnification of 75. Similarly, the 100 mm focal length provides a 128 meter field of view at 500 meters with an image magnification of 93 (EG&G[1], 121). The amount of space occupied by the lens is shown in Figure 3.1-6. The main difference between the lens on the visual camera and the lens on the infrared camera is the IR filter. The visual lens contains a normal lensing system; whereas, the infrared contains the IR filter and an aplanat. The IR filter is a wide band filter made of a Silicon substrate (Spindler, T-19). The aplanat conducts the final focusing and spectral trimming of the image before it is projected onto the image plane (Elachi, 139).

An optional component for the lensing system is an intensifier, as shown in Figure 3.1-6. The use of this device would depend on its actual size, power consumption, and ability to produce a clear image. The intensifier would be based on the Intensified Multispectral Camera produced by Xybion Electronics and would intensify the brightness of the image being projected onto the image plane (Kennedy, 25). The intensifier would, therefore, reduce the need for an autonomous lighting system. The intensifier would be a variable gain intensifier between 0 and 100% and would produce 15000 electrons for every 1 electron entering the intensifier. The intensifier is composed of a photo cathode, a photo anode, a micro channel plate, and a minifier that adapts the image to the size of the Charge Coupled Device (CCD) (Kennedy, 27). The main problem with this component is that it would require an abundance of power, and could degrade the image due to its high intensification.

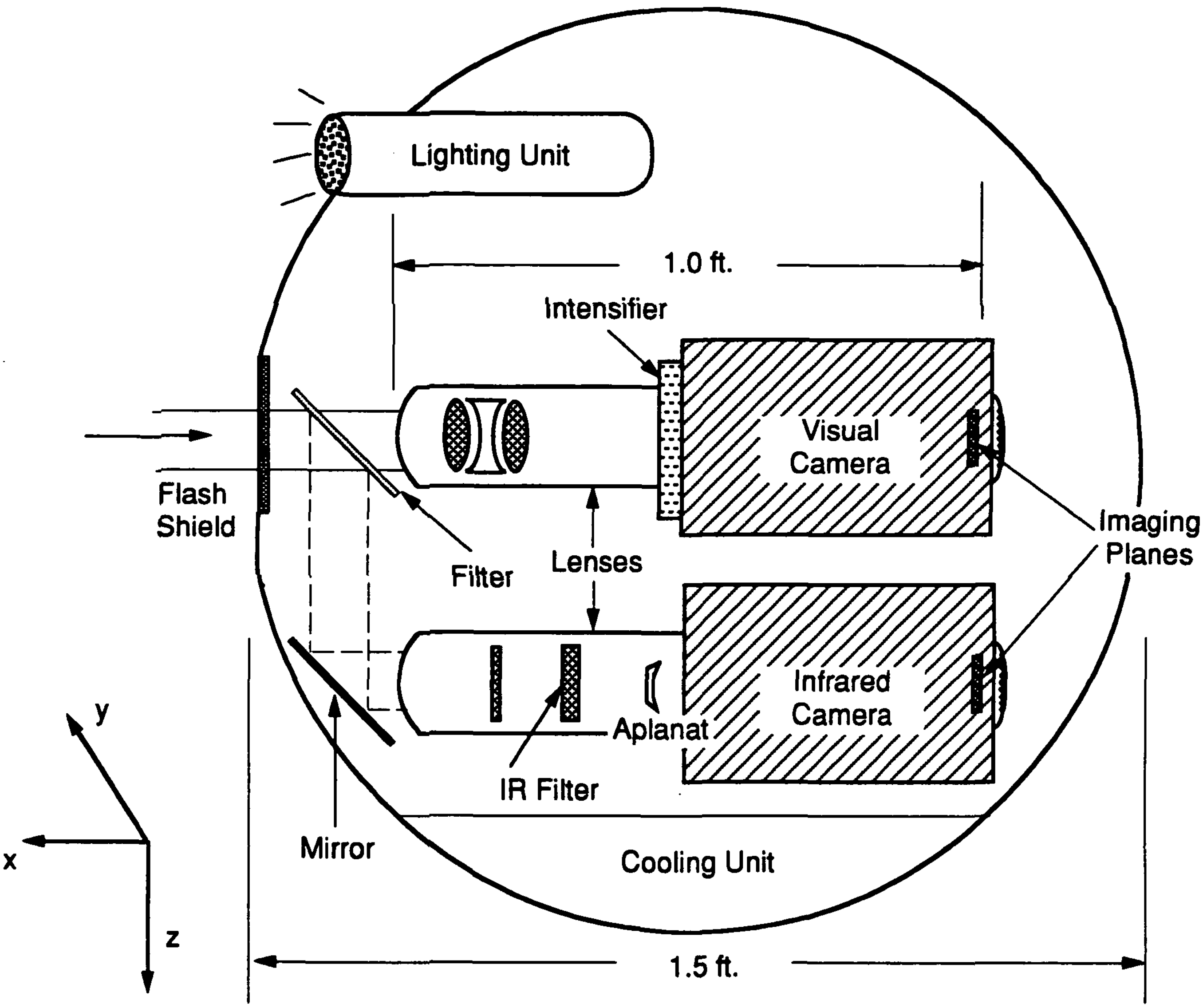


Figure 3.1-6: Side View of Imaging Sphere

3.1.4.3 Camera Encasements

The third major component is the camera encasement. There are two cameras inside the imaging sphere, as described previously: the visual camera and the infrared camera. Both cameras are designed similarly, but they differ in the types of images they record. Also, since visual cameras are designed to record shorter wavelength emissions, they result in higher resolutions (Wertz, 222).

The visual camera is positioned just above the center of the sphere due to space constraints, as shown in Figure 3.1-5. This will cause the camera to translate slightly while rotating, but will not cause any significant image degradation. The visual camera that is used in the sphere contains an imaging plane that is an 8 bit, 2048 X 2048 pixel element CCD array manufactured by EG&G Reticon. The image plane is the area inside the camera into which the image is projected. The pixel array has a center to center pixel spacing of 13.5 μm, which forms a 27.6 mm X 27.6 mm square image plane. (EG&G Reticon[2], 151). The resolution obtained by the camera is defined by the number of pixels in a line of the array. The 2048 pixels provides the AERCAM with the necessary 1mm/pixel resolution while minimizing the size of the lens needed to obtain this resolution. The image plane has a spectral response of 0.45 μm to 1.1 μm, which extends from the visual to the near infrared end of the electromagnetic spectrum. The image plane can be easily modified to range from 0.15 μm to 1.1 μm if needed by adding an optical UV sensor coating (EG&G Reticon[2], 155).

The image plane is housed inside of the EG&G Reticon MC 4020 camera body. The body is made of black anodized extruded aluminum and contains an integral liquid crystal shutter for fast imaging. The body is 3.88 inches wide by 3.88 inches tall by 6.25 inches deep, as shown in Figure 3.1-6. The camera requires 9 Watts of power to operate and only weighs 2.2 lbs. It has been tested to withstand 50 G's of shock and up to 150 Hz of vibration; therefore, it is a very sturdy camera. The camera outputs a 4 MB image and has an operating temperature between -20°C and 50°C (EG&G Reticon[3], 1 - 18). This temperature will be maintained by using a thermal control system for the sphere. The current frame rate of this camera is 0.9 fps. This is slower than the required frame rate of 3 fps; however, according to Paul Lelko at EG&G Reticon, this rate should increase in the next few years .

The infrared camera is located beneath the visual camera to allow the images to be reflected into the camera, as shown in Figure 3.1-6. An off-the-shelf infrared camera has not yet been selected. The main problem with selecting an infrared camera is that the resolution of these cameras is significantly less than visual images. The infrared camera body will be the same dimensions as the visual camera and have the same sized image plane, but the image plane will contain different sensors. The image plane will need to be made of Mercury Cadmium Telluride to maximize the clarity and resolution of the image (Hogan[1], 138) as well as provide the maximum spectral response (Wertz, 225). One option for the infrared camera is to use the technology developed in the Brilliant Pebbles (Space Defense Initiative) design. Brilliant Pebbles uses a smaller resolution image plane (128 X 128 pixels) and is extremely light weight (6 oz.) (Hogan[1], 138).

3.1.4.4 Cooling Unit

The fourth major component in the imaging system is the cooling unit. Due to the sensitivity of the IR camera, interference from nearby heat sources must be eliminated. Therefore, the image path to the IR sensor plane must be cooled. The cooling unit is needed to keep both the image plane of the infrared camera and the optics in the lens cool. It must be small to fit inside the imaging sphere, and will probably use nitrogen or helium as the coolant (Hogan[2], 129). The unit must weigh less than 2 lbs, and operate using less than 2 Watts. The cooling unit is located beneath the infrared camera as shown in Figure 3.1-5.

3.1.4.5 Processing Unit

The fifth major component inside the imaging sphere is the processing unit. Both the visual and IR cameras will have digital data links to the image processing computer. The processing unit includes the processor and the processor boards. In the front view of the imaging system, Figure 3.1-5, the processor is located on the left of the cameras and the processor boards are located on the right side of the cameras. The DSS design team recommends using the Kodak DCS 200 Digital Processor as the primary processor for imaging system. This processor is designed to operate with a digital Kodak camera similar to one used by AERCAM. Therefore, interfacing the camera with the processor should be simple. The DCS 200 is 4.5" X 6.7" X 4". It contains a 200 Megabyte hard drive and is capable of storing up to 50 uncompressed images using DRAM storage. This is equivalent to 15 seconds of data at the imaging system's nominal rate of 3 fps. It is also capable of some data compression using a JPEG compatible technique. Finally, functions on the DCS 200 can be interactively controlled by the user, and it comes configured with a standard SCSI interface port to allow interface directly with a Macintosh, IBM, or higher end computer (Kodak, 1-7). One compression option is to modify the system to use Vector Quantization (VQ) which compresses images by a factor of 12:1.

For additional storage and processing of imaging data, computer systems similar to those of the Brilliant Pebbles satellites can be used. These are small, lightweight systems that can process on the order of 100 fps (Hogan[2], 128). They are less than 6" X 8" X 2", and testing for their flight certification has already begun (Wood, 18-20). The systems are also capable of additional forms of data reduction and data compression, such as Vector Quantization, described later in the report.

Figure 3.1-7 shows how far the processing unit extends into the sphere. The figure was drawn using a 10:1 scale, and accurately shows how much the sphere extends into the satellite.

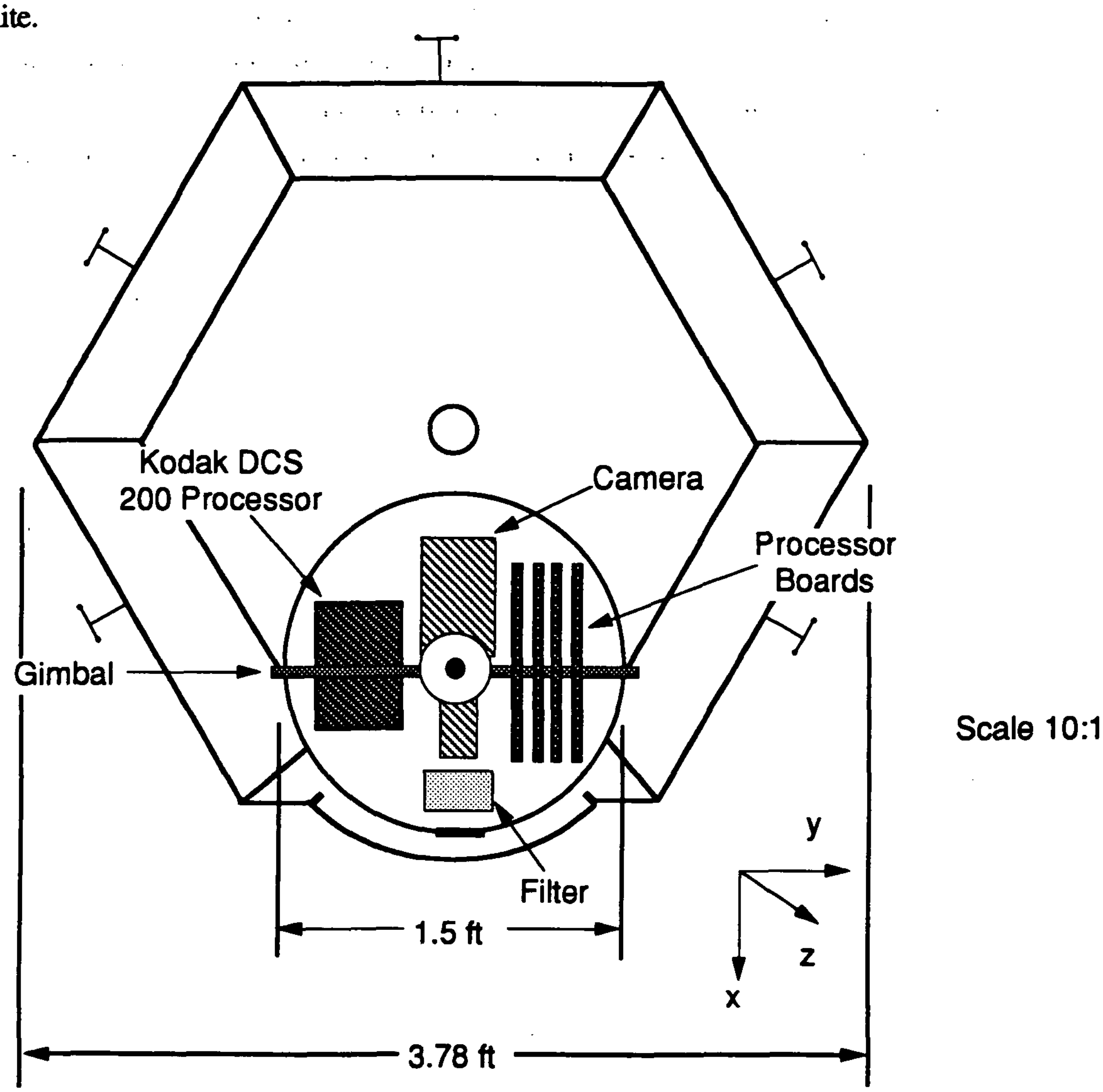


Figure 3.1-7: Imaging Sphere Placement

3.1.4.6 Pointing Actuators

The sixth major component in the imaging sphere is the pointing actuators. As explained previously, the function of these actuators is to point, pan, and tilt the camera about the two axes; therefore, two actuators are needed. Figure 3.2-4 shows that they are located in the left part of the sphere, above the processing unit. When the camera is not rotating, the gimbals lock to prevent the sphere from moving. When the camera begins to point, the gimbals unlock and the pointing actuators begin to rotate. By conservation of momentum, when the actuators rotate one way, the sphere will rotate the opposite way.

Due to the difference in sizes and moments of inertia of the actuators and the sphere, the actuators need to rotate at a very fast rate to cause the sphere to rotate slightly. One actuator has a moment of inertia of 0.00068 kg m^2 and will rotate at the rate of 6000° per second to allow the sphere to rotate the opposite direction at a rate of 10° per second.

3.1.5 Imaging System Overview

The imaging system that was designed meets all of the initially specified requirements. The overall mass of the imaging sphere, not including the gimbals, is 6.4 lbs. This leaves 3.6 lbs. available to design the lighting system and cooling unit. The total power required to operate the imaging system, including the cooling unit and lighting unit is 15 Watts. The lenses and the image plane were designed to provide the 1mm/pixel resolution at 10 meters and 128 meter field of view from 500 meters. The entire spectral range requirement is covered with the exception of 0.3 mm to 0.45 mm in the visual region. Currently, the frame rate of the camera is 0.9 frames/second, but it is expected to increase in the coming years. The panning requirement was exceeded by 10° in either direction by providing ±40° of panning from the center. Finally, by using latches and a detachable panel to keep the imaging system in place, the entire imaging system is removable.

3.2 Communications and Computer Systems

The computer and communications systems on the AERCAM satellite must be capable of processing and transmitting large amounts of data. On the order of 50 Megabits per second of data may be produced by the imaging system. In addition, two-way communication between the satellite and commanding stations on SSF and the ground must be provided for commanding and monitoring of AERCAM. These tasks require communication hardware and software and image processing equipment not only on the satellite but also at the ground and SSF user stations.

This section of the report specifies the processing capabilities required for computer systems on the satellite and at the user stations. It briefly discusses the architecture and division of tasks for the computing system as well. It presents various techniques for reducing or compressing the imaging data for transmission. Finally, it specifies hardware and procedures to meet the communications requirements of AERCAM.

3.2.1 Computing Systems

The computing system on the AERCAM satellite will be a distributed system designed for three-fault tolerance in its critical functions. Its primary functions are control of the spacecraft, contingency situation detection and reaction, and data handling. It will handle the coordination and management of all information moving to and from the satellite, and it will coordinate nominal and contingency operation of all autonomous functions for the satellite, including safing.

The computing system will coordinate the reception and transmission of a number of types of data. Appendix I provides a list of suggested data that must be transmitted to and monitored by the ground control station. This list was taken from Space Mission Analysis and Design, by J. R. Wertz. Below is a list of the major types of data the computers will process and the ways in which they must be processed:

- Commanding data
  - Receive
  - Decoded
  - Distribute
  - Return command responses
- Imaging Data
  - Gather
  - Store
  - Reduce/Compress
  - Format
  - Transmit
- Satellite Health & Status Data
  - Monitor internal systems
  - Transmit

To control the satellite, the computing system will act as a general purpose computer, coordinating action between subsystems within the satellite. It will contain a number of predefined, programmable sequences to control the autonomous functions of the spacecraft interactively with the changing external environment. It will also contain autonomous logic to react to various spacecraft failure and contingency situations. The computing system will most likely employ some form of artificial intelligence to enhance its reaction to a dynamic environment. It is important to note, however, that all autonomous functions of the satellite may be over-ridden at any time by ground or SSF users.

The autonomous and artificially intelligent logics, and hence the majority of the computing system requirements, are primarily being specified by the customer.

3.2.1.1 Computing System Architecture

The control of the satellite and its communications system is the most important and safety critical function of the AERCAM computers. Image processing, though important, is not critical, and it places a larger burden on computer memory. Also, image processing does not require fault tolerance for safety, whereas satellite control does. In light of these factors and to make maintenance easier, DSS recommends using a distributed architecture for the computing system, shown in Figure 3.2.1-1. In the figure, dashed lines represent the flow of data, and solid lines represent the flow of commands. Satellite control and image processing will be separated into physically distinct units, housed in separate locations within the satellite.

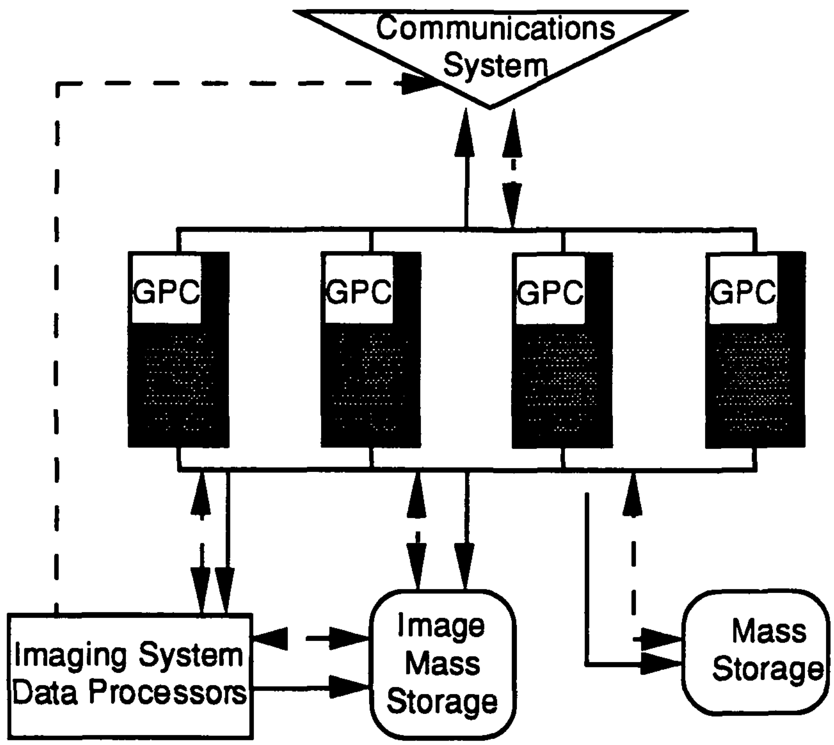


Figure 3.2.1-1: Block Diagram of AERCAM Computing System

3.2.1.2 Satellite Control Computers

Control of the communications system and primary satellite functions will reside in four redundant general purpose computers (GPC) housed within the satellite. A voting scheme, probably "sift-out modular redundancy" (Brat, 11), could be used to reject faulty data generated by any one of the four AERCAM computers. In addition, up to three of the computers can fail, and the satellite will still be operational. In order to prevent overload or failure of the critical computing functions, the primary system will be capable of locking out data input by the imaging system.

Based on research of the Brilliant Pebbles project, the DSS design team believes that four redundant computers could be provided on the AERCAM without significant mass or volume impacts to the satellite. To further reduce the weight, the computer can access a data base stored on a single mass storage unit for processing normal satellite functions. Only the critical safing procedures must be stored in four redundant units.

For general purpose computing Ada is recommended for the software language. It is widely accepted, Department of Defense standard, and capable of easily managing distributed and embedded systems (Wertz, 569). A throughput estimate for the control functions of the satellite is shown in Appendix D.

For autonomous control of the satellite's internal functions, maneuvering, and proximity detection and reaction, an artificially intelligent system is required. The system needs to be a learning system, containing a continuously updated world model and action planning based on a predetermined hierarchy of rules. This system would, of course, be directly integrated with the proximity detection sensors. It would require three fault tolerance, as mentioned above, since loss of autonomous safing is considered a catastrophic failure. Design of this system is being completed by Mr. Dennis Wells at NASA-JSC, and is beyond the scope of the DSS design team's project.

3.2.1.3 Image Processing Computers

The image processing system will be housed within the imaging sphere along with the rest of the instruments package. This system will not have fault tolerance, because its functions are not critical and they require large amounts of computing power. If it should fail, however, it can easily be removed with the sphere for servicing by a crew member.

The image processing computers are designed only to generate, process, and store imaging data. They do not control imaging system functions. The computers used for the image processing system are described in greater detail in section 3.1.4.5.

3.2.2 Data Handling

Due to weight and power limitations for the satellite and SSF communications systems, the DSS design team has emphasized reducing the amount of data that must be telemetered to the user stations. This may be accomplished by using data reduction and compression techniques on board the satellite, requiring increased processing capabilities at the user stations.

For the purpose of this report, data reduction refers to the permanent removal of data from the telemetry queue before transmission from the satellite. Data compression refers to the reformatting of data before transmission to reduce the number of Megabits required to send a given amount of information. Compressed data requires processing at the user station to reconfigure the data to a usable format.

3.2.2.1 Data Compression

There are two primary methods of data compression. The first method is classified as compression only because it is a transmission technique rather than a means of eliminating data. This technique is the digitization of data. The imaging system will not produce continuous data, but instead will produce images at a rate of 3 fps. By producing and transmitting this data in relatively slow, digital format, data transmission is both reduced and simplified. Digital data can be more precisely transmitted because they are less susceptible to distortion and interference than data in analog transmission (Wertz, 453). They can also be easily regenerated so that noise and disturbances do not accumulate in transmission. Finally, digital links have extremely low error rates and high fidelity through error detection and correction.

The second, and most significant form of data compression is Vector Quantization. This type of data compression could reduce the data rate by a factor of 10 or 12 (EER Systems, 1). VQ is accomplished by replacing groups of pixels with "vector representations" for transmission (EER Systems, 8-9). These residual vectors are then decompressed, at the user station, by use of a computerized "codebook". This type of compression requires some increase in image processing at the user station, but the maximum errors introduced by compression are smaller than the errors introduced elsewhere in the system. Therefore, VQ is statistically lossless. The VQ concept is licensed to EER Systems, Inc. and requires the use of a Codebook Processing Chip, also licensed to EER and patented at Utah State University.

Although the DCS 200 image processor is capable of JPEG compression, Vector Quantization can more significantly decrease the load on the communications system. In addition, it has already been integrated into the ATHENA™ satellite, a satellite with imaging capabilities similar to those of AERCAM. ATHENA™ is planned to launch in 1994 (EER Systems, 1).

3.2.2.2 Data Reduction

Data reduction techniques, by definition, result in the loss of some data. Therefore, they would be employed at the discretion of the user, although they may be even more valuable in terms of reducing the data rate than either of the data compression techniques mentioned above. Below is a list of data reduction techniques which would be useful for the AERCAM system. Each reduction technique is defined in greater detail in Appendix E.

- Frame grabbing
- Pixel averaging
- Intensity filtering
- Pixel memorization
- Reduction in number of scanned pixels
- Reduction of scan rate

Each of these data reduction techniques is useful for different applications. They may be employed separately or in various combinations, as the user desires. They may also be selected autonomously by the computer system according to criteria specified by the user. Data compression has a limited useful reduction capability, whereas data reduction has an unlimited capability accompanied by a commensurate loss in image quality. The DSS design team recommends that Vector Quantization be used nominally for data compression.

3.2.3 Communication System

The primary objective of the communication system is to provide a continuous link to the ground and Space Station for telemetering of imaging data, monitoring of satellite health and status, and commanding of the satellite. Therefore, there exist two important steps in the AERCAM communication path - link from AERCAM to the Space Station, and link from the Station to the ground. The hardware specifications which the DSS team has selected were evaluated not only by their reliability, but also by their simplicity, size, and power requirements.

The communications system on the satellite will be capable of the following primary activities:

- Carrier tracking for 2-way coherent communication
- Uplink detection and reception
- Downlink telemetry modulation and transmission for command response, health & status, and imaging data

To support commanding, the communications system will receive and decode commands from the receiving/transmitting antennae on the Station. It will also transmit prepared health and status data and command responses to the receiving antenna on the Station. Finally, the system will handle transmission of the prepared and compressed imaging data to the receiving/transmitting antennae on the Station. It will not, however, perform any data compression or processing other than modulation.

Continuous communication is of vital importance to the safety and successful operation of a remotely controlled satellite operating so close to the Space Station. Therefore, the communication system is being designed for redundancy. It will be capable of autonomously detecting faults and recovering communication links through stored software sequences. By using multiple omni-directional antennas, the system was designed to require no mechanical pointing. Under certain failures, however, the spacecraft attitude control system may need to be relied on for emergency pointing. Should communication be lost and not recoverable, the communication system would detect this situation by the loss or termination of a standard heartbeat from the Station. It would then notify the AERCAM spacecraft in order to begin contingency data storage and to execute pre-determined safing procedures.

The communication architecture has been designed to compensate for the structural and electromagnetic interference present very near the Station. It is also be compatible with SSF and Mission Control Center communication schemes and protocols.

3.2.3.2 Communication System Specifications

The AERCAM communication system is based off a space-to-space communication system baselined in the original SSF design. It relies on eight satellite antennas and ten antenna arrays on the Space Station. Each antenna is a normal, circular loop antenna, one wavelength in diameter.

The communication system receives at the satellite at a frequency band between 14.0 GHz and 14.3 GHz. The satellite transmitting frequency is between 14.5 GHz and 14.9 GHz. The represent a median carrier wavelength of 0.027 meters. The satellite transmitter nominally draws 0.5 Watts of power.

The link budget for AERCAM was calculated using a transmission path length of 1 kilometer, which is twice the maximum distance from the Station that AERCAM is anticipated to nominally operate. Based on the above characteristics, the AERCAM link budget yielded a signal-to-noise ratio of 24.46 and a signal transmission margin of 18.5 dBWatts, both excellent performance characteristics for a communications system.

3.2.3.2 Communication System Architecture

In order to support communication with both the SSF and the ground, there will be receiving/transmitting antennas on the Space Station. The AERCAM satellite will transmit data directly to antennas on the Station, which will in turn transmit data to the ground through the TDRSS network. This top-level architecture was chosen for three major reasons. Direct link to the ground was unfeasible due to the large amount of time that the ground would experience loss of signal. The weak transmitting capability of a small satellite would result in poor signal quality. And finally, transmission directly from the satellite to the TDRSS would require a transmitting antenna and transmission power (>100 Watts) larger than the satellite is capable of supporting.

Although communication between the AERCAM satellite and the Space Station is accomplished across relatively short distances, it is subject to a number of possible interferences. To accommodate for structural and electromagnetic interference encountered at close proximity to the Space Station, both the Space Station and the satellite will be equipped with multiple antennas. All of the antennas (on both the Station and the satellite) will receive a periodic heartbeat. The antenna which has the strongest reception of the heartbeat will be designated by the computer system as the "active" antenna. This "active" antenna will be the only antenna which the computer system recognizes for the reception and transmission of data. This concept is shown schematically in Figure 3.2.3-1

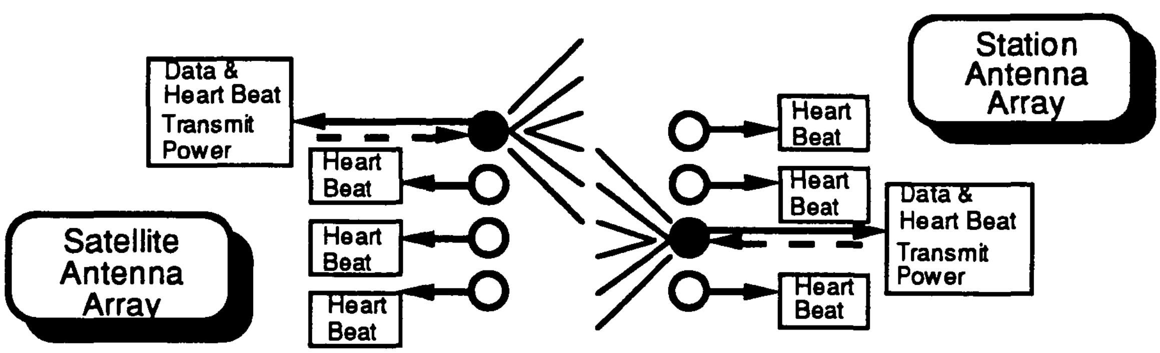


Figure 3.2.3-1: Active Antenna Selection

Communication to the ground may be accomplished in one of two ways. If the system used by SSF for communication with TDRSS is capable of handling the data produced by AERCAM, then standard Station protocols will be adhered to, and the AERCAM data will be telemetered with the SSF data through TDRSS to the ground. Commanding from the ground would also be handled through the Station. If the Station is incapable of handling the AERCAM data, then a transmitting antenna dedicated to

AERCAM will have to be added to the Station truss. This would most likely be a high gain antenna located with the AERCAM docking/servicing platform on the SSF truss structure; and it would require power support from the Station, since a separate power-producing facility dedicated to AERCAM would be cumbersome and inefficient.

Regardless of the architecture for ground communications, however, data must be delivered to and processed by the user control station on SSF. This capability will require a dedicated computer, console, and control deck. If the Station computing and console accommodations are capable of supporting user activities, then they should be used. Otherwise, a separate user station must be developed. Either way, a separate control deck for remote commanding of the satellite must be built.

3.2.3.3 Satellite Communication System

The satellite's primary communication system will be housed next to the computer system, opposite the imaging sphere. Figure 3.2.3-2 shows a schematic of the data handling and communication system.

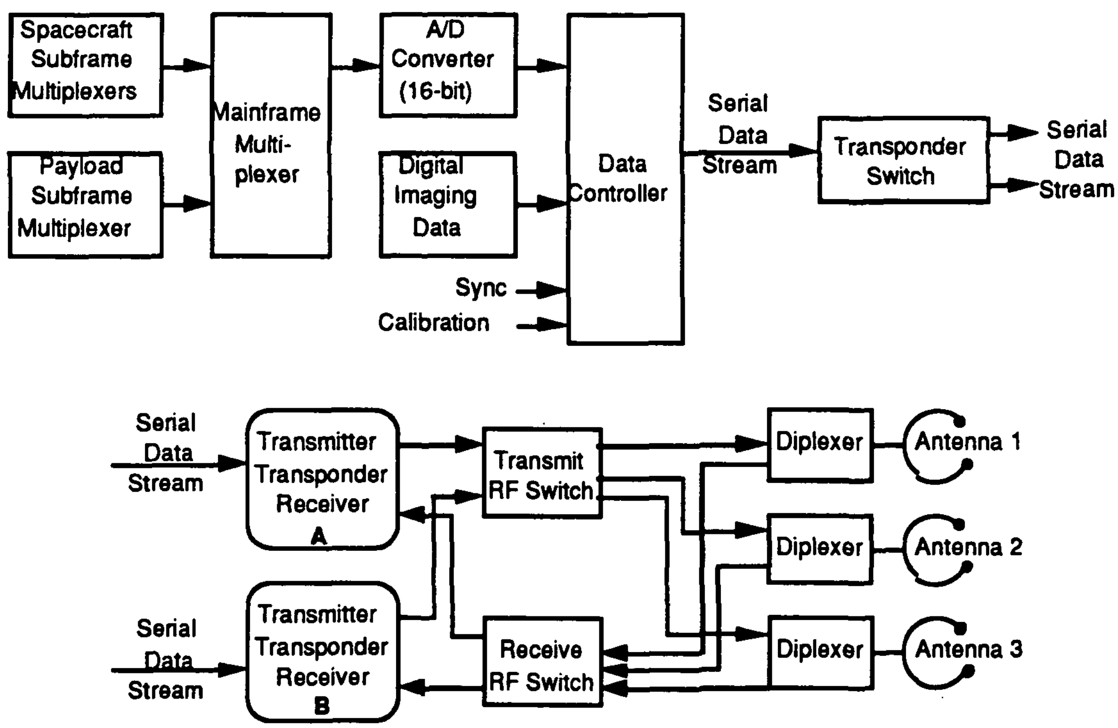


Figure 3.2.3-2: Satellite Data Handling and Communication System

Separate subframe multiplexers are used for data generated by the spacecraft bus and by the instruments package (Delgleish, 62). This data is then sent through the mainframe multiplexer and converted to digital format by a 16-bit A/D converter. This data is then combined with the digital imaging data produced by the cameras and routed as a serial data stream through a data controller to a transponder.

Communication with the control station is a critical function, and contingency safing under human control is preferred over autonomous safing. Hence, redundancy for the communication system is desired. Loss of the communication system would not be catastrophic, however, if the satellite is capable of autonomous safing. Redundant data controllers and transponders are also massive (~5 Kg). Therefore, the DSS team has chosen to recommend only dual redundancy for the transmitters. As shown in Figure 3.2.3-2, this redundancy is provided by using dual transponders with parallel transmission and reception paths (Wertz, 338). The active transponder is selected by the transponder switch.

An RF transmit switch and an RF receive switch select one of eight antennas for transmission and reception of data. The antennas are equipped with diplexers to allow both transmission and reception using a single antenna. The eight satellite antennas are circular normal-loop antennas approximately 0.02 meters in diameter. These antennae are small, lightweight, low-gain antennae, capable of omni-directional reception and transmission (Johnson, 55). For these reasons they are best suited to fulfilling the AERCAM communications requirements (Ling, 10/9/92). Their configuration on the satellite bus is shown in Figure 3.2.3-3.

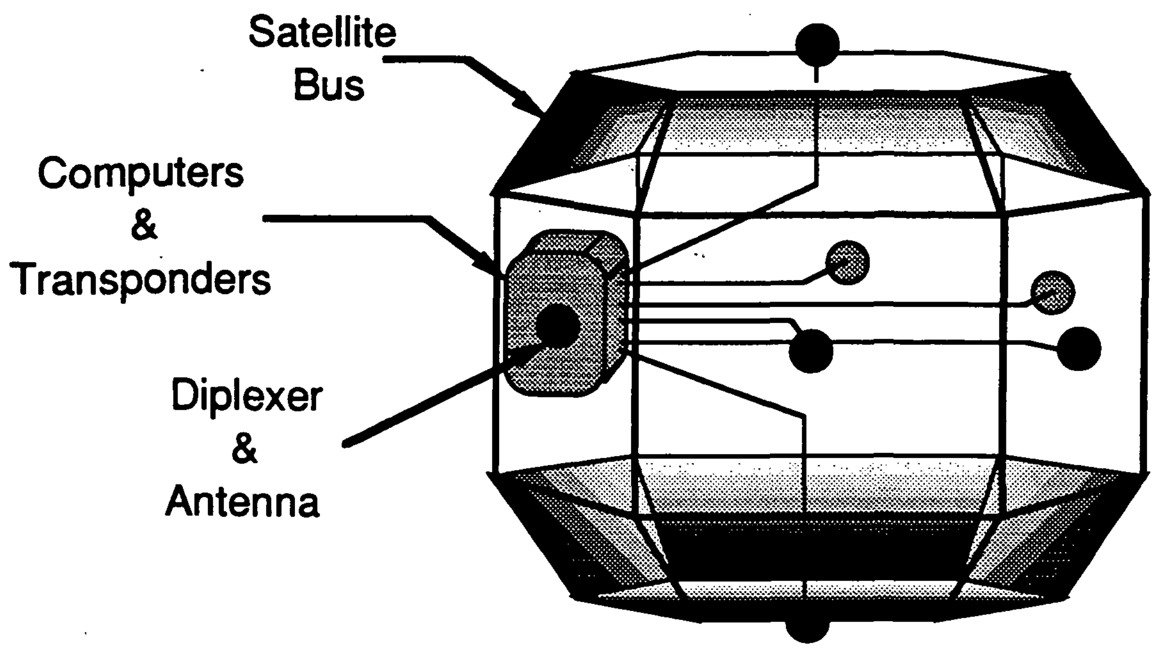


Figure 3.2.3-3: Satellite Antenna Configuration

The DSS design team recommends either Hollow tube or Optical fiber waveguides as the best hardware for transmitting the signal from the transponder to the antenna (Ling, 11/9/92). The waveguides, though larger than other transmission lines, are only 15.8 mm X 7.9 mm in cross-section (Johnson, 42-52). They also have excellent attenuation characteristics - 10 dB per 30.5 meters. The largest signal attenuation for the satellite would be less than 5 dB.

3.2.3.4 Station Communication System

Based on information provided by Mr. Bob Barber of NASA-JSC, the most feasible method for communication between AERCAM and SSF would require multiple transmitting/receiving antennae on Space Station. The system is based on the old "space-to-space" Ku-band communication originally proposed for the Space Station. Although the system is no longer baselined for SSF, it has had the most development to date, and therefore most closely meets the requirements of the SSF program office (Barber, 11/15/92).

The antennas on the Station are configured on "conformal arrays," similar in appearance to the mirrored balls in a dance club. Each conformal array holds a number of antennas and is capable of omni-directional transmission and reception. The arrays are 4.6" in diameter and 4.3" high, and they are mounted on 3 ft. booms. The antennas are right hand circular polarized, and for the purpose of the link budget were assumed to be 1 wavelength in diameter (~0.02 m). In the original system, the number of arrays ranged from a few to as many as eight to ten. For AERCAM, the DSS design team recommends using ten arrays - two on either end of the Station and two sets of four on each face of the Station (forward, aft, nadir, and zenith). A possible configuration, taking into account blockage on the Station truss structure, is shown in Figure 3.2.3-4. This configuration avoids interference by physical obstacles, and allows for communications all around the Station, including outside of the solar arrays.

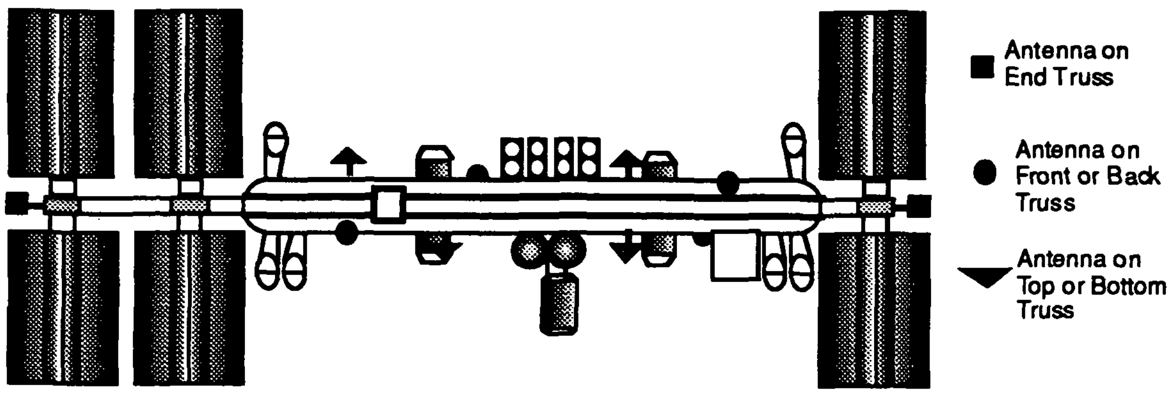


Figure 3.2.3-4: Station Antenna Configuration

The SSF system is capable of supporting up to 44 Mbps to/from a vehicle orbiting nearby (Barber, 11/15/92). It is also capable of supporting up to four vehicles simultaneously, and it was originally designed with the transfer of video data in mind. Therefore, this system, which has already had significant development, is well-suited to use for the AERCAM project, and may also be used simultaneously for other purposes.

3.3 Satellite Bus

This section describes the areas of design for the satellite bus. These areas include the satellite structure, the power system, and the propulsion system. Preliminary design for all of these areas has resulted in some sizing and performance estimates. Each respective section will describe the selection of the chosen system, present results from the preliminary design, and discuss the recommendations for AERCAM.

3.3.1 Structure

The main objective for the design of the satellite's structure was to achieve a small, simple, and modular structure. This objective was accomplished with a 20 sided hexagonal satellite. The hexagonal shape with 20 panels offers more directions for the collision avoidance sensors than a cylindrical or rectangular satellite. The design is small and simple while maintaining attitude control through three axis stabilization. The satellite will be modular and serviceable with removable panels and grapple and servicing ports.

The dimensions of the satellite are shown in Figure 3.3-1. The satellite is roughly one meter long, one meter tall, and one meter thick, and the volume contained by the structure is about 0.725 cubic meters. Figure 3.3-2 illustrates how the satellite looks externally while Figure 3.3-3 reveals the internal view of the satellite. Important subsystems, such as the instrument package, the computer, the propulsion system, and the power system, are shown in the hidden view. The subsystems are arranged within the satellite to meet the three axis stability requirement. The computer and camera systems are located opposite each other in the satellite, as are the battery and propulsion system. This creates a more symmetric mass and thereby reduces the mass products of inertia. The reaction wheels are at the center of mass of the satellite. The placement of the camera window, the communication antennae, the servicing port, the grapple port and the thrusters are included in the external view of the satellite.

Collision avoidance and proximity sensors will be placed on all panels of the satellite to provide omni-directional viewing of the environment for the satellite. Each panel will have several sensors in small clusters. The sensors will be connected to the computer and power system to allow for data transfer and electrical power, respectively. Visual, infra-red, microwave, and X-ray sensors, and laser rangers comprise a sensor cluster. Each sensor cluster is envisioned to be no larger than a human finger. Although their resolution is less than that of the primary imaging system, they will be capable of providing detailed ranging and translational and collision rates for nearby (~100 m) objects. This array of proximity detection sensors is being designed in more detail by groups at the NASA Johnson Space Center.

Modularity of the satellite will be achieved through removable panels and subsystems. The panels will be attached on the frame or skeleton, providing structural support for the satellite. The panels, themselves, may be composed of less structurally rigid material than the support structure, thus aiding in energy attenuation and perhaps reducing the weight of the satellite bus. Portions of the panels will be able to be lifted in order to detach sensor cables, and removed to allow access to the subsystem needing repair or replacement. The removable portion of the panel will not interfere will antennae or collision bumpers. Subsystems such as the computer, the imaging unit, and the reaction wheels will be able to be removed by detaching the cables. The cables, which will include both wires and tubing, will be designed for easy detachment. The ends of the satellite, where the propulsion and power systems are located, will be removable. Propulsion tubing and electrical wiring will also be detachable. This design specification will be complicated due to the amount of wiring and tubing in the end sections. In theory the propulsion and power systems can also be lifted from their frame supports to allow for repairs and replacements.

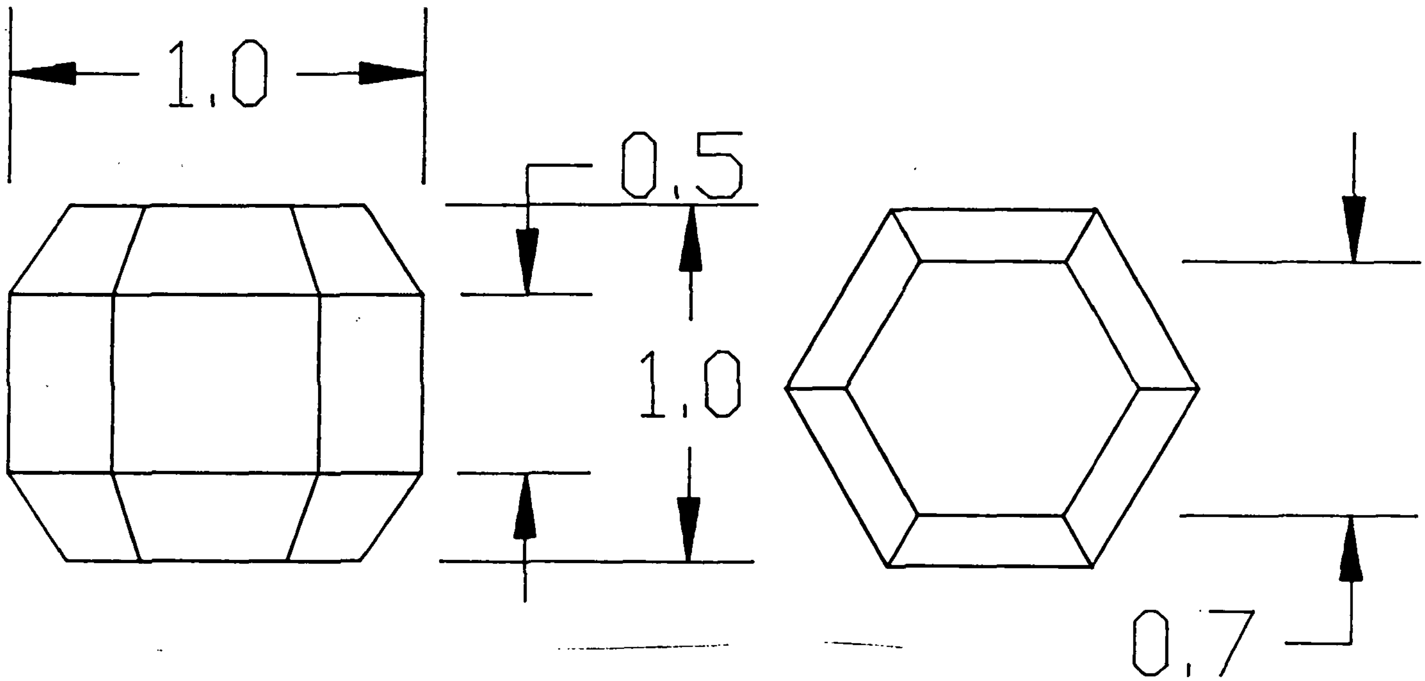


Figure 3.3-1: Satellite Dimensions in Meters

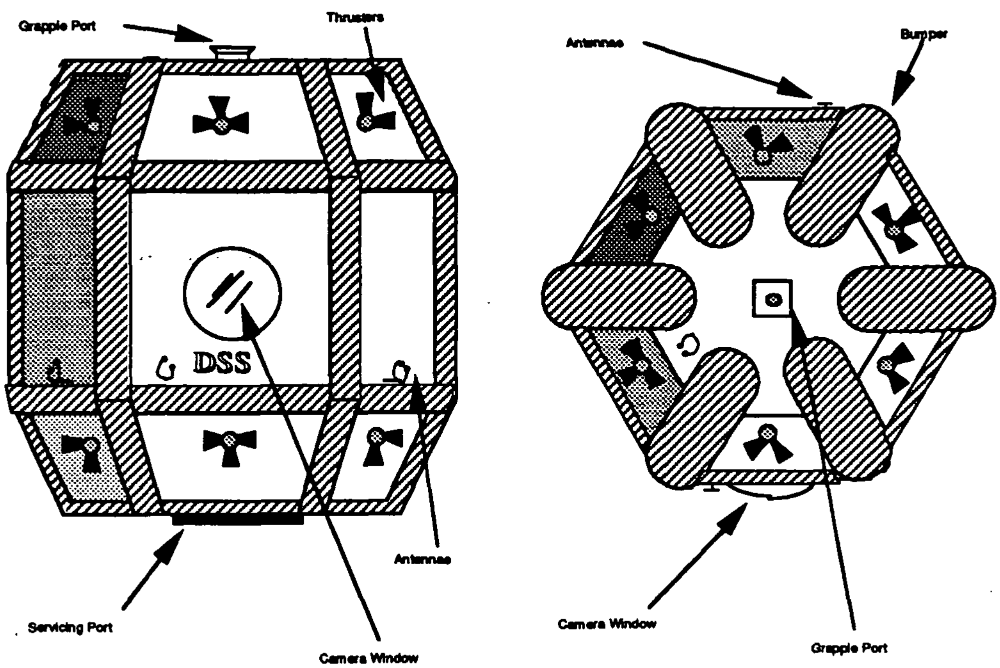


Figure 3.3-2: External View of Satellite
(Front view, satellite on its bottom; top view)

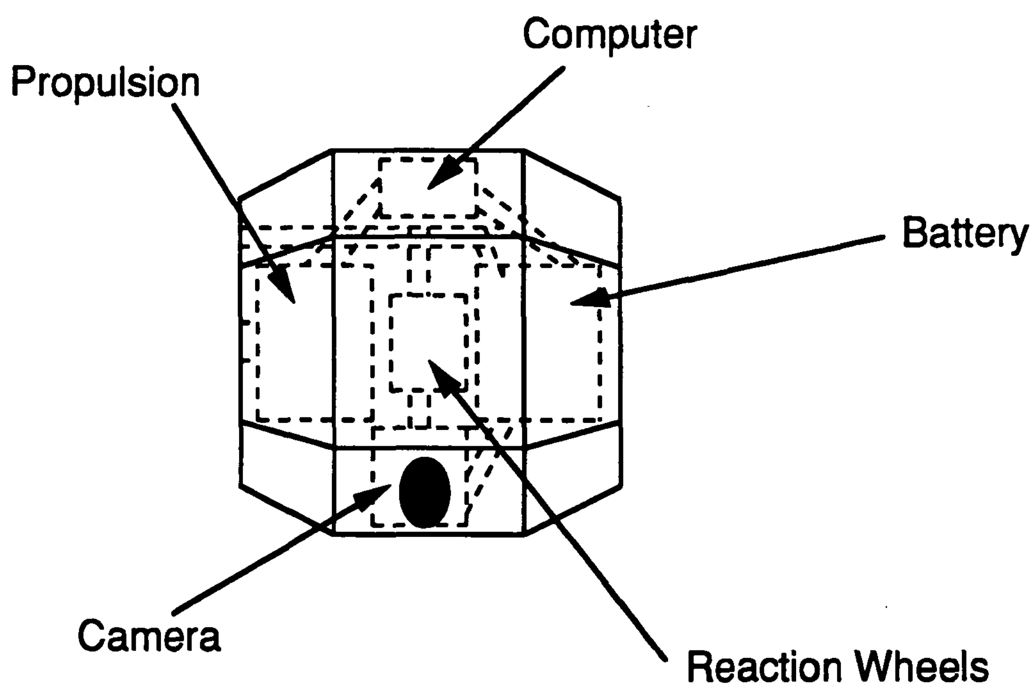


Figure 3.3-3: Hidden View of Satellite Subsystems
(Side view, satellite laying on its side)

The mass of the satellite was determined through the estimation of allowable masses for the subsystems (Wertz, 255). The mass variable subsystems are the power and propulsion systems, which vary due to different possible operation times. A small, medium, and large satellite were considered and would have operation times of four, six, and eight hours, respectively. Table 3.3-1 details these three versions.

Table 3.3-1: Satellite Bus Design Comparison

	Small	Medium	Large
Tank Size	0.5 ft^3	1.0 ft^3	1.5 ft^3
Propellant Mass	15.6 lbs	31.2 lbs	46.8 lbs
Battery Operation	4 hours	6 hours	8 hours
Battery Mass	15.4 lbs	23.1 lbs	33.1 lbs
Total Mass	129.6 lbs	153.8 lbs	180.1 lbs

The version DSS recommends for AERCAM is the small, four hour satellite bus. This design was chosen because of its relatively small mass and suitable operation time for AERCAM's mission. For this design the battery would have a mass of seven kilograms, and the tank would hold 0.5 cubic feet of propellant. Table 3.3-2 shows the mass budget for the recommended satellite with respect to the subsystems.

Table 3.3-2: AERCAM's Mass Budget for Small Version

Subsystem	Allowable Mass (Kg)
Structural Components	13.0
Propulsion w/out Propellant	8.5
Instrument Package	9.0
Comm & Data Handling	4.0
GN&C	5.5
Battery	7.0
Dry Mass Total	47.0
10% Mass Margin	4.7
Propellant (0.5 ft^3 of CO_2)	7.1
Total Mass	**58.8**

3.3.2 Power

The selection of the power system depended upon the system's ability to meet the requirements. The power system was required to provide a maximum of 40 Watts of output power which includes 20 Watts available for thruster firings and reaction wheel adjustments and 20 Watts reserved for camera and computer operation. The power system was also required to supply ample and consistent power while minimizing mass and size. Finally, the power system was needed to be simple and modular.

The satellite's main operating mode was an important consideration in selecting the power system. Three types of power systems were reviewed for the selection process: solar, fuel cells, and batteries. Table 3.3-3 shows a decision matrix that was used in determining the power system. The most important criteria for the selection were size, modularity, reliability, lifetime, serviceability, and mass.

Table 3.3-3: Power System Decision Matrix

	Modularity		Reliability		Lifetime		Size		Mass		Servicing		Total
	3		4		3		5		5		2		
Battery	5	15	5	20	4	12	3	15	2	10	4	8	80
Solar	1	3	1	4	4	12	2	10	3	15	5	10	54
Fuel Cells	4	12	4	16	2	6	3	15	2	10	2	4	63

A solar powered satellite is limited in several ways. To supply 40 Watts of power, a solar power system would need about 0.4 square meters of solar array surface area for a sun incidence angle of 45 degrees. For a small satellite with collision avoidance sensors, collision attenuation devices, and a camera window, surface area for solar arrays will be scarce. The close proximity of the satellite to the Station also prevents consistent sunlight which is needed by the solar arrays. For AERCAM operation, solar power does not prove to be an ample or reliable power source although a few solar cells could be used to provide supplemental power.

Fuel cells, which rely upon chemical reactions to produce electricity, have also been considered because they are reliable, and the products of the reaction could possibly be used as a cold gas propellant. Since most fuel cells provide power in the kilowatt range (more than twice our peak power requirement), and more analysis of the reaction product is needed, this power system will not be recommended as the primary power system.

Because the satellite will be used for specific purposes and will not be in operation for long periods of time, batteries can meet the power requirements. The battery can be recharged by the station during the nominal docked mode. Using batteries would simplify the satellite while eliminating the additional weight and cost associated with using solar arrays with batteries. A battery power system clearly outperforms solar and fuel cell power systems; therefore, batteries have been chosen as the primary power system.

The study and design of the power system has focused on the use of batteries as the primary source of power for the satellite. Appendix G details the selection process for the battery. After comparisons between Nickel Hydrogen (NiH_2) and Nickel Cadmium (NiCd) battery cells, the battery was chosen to consist of 24 NiH_2 cells which can provide 40 Watts of power. The Depth of Discharge (DOD), which is the percentage of the total battery power that is discharged in one cycle, and the energy density are 40% AND 60 watt-hours per kilogram, respectively. The DOD and energy density values were based on current industry space tested batteries (NASA, 631).

Loss of power to the computer or propulsion systems constitutes a catastrophic failure. Therefore, the power system has been designed for multiple fault tolerance. This has been accomplished by wiring the 24 cells in parallel and providing three separate power channels to each of the critical subsystems on the satellite. Limited cell failure, therefore, will not affect the satellite's immediate operation, or its ability to dock or perform avoidance maneuvers. In emergencies, 20 Watts of power is guaranteed to both the computer system and the propulsion/attitude control system. If power is available, the sensor arrays and the communication system are the next systems to be guaranteed power.

The mass of the battery depends upon the discharge time, and, for a four hour, six hour, and eight hour battery, the masses are 7.0 kg, 10.5 kg, and 14.0 kg, respectively. The four hour operation time is beneficial because it decreases the mass of the power system. The size of the battery is roughly 16 inches by 16 inches by 13 inches, and the volume of the battery is 1.9 cubic feet. The four hour battery is recommended because it can provide sufficient operation time with considerable mass savings over the six and eight hour batteries.

3.3.3 Propulsion

The design of the propulsion system involved the modifications of an existing system. The system modified was the Simplified Aid For EVA Rescue (SAFER). Developed by Mr. Dennis Wells of NASA, the SAFER is a two cubic foot propulsion device used by astronauts as a backup for EVA maneuvers. The SAFER has a 600 pound maximum capacity and can provide a ten feet per second delta-V at maximum capacity. The propellant is Nitrogen (N_2), and the propellant tank is 400 cubic inches in volume. The system has a single fault tolerance and 24 thrusters providing one pound of thrust each (Wells[1], 10/92). The SAFER is modified through a decrease in the mass capacity thus providing a larger delta-V. Therefore, if the satellite's mass is less than the current 600 pounds, then the delta-V increases from the current ten feet per second. The design of the satellite in the range of 100 to 200 pounds facilitates this modification. The satellite is designed to be a three fault tolerance system because of the close proximity and maneuvers of the satellite with respect to the Station. A three fault tolerance system is considered to be extremely safe, and can be achieved by the redundancy of this system.

Propellant is another area of design modification for the SAFER. Different cold gases can be used in the place of the current N_2. Cold gases are propellants which have low specific impulses and are generally very safe. Cold gas propulsion systems rely upon the expansion of high pressure gases for thrust (Wertz, 583). Cold gases that the DSS team studied are freon, argon, carbon dioxide, nitrogen, and methane, and the specific impulses for these gases are 55, 57, 67, 80 and 114 seconds, respectively. For details on the propellant study, refer to Appendix H.

Because of its performance and availability, carbon dioxide was chosen to be the propellant for the propulsion system. Carbon dioxide is a waste gas on SSF; therefore, it is readily available. Performance included the propellant mass needed for a delta-V and the volume needed to contain that amount of mass at 3500 psia. Carbon dioxide performed well, with only freon performing better. Tank size will vary with the amount of propellant being stored, and for carbon dioxide at 3500 psia, the propulsion system can provide about 30 delta-V's of 5 feet per second if the tank volume is 0.5 cubic feet. The tank size recommended is 0.5 cubic feet because of the mass savings over a larger tank with more propellant.

Another area of propulsion design is the sizing and positioning of the thrusters on the satellite. Positioning the thrusters so that the satellite has three axis translation and three axis rotation was a very important design objective. The thrusters should be sized such that they can provide a 100 feet per second delta-V as well as relatively small delta-V's for small attitude adjustments, station keeping, and proximity operations (Wells[2], 11/13/92).

The recommended 3 fault tolerant propulsion system will have 40 thrusters, with each thruster providing one pound of thrust. Figure 3.3-4 details the thruster positioning for one end of the satellite. The other end would have the same thruster placements. Figure 3.3-4 also shows the thruster placement terminology. The terminology is based on the panels of the satellite. For example, the camera is on the forward panel. Thrusters are placed on twelve out of the 20 panels, and they provide three axis translation and rotation. The thrusters can fire in six different directions with six thrusters pointing along each forward, aft , right , and left axes. Eight thrusters fire along each up and down axes. The thruster positioning for AERCAM is unique, yet it is based upon figures and terminology provided by Mr. Wells (Wells[3], 11/13/92).

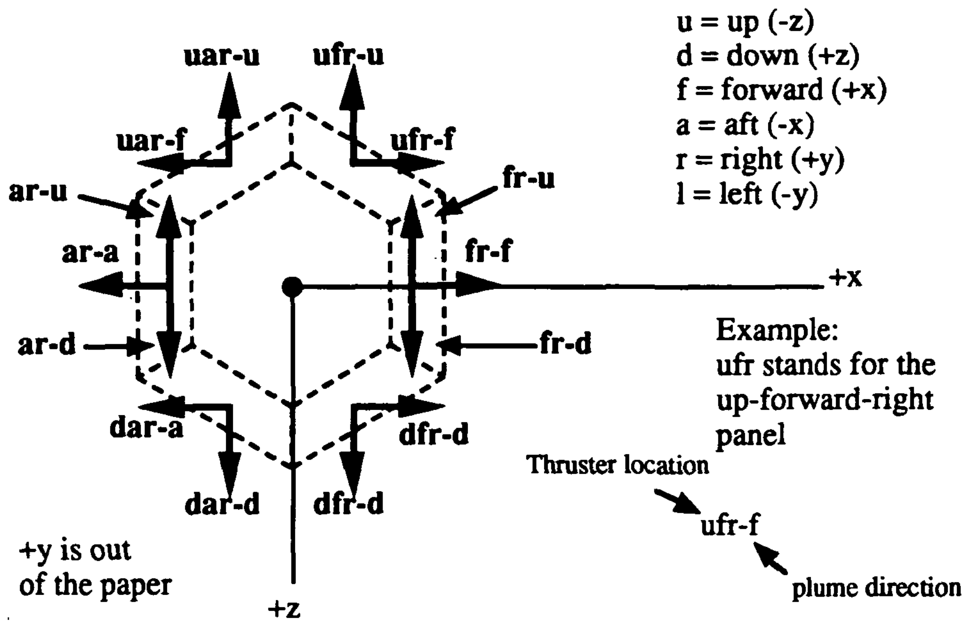


Figure 3.3-4: Thruster Positioning and Terminology

3.4 Docking and Servicing

The AERCAM docking and servicing system (ADSS) is a system that has evolved with the design of the AERCAM satellite. As the satellite went through the design process and gained and lost capabilities, new docking and servicing systems presented themselves, others became more viable, and yet others were deemed unfeasible. There were three docking and servicing scenarios associated with the final AERCAM satellite design. First, AERCAM could perform a controlled collision with SSF (i.e. self-dock). Also considered was the use of an extendible boom. This boom, of course, would have to be added to the design of SSF. The use of the boom would be as follows: it would extend from its nominally retracted position (either autonomously or with human guidance), "grapple" the satellite, and finally retract again.

Trade studies performed on the above two scenarios supplied the third scenario, which was ultimately recommended. This is the scenario termed the ADSS, and it involves the use of the Space Station Remote Manipulator System (SSRMS) and the Special Purpose Dextrous Manipulator (SPDM). It is a compromise between the two previously mentioned scenarios. The ADSS combines the minimal Station scarring of the self-docking concept with the use of a robotic arm to avoid having the satellite perform a controlled collision with SSF. In addition, docking of an AERCAM satellite would not be dependent on the reliability of an additional complex mechanism like the extendible boom. ADSS also stresses the use of standard or already existing/planned hardware for SSF. Use of a human controlled arm is the safest method of docking. AERCAM's sensor array, precision attitude control, and propulsion system can easily support autonomous or human assisted docking independent of the SSRMS and SPDM. This adds flexibility and reliability to the AERCAM docking method.

The ADSS uses the SSRMS and SPDM combination, controlled by a crew member, to grapple the AERCAM satellite. The SPDM has the standard robotic end effector shown in Figure 3.4-1, has multiple degrees of freedom, and is mounted on a track running the length of SSF. Hence, it is easily capable of supporting a variety of docking platform locations.

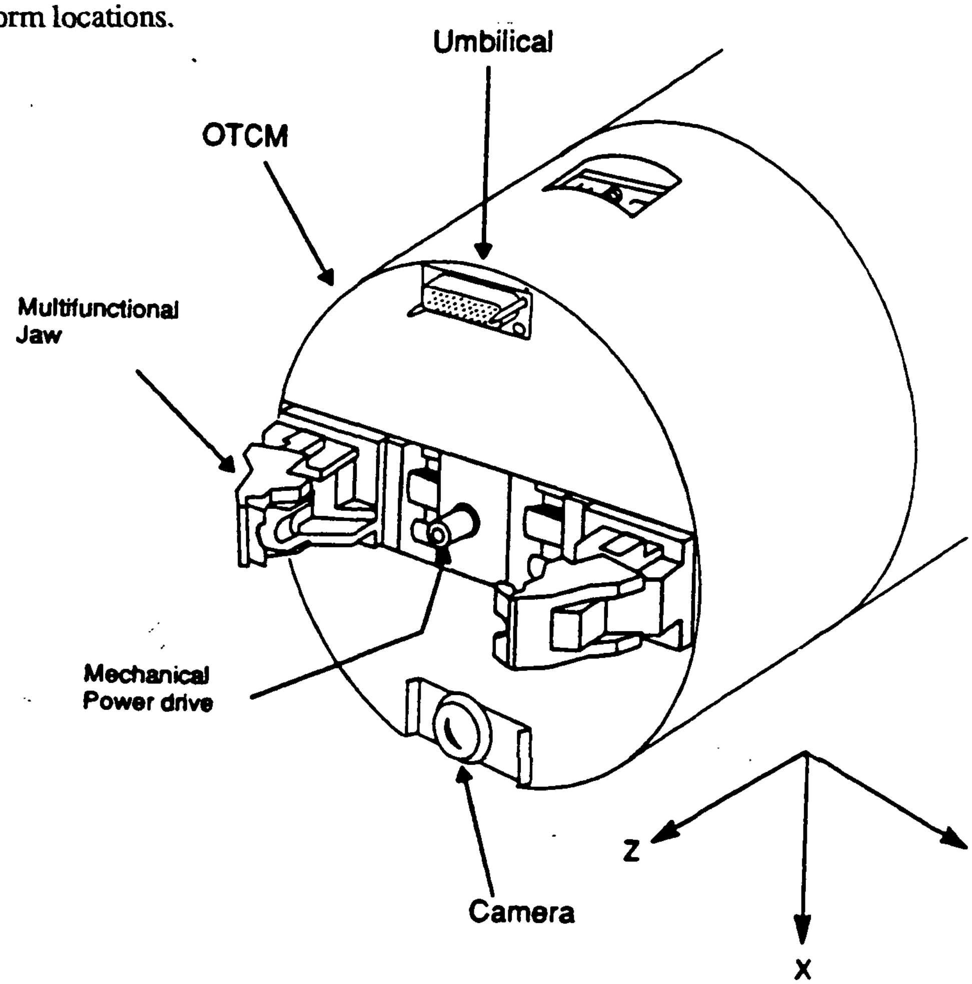


Figure 3.4-1: Standard Robotic End Effector
(Source: Robotic Interface Standards, 4-2)

The use of a standard end effector allows for the use of a standard grapple fixture, such as an H-Handle fixture or a Micro Interface fixture, both shown if Figure 3.4-2. The Micro interface, being of less weight and smaller footprint, was chosen, and its dimensions are shown in Figure 3.4-3. In addition, a dextrous handling target must be used with either of the grapple fixtures considered. The Micro Interface and dextrous handling target combination is shown in Figure 3.4-4. Detailed design specifications on the Micro Interface can be found in Appendix J.

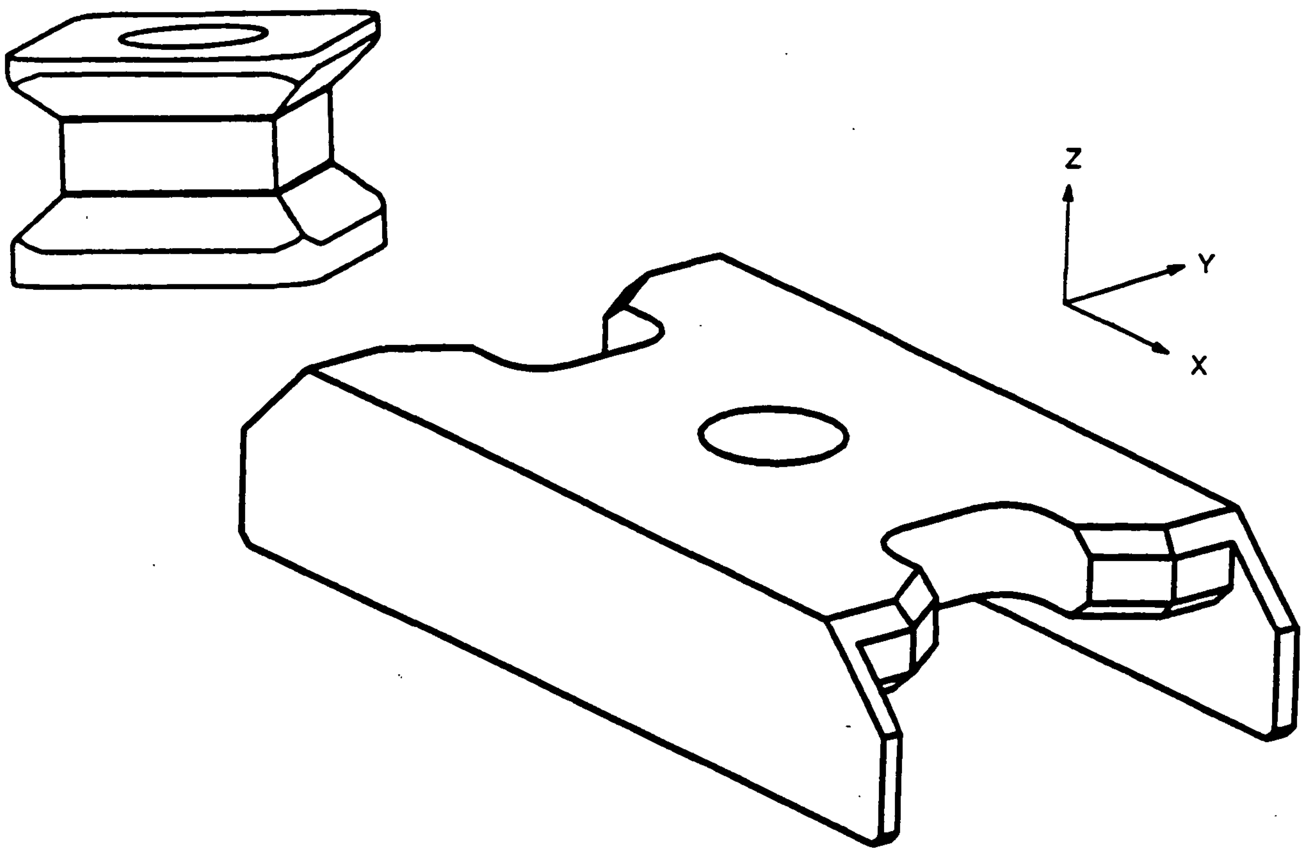


Figure 3.4-2: Grapple Fixtures
(Source: Robotic Interface Standards; 3-8, 3-13)

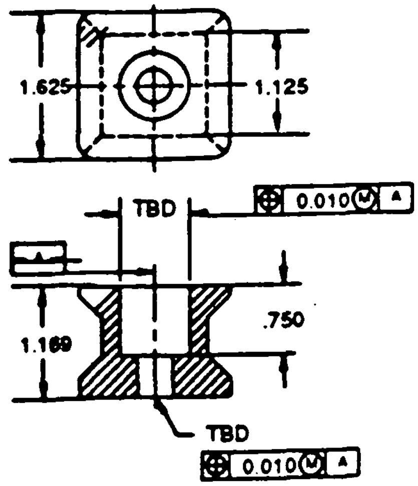


Figure 3.4-3: Micro Interface Dimensions
(Source: Robotic Interface Standards; 3-14)

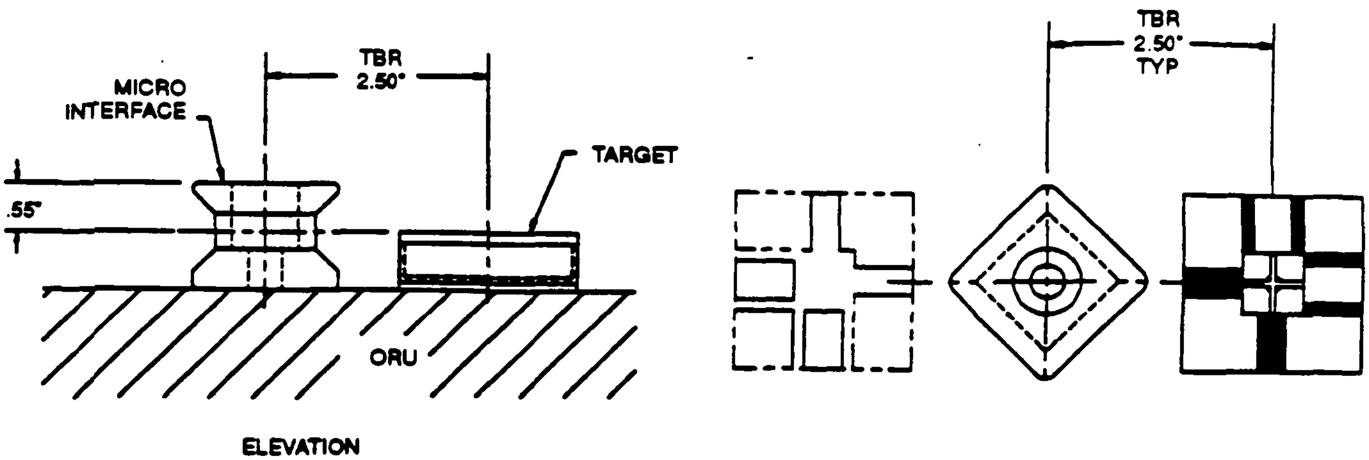


Figure 3.4-4: Integrating the Dextrous Handling Target with a Micro Interface
(Source: Robotic Interface Standards; 3-34)

Once the SPDM has retrieved the AERCAM, the satellite would then be placed on a docking platform. This platform is a standard attach platform which will be widely used on SSF for such applications as the cryogenics tank modules, propulsion modules, and dry cargo carriers. The platforms are pre-integrated into each truss segment as needed while on the ground, and then remotely deployed once on orbit. In its launch configuration, the attach platform is stowed into the truss without interfering with the truss diagonals. Once on orbit, the platform's deployment is controlled by torsion springs, and a locking device locks the support leg in position. Figure 3.4-5 shows the attach platform in both its stowed and deployed configurations. An attach platform has three alignment guides and a single capture latch. Resizing of a standard-sized attach platform may be necessary to accommodate an AERCAM satellite. Deployment of an AERCAM satellite would be accomplished in a straightforward manner. After the capture latch opened, releasing its grasp of the satellite, the cold gas thrusters would propel the AERCAM away from its docking platform readying it for service.

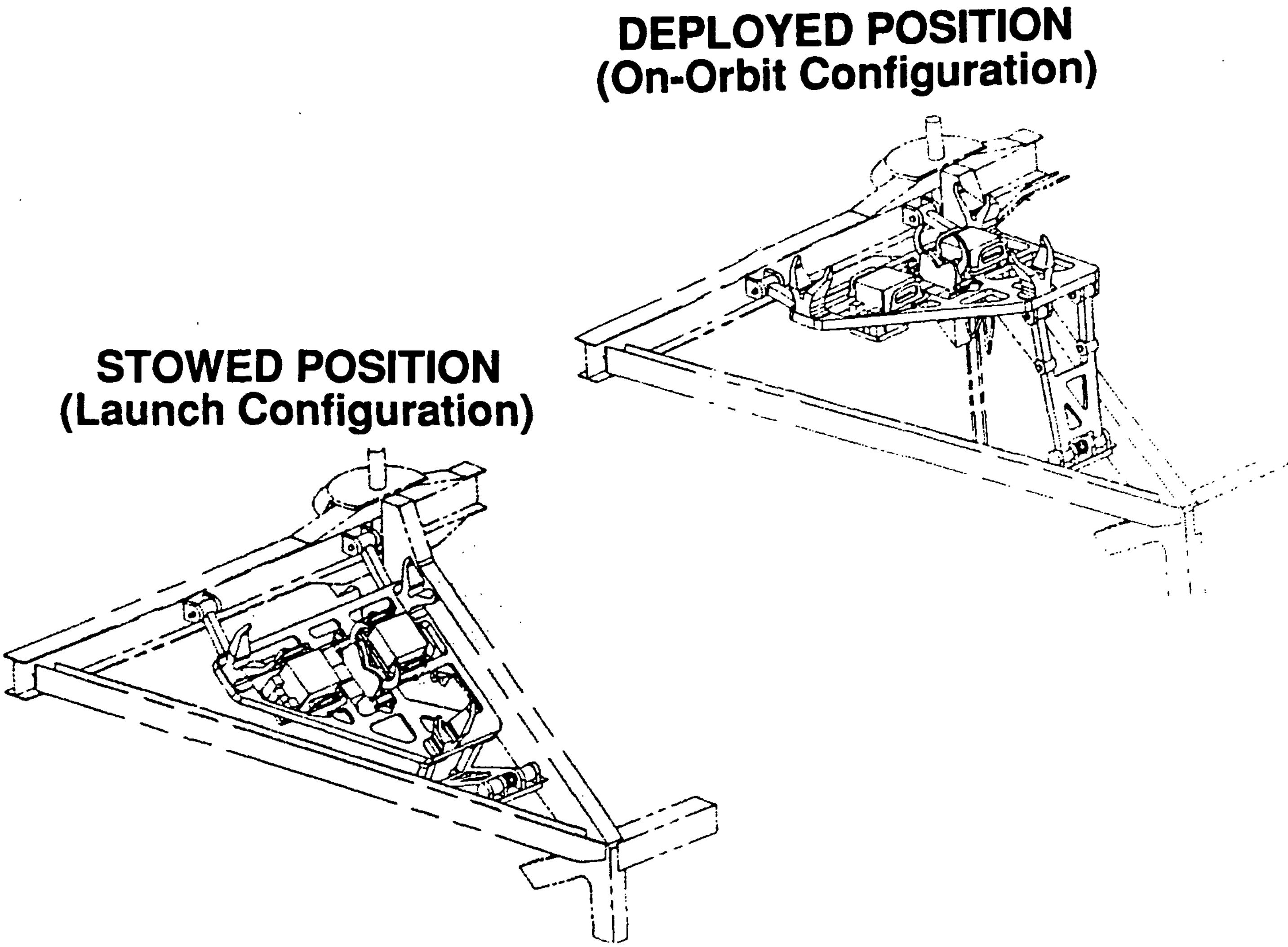


Figure 3.4-5: Attach Platform Configurations
(Source: Delta PDR, Book 2, 247)

The AERCAM servicing system consists of two basic areas, recharging and refueling. Recharging AERCAM's battery requires 122 Watts at 33.6 volts. Standard on

SSF are 120 volt lines. AERCAM's fuel tank will be refueled with waste gas from SSF. The gas in the half cubic foot fuel tank is required to be at 3500 psi. There are two scenarios currently being considered for waste gas disposal on SSF. Hence, the design of the servicing portion of ADSS hardware has been postponed until a decision is reached on which scenario will be implemented.

The first scenario involves the continuous venting of low pressure CO_2. If this scenario is chosen, the ADSS servicing hardware would intercept a venting gas line(s) and siphon the CO_2 into a separate compressor, raising it to 3500 psi. From the compressor, the gas could then be sent to the satellite's fuel tank.

The alternate scenario being considered is termed the Supplemental Reboost System (SRS) and the decision regarding its implementation has been deferred until approximately 1994, according to Mr. Scott Baird (NASA/JSC). The SRS plans to use waste gas from SSF to assist in reboosting the Station when necessary. In this scenario, waste gas refers to a mixture of CO_2 and methane in a 40-60 ratio. The gas mixture would be stored in a 5.6 cubic tank at 1000 psi in the Waste Gas Assembly (WGA), see Figure 3.4-6, for up to two days before being vented. The ADSS servicing hardware, in this case, would siphon waste gas directly from the WGA, compress it an additional 2500 psi, and fill AERCAM's fuel tank.

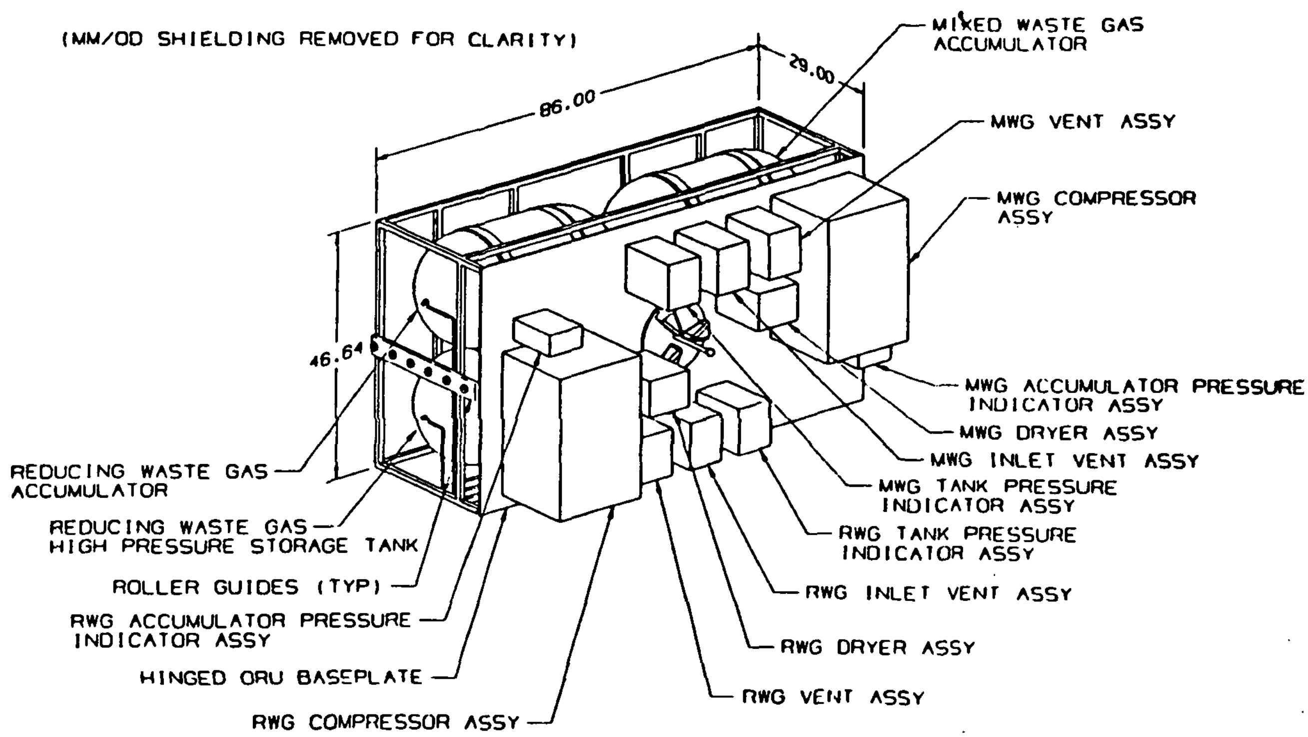


Figure 3.4-6: Waste Gas Assembly

(Source: Delta PDR, Book 2, 283)

The location of the AERCAM docking platforms is another design parameter that is to be determined by which waste gas disposal scenario is implemented. In the first scenario, where CO_2 is continuously vented, the ADSS servicing hardware would be located wherever it would be most convenient to interface with the waste gas disposal system. The docking platforms would be located as close to the ADSS hardware as feasible to minimize the amount of additional hardware required, such as flex and power lines. If the SRS is chosen, on the other hand, the servicing hardware, and therefore the docking platforms, would be placed in close proximity to the WGA. The WGA is to be located on SSF segment M1 (if implemented) as shown in Figure 3.4-7.

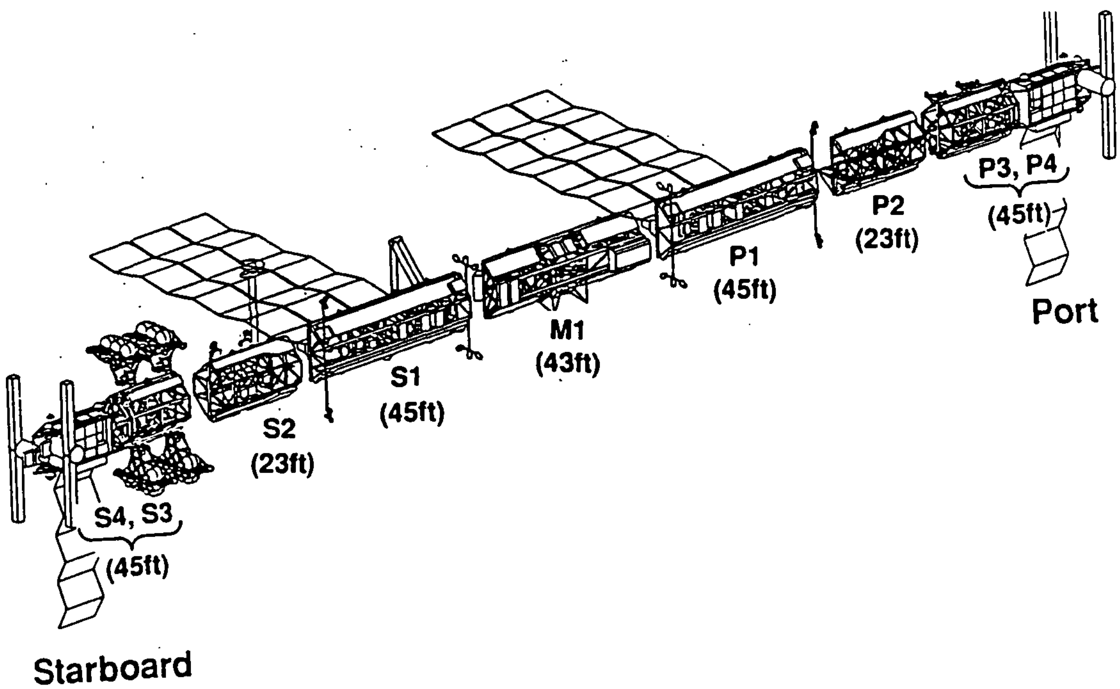


Figure 3.4-7: SSF Segment M1
(Source: Delta PDR, Book 1, 47)

3.5 GN&C and Orbits

In order for AERCAM to be a useful and helpful addition to the Station's environment, proximity operations and flight modes must be identified and defined. Attitude requirements for camera pointing must also be addressed. This section describes AERCAM's arrival and departure scenarios, flight modes, and attitude control methods.

3.5.1 Departure

AERCAM must leave the docking mechanism and the truss of SSF in order to enter the flight modes for autonomous operation. The docking mechanism will release and AERCAM will enter a free-flying mode. The cold gas thrusters will then fire a Δv of approximately 1 foot/second in the negative V-bar direction, shown in Figure 3.5-1, requiring a Δpropellant of 5.7 in^3 of CO_2. The thrust will move AERCAM away from SSF to the minimum operating distance of ten meters in approximately 30 seconds. The negative V-bar direction is advantageous because it allows AERCAM to translate along the axis with the least amount of decay of its position with respect to SSF.

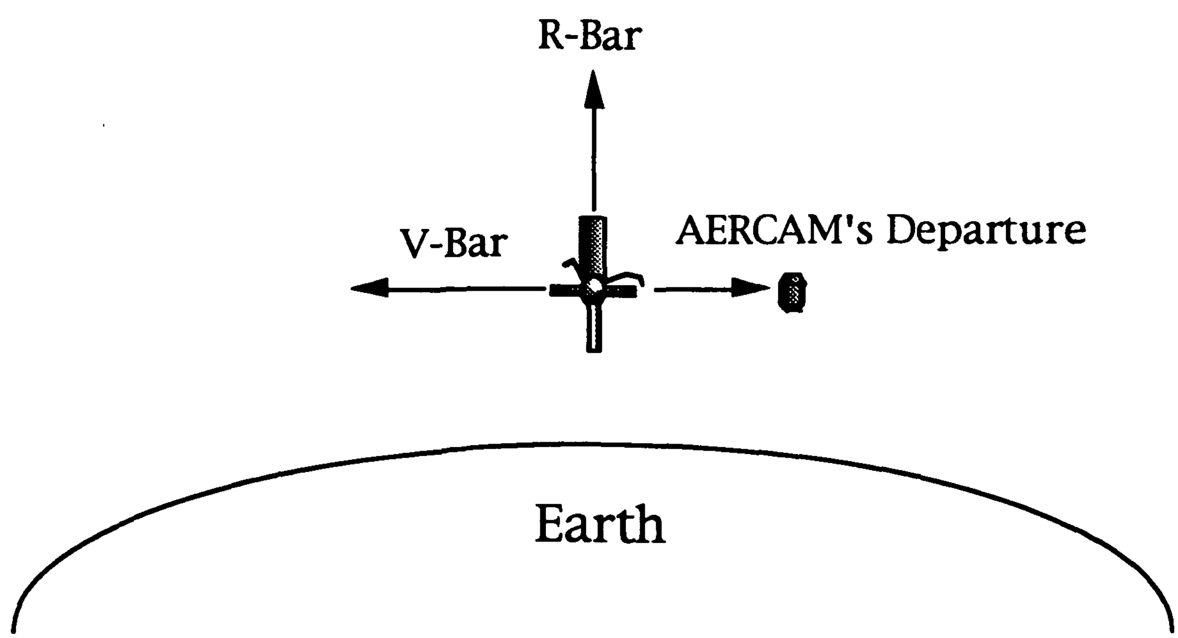


Figure 3.5-1: Departure Scenario

AERCAM will initially translate to a distance of ten meters from SSF. This distance is defined as the minimum distance from SSF in which AERCAM can safely operate autonomously. The translation will be made autonomously. That is a departure mode will be activated so that AERCAM will check the surroundings for any debris or astronauts, but will also obtain some type of clearance from a human operator. This way, AERCAM will provide consistently safe launches in a controlled fashion, and its final position will always be predictable. From this position, AERCAM will then enter one of the three operation modes defined below.

3.5.2 Operation Modes

Three independent modes of operation have been identified: translation, sub-orbiting, and position holding. AERCAM will always be operating in one of these modes, unless it is docking or deploying from the Station. Each of these modes will now be discussed.

3.5.2.1 Translation

Translation occurs when AERCAM moves in close proximity to SSF due to thruster firing. This mode is required for AERCAM to position itself for camera operation or orbit maneuvers. Translations can be controlled by the SSF command station with crewmember input directing AERCAM where it needs to go through a common hand controller. Under nominal operation, the maximum velocity of AERCAM relative to the Station is 1 foot/second.

Translation can also be activated through a predetermined set of commands entered by the crew at a command console, with the Δv's and trajectories determined using Clohessy-Wiltshire (CW) equations in the onboard software package. The magnitude of the Δv's required for translation to various locations is illustrated by the graph in Figure 3.5-2 from the data obtained from CWPROP, CW equation software that was developed by Don Pearson of Johnson Space Center of NASA.

Autonomous proximity detection and reaction systems will always be active and able to provide an acceleration of 1 foot/second/second in case a crewmember inadvertently tries to fly AERCAM through the truss of the Station. These systems may, of course, be overridden at any tome to allow for direct user control of thruster firings.

This graph shows that for AERCAM to translate the length of the Station in five minutes, a Δv of approximately 1.5 feet/second is required. This mode will allow the crew to enter the initial and final position of the satellite with AERCAM determining the flight path. These modes are crucial for AERCAM to operate around the Station.

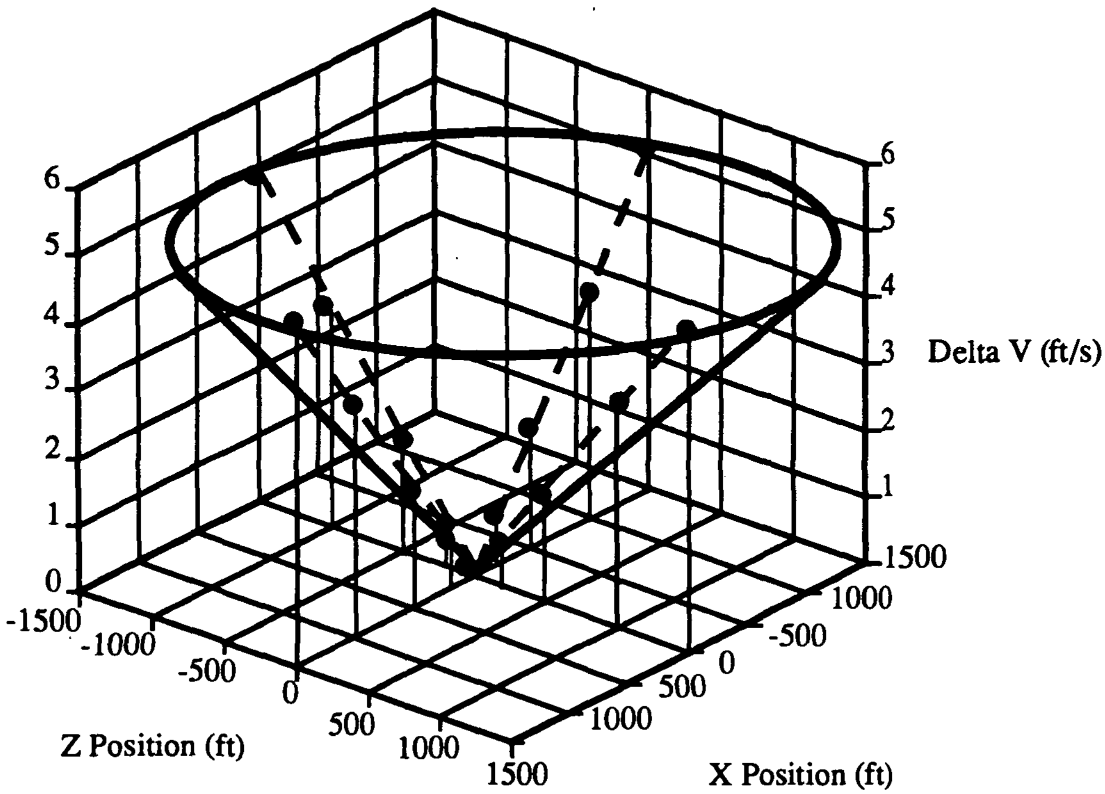


Figure 3.5-2: ΔV's Required for a Transfer Time of 5 Minutes

3.5.2.2 Sub-orbit

One of the two flight modes defined for video operation of AERCAM is the sub-orbit mode. AERCAM will orbit the earth in a near circular orbit 500 meters from the Station. A slight eccentricity in the orbit will cause AERCAM to orbit the Station as viewed in the SSF reference frame. The orbit was modeled with a TK Solver model (see Appendix K) using the CW equations. Figure 3.5-3 shows the motion of AERCAM around SSF as seen from fixed inertial space. The two by one ellipse appears when the Station is held to be the fixed inertial frame. A constant range can not be seen with the two by one ellipse. The constant range occurs when the Station and satellite are both moving, as seen in the Earth fixed reference frame.

This data is based on a constant distance of 500 meters from the station. This distance was chosen because it allows the entire Station to be contained in one image frame of the satellite cameras. Orbits at a closer range are also possible, but would not allow for viewing the entire Station in a single frame.

The orbit flight mode will give AERCAM the opportunity to get full views of the SSF from any angle. The orbit is designed such that the period of the satellite will match the Station's period around the Earth. It is important to note that the satellite's rotational period is equal to its period of revolution about SSF to aid in camera pointing. This orbit mode is the only way complete images of SSF will be possible unless they are taken from a shuttle during approach or departure.

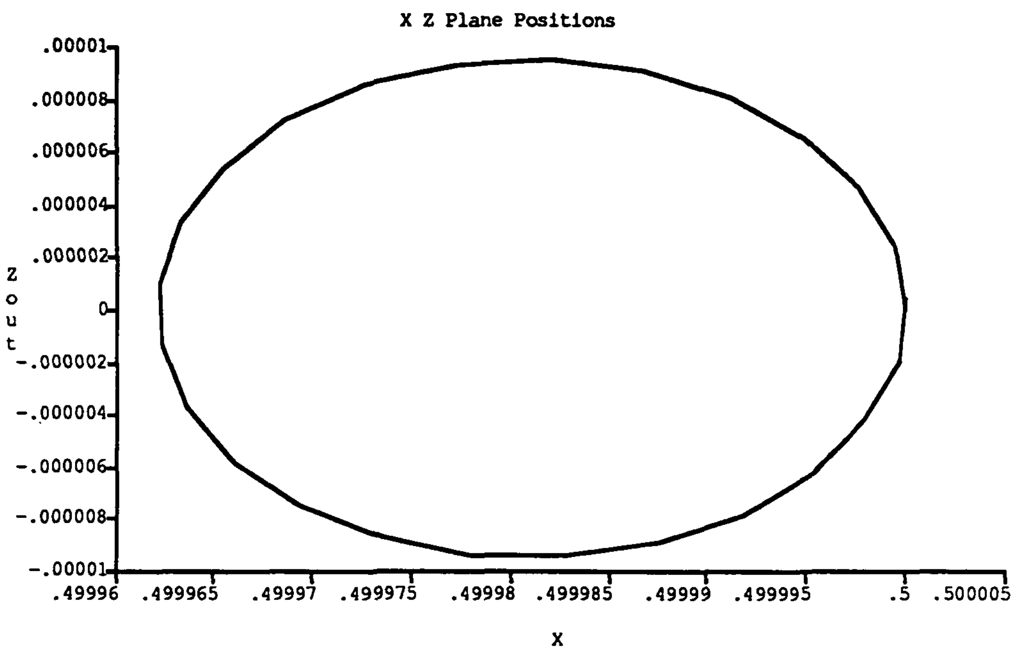


Figure 3.5-3: Orbital Motion of AERCAM about SSF

3.5.2.3 Station Keeping

Station keeping is a mode of flight where AERCAM fires a series a Δv's to maintain a constant position relative to SSF. This type of flight is invaluable to SSF maintenance and EVA maneuvers as it allows crewmembers and ground crews to observe all phases of an EVA or event that takes place on the surface of SSF. It also allows the crew and ground to observe a single element of the Station in great detail over a prolonged period of time.

This mode can be used at various distances ranging from 10 to 500 meters from the Station, depending on the resolution and field of view required for the images. It can also be sustained from any position relative to the Station, including locations beyond the solar arrays.

The Δv's required for this type of flight are on the order of 0.1 to 1.0 foot/second per minute. The frequency of thruster firing is important. Obviously, if the time between each firing is increased, then the Δv's must be increased as well. In addition, as larger time delays between firings are used, more perturbation of the flight will occur. Based on the way the Δv's are applied, the total Δv to return to a position never changes, but the direction in which it is applied does. In order to keep attitude control simple, it is desirable to reduce the time between firings as much as possible, so that the direction of the thrust is simple to determine. Figure 3.5-4 shows how this direction changes as the satellite drifts from the Station. Position 1 represents a small burn and small delay time between burns, and position 2 represents a large burn and large delay time. For this type of position holding, pulse-width modulated or pulse-frequency modulated closed loop regulatorless thruster systems should be used. These systems provide infinitely small time delays with infinitely small burn time, effectively holding AERCAM still at all times.

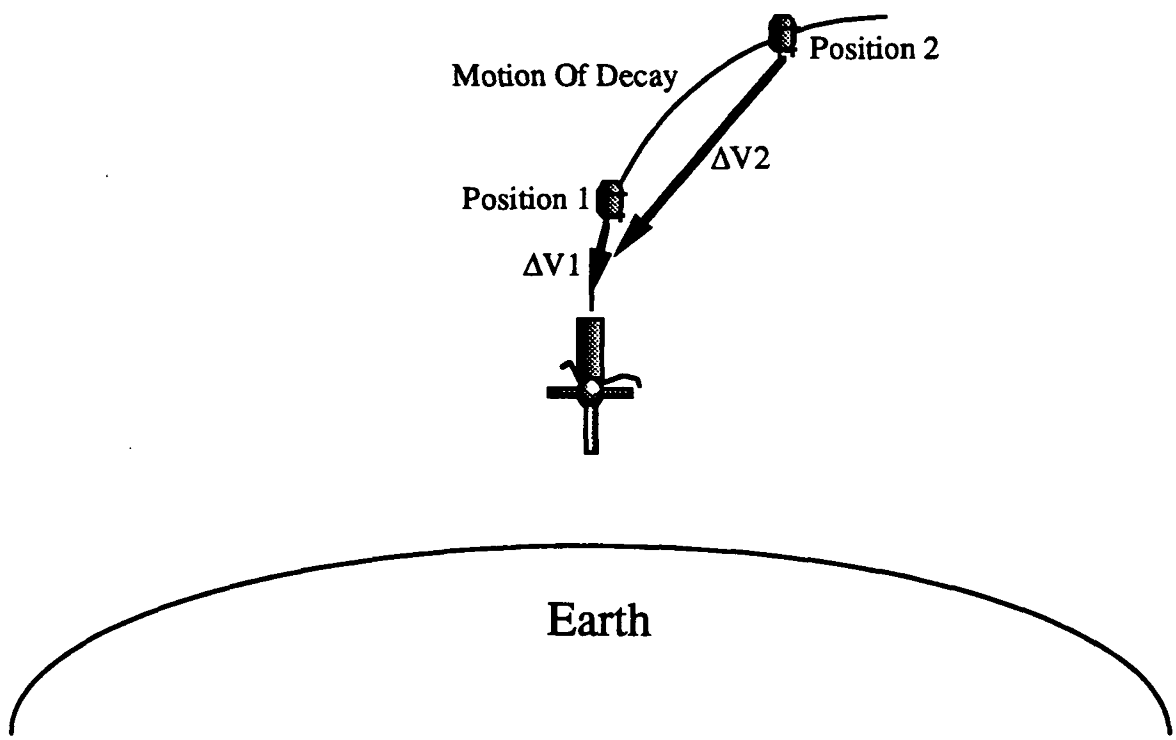


Figure 3.5-4 Decay of Station Keeping Positions

Nominally, through active pointing of the camera, AERCAM will be able to track objects which it is viewing. The camera will not be sensitive to drift of less than two centimeters from a viewing distance of 10 meter (d1 in Figure 3.5-5). This is equivalent to $\theta = \pm 0.1°$. DSS recommends attitude control, however, that will not reorient the satellite until a perturbation of greater than one meter is detected(d2 in Figure 3.5-5). This deadband can be re-specified by the user for other than nominal distances.

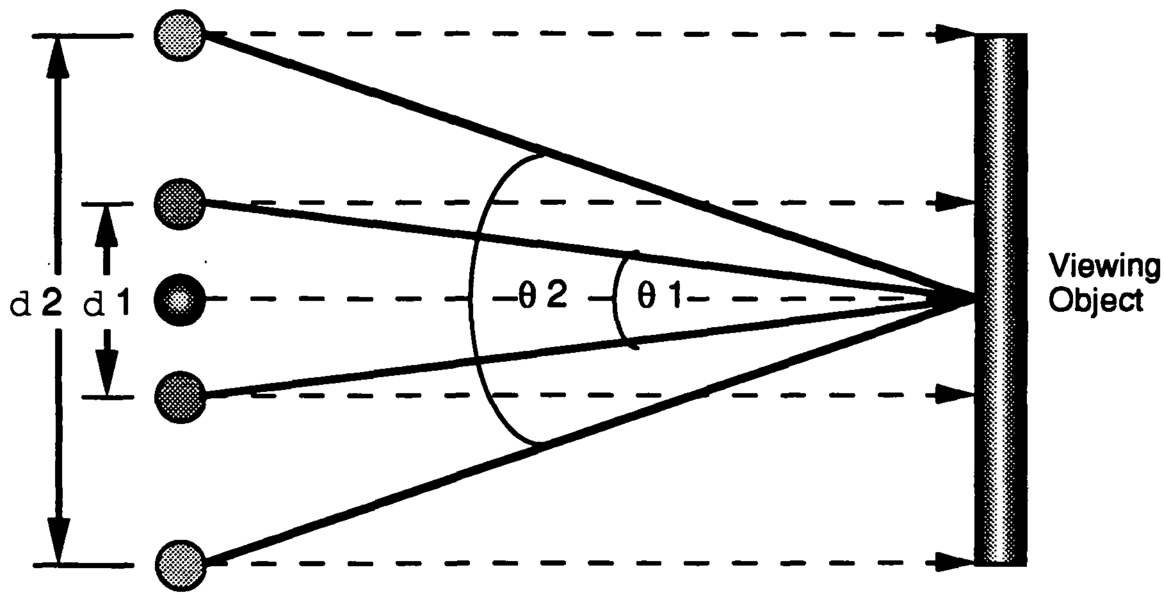


Figure 3.5-5: Viewing Object Deadbands

3.5.3 Arrival

To return from operations, AERCAM will move to a docking position 10 meters from the SSF truss. The stationary satellite will then be grappled by the SPDM which will maneuver the AERCAM to its docking platform. The docking position will be on the negative V-bar to reduce the drifting effects.

One area of concern for the project is how to allow the arm the safely enter the AERCAM control zone. Within this zone, all objects are to be identified and avoided. The arm must enter the zone, make contact with AERCAM , and then place the satellite on the docking platform. To do this safely, the arm, as well as the Station will be monitored with laser ranging and the proximity sensing array. The computer system, using image recognition and input from the sensors will determine if the arm is moving toward the AERCAM or if the Station and the arm are moving toward AERCAM. The only scenario that is acceptable to AERCAM is if the arm is moving toward the satellite. This will insure that docking is under control and that grappling is occurring as planned. If objects other than the arm are identified within the collision detection zone, or if they are determined to be moving at too great an impact velocity, the AERCAM will begin safing procedures, unless those procedures have been locked out by the user.

Image recognition will allow only the identified arm to grapple the satellite. The ranging and proximity sensors will determine relative velocities and alert for possible hard impact with the arm.

3.5.4 Attitude Control

The guidance system of GN&C is mostly autonomous. Attitude control of the satellite requires an active control system, which uses cold jet thrusters and momentum wheels to adjust the satellites spin rate and attitude, and reacts to small disturbances. Sun and earth horizon sensors are required for navigation and attitude determination for the satellite. Typical sizes of these sensors are shown in Table 3.5-1. As information regarding the proximity of AERCAM to the Station is determined, corrections to the attitude must be made. The cold gas thrusters provide the gross adjustments, and the reaction wheels provide the fine adjustments. It is important to note that the reaction wheels will be located at the center of mass of AERCAM so the they will only impart pure rotation and not any translation. There will be four reaction wheels positioned in a pyramid shape with three sides and a bottom. This will provide two fault tolerance.

Table 3.5-1 Sensor Sizing

Sensor Type	Weight (kg)	Power (W)	Accuracy (degree)
Sun	0.2-1.0	~0-0.2	0.1
Earth (horizon)	2-3.5	2-10	.05

3.6 Safety

The AERCAM satellite will operate in an orbit ranging from 10 to 500 meters away from SSF. This close operating range makes colliding with the Station a possibility. Station safety is, therefore, one of the primary concerns of the AERCAM project. Being considered safe for SSF is defined here as having a negligible effect on SSF should a contingency situation arise.

To try to ensure that contingency situations do not arise, AERCAM is being designed with three fault tolerance and a collision avoidance system. However failure modes must still be considered, and two failure scenarios have been identified. The first scenario is loss of the command signal. This failure could occur due to a Station structure blocking the signal, a communications failure on the Station, or a communications failure aboard AERCAM. In this event AERCAM's fail-safe mode would engage and thrust the satellite away from the Station using its last known position as a reference point. The second failure mode involves a critical failure in all or part of the GN&C system, which could be caused by a micrometeorite strike, for example. In this scenario, control of the satellite would be lost both remotely and autonomously and a collision with the Station could occur. The problem that must then be addressed to make the AERCAM satellite safe is collision attenuation.

3.6.1 Fault Tolerance

Failure of a number of AERCAM's systems could jeopardize the success or safety of the mission. These system have been identified and designed for multiple fault tolerance where possible.

3.6.1.1 Critical Failures

Failure of the attitude control and communications systems, as well as the proximity sensor arrays, constitute critical failures. These would require aborting the AERCAM mission and perhaps entail the loss of the satellite. Autonomous safing and knowledge of the latest world model will allow for the satellite to fire away from the Station even in worst case scenarios. Therefore, these systems have been designed for some fault tolerance where such tolerance could be implemented without significant impact to the satellite.

The reaction control wheels are designed using four wheels in a pyramidal configuration. This provides two fault tolerance. If the reaction wheels fail, however, the propulsion system will still be available for attitude control.

The communication system, with eight diplexed antennas, has been designed for multiple fault tolerance as well. The communication system employs dual transponders with parallel transmission and reception paths to provide for single fault tolerance. Although loss of the communication system may result in loss of the satellite, automated procedures on the satellite will still be able to ensure the safety of the Station.

Multiple sensor clusters are located in various positions around the satellite, providing fault tolerance if one or more of the clusters should fail. Even if the entire sensing system failed, knowledge of the latest world model would give AERCAM sufficient information to determine a safe trajectory away from the Station.

3.6.1.2 Catastrophic Failures

Failure of the power, computer, or propulsion systems could result in catastrophic failures, implying not only loss of the satellite, but the possibility of an unavoidable collision with the SSF. DSS determined that three fault tolerance was best possible safing that could be practically provided. This level of fault tolerance is remarkable and generally difficult to achieve in space systems.

The computer system, therefore, has been designed for three fault tolerance through use of redundant units, as described in section 3.2.1.2 of this report. The computer system is also capable of locking out data from the imaging system, to avoid tying up key computer activities in emergency situations.

The propulsion system, through the use of 40 small thrusters, also provides three fault tolerance. In fact, three complete thruster sets are provided in this configuration.

Finally, the power system is equipped with 24 battery cells to provide multiple fault tolerance in this system. Loss of power to a crucial system would negate the fault tolerance of that system. Therefore, the computers and propulsion system are connected to the batteries through four separate, redundant power channels. These systems are also given highest priority for power usage, allowing the other systems to be locked out from the power supply if necessary.

3.6.2 Collision Avoidance

Avoiding collisions is obviously the most desirable scenario for Station safety. In order to avoid collisions, the AERCAM satellite is equipped with an array of sensors which continuously maintain a collision detection zone (CDZ) around the satellite, as shown in Figure 3.6-1. The sensors which monitor the CDZ are arranged in miniature clusters on multiple faces of the satellite to provide omni-directional viewing. The sensors being considered for use in this application are microwave, infrared, X-ray, optical, and laser ranging. This sensor system is currently under development by Mr. Dennis Wells of NASA.

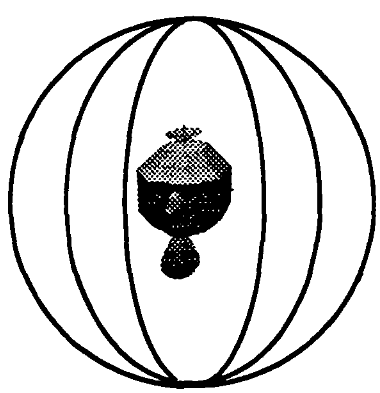


Figure 3.6-1: Collision Detection Zone

AERCAM's onboard computer will employ image recognition software so that it can identify any object or portion of the Station that penetrates the CDZ. This allows AERCAM to know exactly where it is in relation to the Station and then find a free path away from the Station should the need arise. The proximity detection and reaction systems identify hazards based on the proximity of nearby objects and the collision velocity those objects have relative to the AERCAM. When a hazard is identified, the AERCAM accesses a dynamic world model to identify the clearest trajectory to a safe location and the ΔV's required to ensure an adequately speedy departure for the hazard area.

The CDZ can be modified by the crew/ground to allow for proximity operations such as grappling by the SSRMS, and proximity detection and reaction system can be shut down altogether, if the user so desires. Though the satellite would maintain an updated world model, all maneuvering would require direct human control, and the satellite would cease autonomous response to its environment.

Loss of communication with the user control station is identified upon loss of a standard heartbeat, constantly monitored by the AERCAM communication system. This triggers the AERCAM to automatically begin safing procedures, including reactivation of the CDZ if it has been inhibited. This safing procedure is standardized, resetting all user-modified characteristics of the automated proximity detection and reaction system.

3.6.3 Collision Attenuation

In the event that AERCAM is unable to avoid a collision with SSF, AERCAM is equipped with an energy absorption device. The purpose of this device is to make the impact safe for the Station. For this to happen, the satellite absorbs all or most of the impact energy itself. The three energy absorption devices that were considered included an active airbag system and two passive shielding systems.

The active system is modeled after an automobile's airbag system. In this concept, airbags (Figure 3.6-2), would be inflated on the appropriate side of the satellite prior to any impact that AERCAM deems to be unavoidable.

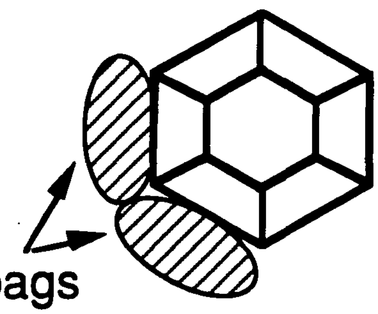


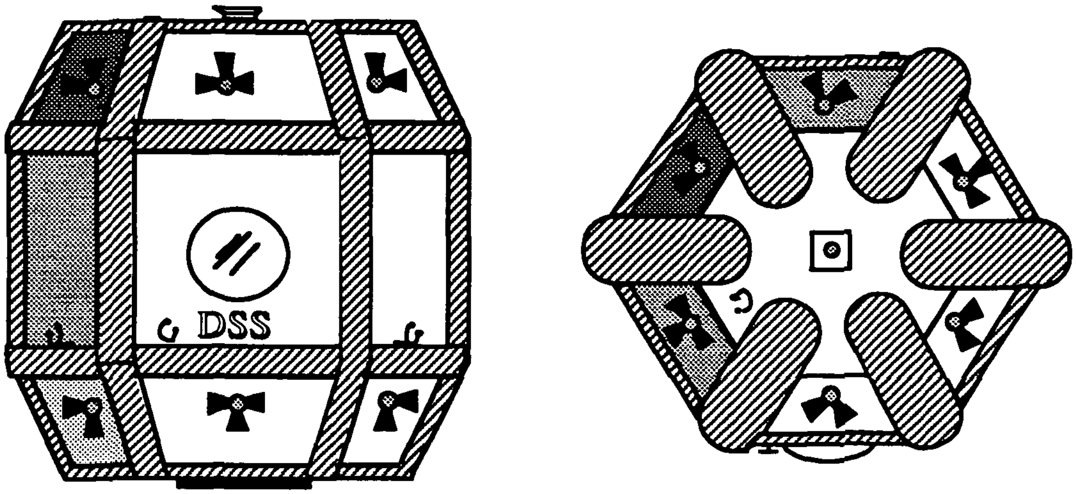
Figure 3.6-2: AERCAM Airbag Concept

This airbag system would differ from the airbag system in a car. In a car the gases are injected into the bag at about 130 mph (Wang, 90). Injecting gases at this rate in the satellite system could cause attitude stability problems; therefore, in the satellite system the airbags would need to be filled at a much slower rate. The advantage of this system is that it would absorb impact energy very effectively and cause very little damage to either the Station or the satellite. However, a typical automobile airbag requires 60 liters of gas to fill it, (Shelkle, 90) and AERCAM would probably need a bag of equal or larger dimensions to provide adequate protection, while only having a maximum 41 liters of gas onboard. This is also an active system requiring that the computer, sensor, and power systems all be functioning nominally. For these reasons, the airbag concept was considered nonviable.

The concepts for the passive systems involve using an energy attenuating material placed on the exterior of the satellite. One passive concept involves making the entire satellite skin out of or coating it with the energy attenuating material such as aluminum honeycomb or a double hulled plastic with a foam core. This method is not only passive, it requires minimal modification of the satellite bus structure. In addition, if some form of a strong double hulled skin were used, it would aid in attenuating the effects of a micrometeorite strike. Selecting a material with adequate particle dispersion and energy attenuation characteristics, which still meets the structural, mass, and thermal control requirements of AERCAM has been difficult, however.

The other concept just placing energy attenuating material at the vertices of the satellite. The three energy attenuating materials considered were gas, aluminum honeycomb, and foam. Foam, however, has not been space tested and it is uncertain if it is structurally stable in the space environment. Aluminum honeycomb could provide adequate energy attenuation with crushing strengths anywhere from 25 psi to 6000 psi but even the lightest aluminum honeycomb weighs lb./ft^2 (Bandak), and this could add up to

70 lbs to the satellite. Therefore, gas filled bumpers placed at the vertices of the satellite are considered to be the most viable choice for an energy attenuating system (Figure 3.6-3).



Side View

Top View

Figure 3.6-3: AERCAM with Energy Attenuating Bumpers

The gas of choice is an inert gas, such as nitrogen, and the bags would be compartmentalized and covered with a kevlar or mylar material to afford some protection from punctures. Although this system requires more maintenance due to leaks and punctures, than an aluminum honeycomb system, the savings in weight over aluminum honeycomb could be substantial. Each pressurized bumper could be individually attached to the satellite with a strong adhesive or velcro for easy replacement of punctured or failed bumpers.

3.7 Public Relations

The AERCAM satellite could be used to improve public relations for NASA and Space Station Freedom by promoting space activities. AERCAM would allow the general public to see their taxes spent wisely, decrease apathy toward space programs, and spark children's interest in space. Its automated functions and free-flying observation is a tangible display of the significant advances in space hardware and software. Its compact, powerful, and versatile systems offer an impressive improvement in space observation, and could be as revolutionary and profitable as the Imax camera has been. In addition, its application can be extended far beyond use on the SSF.

In conjunction with public relations, a school program using AERCAM would increase children's awareness and interest in space. Dr. Wallace Fowler at the University of Texas assisted DSS with the school program which is discussed below.

3.7.1 Assumptions

The AERCAM's nominal mode has been defined as deployment from SSF whenever it is needed. However, when the AERCAM is in the orbit mode, one or two revolutions could be dedicated to a school program to promote SSF to children and their parents. Although this is not the nominal mode of the satellite, occasionally it could be used for educational purposes. While in the orbit mode, the AERCAM will move around SSF in a nearly circular orbit with a constant angular rate so that the camera will always face the Station. The optics could be controlled from SSF or the ground. Ground control would be via a link through SSF's autocommunications system so that no crew time is required.

3.7.2 Opportunity

Although the amount of time available for the school program is limited, the opportunity for school children to control the optics is tremendous. The children could use special controllers brought to their schools via the NASA Space Exhibits (semi-tractor trailers). The controlling child, classmates, and children across the United States could watch and listen to the controlling child and SSF on split screen TV--either on NASA select or another dedicated TV channel.

To prepare for the experience, the school teachers would obtain learning packages from NASA which describe the parts and functions of the satellite and SSF, allow the children to build models of SSF and the satellite, and describe SSF operations. Models would allow the kids who do not get a chance to control the satellite to still be involved in the program.

3.7.3 System Components

The components of the system include the AERCAM satellite with controllable optics, control units that travel with the NASA Space Exhibits, and the SSF communication system which relays the information. The children would control optical functions such as zooming and scanning across the Station by using the Space Exhibit controllers. The controllers will be simple to use, but their fancy looks will attract the children's interest.

3.7.4 Scenario

A satellite naming contest could be conducted before SSF is completed to attract children to the idea of controlling a satellite as soon as possible. Any child could submit a name for the satellite, and the winner's school would get to be the first to control it. The name entries could be judged voluntarily by NASA employees.

Once AERCAM is in use by SSF and time is available for the school program, schools can submit applications showing their interest in participating in this program. Either a lottery system or an essay-type contest could be conducted to determine which school would have the next opportunity to control AERCAM. Those schools which do not get to control the satellite will be informed of the control time, and can view the child and the satellite's images on TV. Notice to the schools will be given as early as possible.

NASA Space Exhibits can travel across the country educating the students about Space Station Freedom, AERCAM, and space exploration in general. The representative can excite the children and let them know that they might get to control a satellite or at least watch one of their peers control one on TV. In addition, the exhibit can teach the children other things, such as attitude control, cameras, and time delay in the satellite's response. Children could also learn about jobs in space industries, such as how to become an astronaut.

3.7.5 Advantages

Advantages of using the AERCAM satellite for the school program are that it would introduce many children to Space Station Freedom, the AERCAM satellite, and space exploration. It would spark their interest in space and would introduce fun and interesting jobs they could do when they grow up. Getting the children's interest in space now is a large step in keeping their interest when they are adults.

3.7.6 Disadvantages

This program is highly dependent upon whether SSF needs to deploy the AERCAM satellite into orbit. If the need to deploy AERCAM arises only for emergencies, it will probably not be feasible to utilize the satellite for the school program. Even if AERCAM is deployed occasionally for non-emergency uses, deployment may occur at times that are not predictable. This would make it difficult to plan control times for a school.

3.7.7 Conclusion

The school program would be beneficial for NASA in promoting children's interest in space. However, since the concept of the AERCAM no longer allows for a major public relations program, Degobah Satellite Systems concluded its study on public relations with the preliminary design report.

4.0 Recommendations For Future Work

Although DSS has developed most of AERCAM's subsystems to the preliminary design stage, the limited time frame within which this project was undertaken meant that some subsystems had to be left at the conceptual stage and others not studied at all. The areas that require more detailed studies are outlined below.

4.1 Sensor Arrays and Sensor Control Logic

The design of the sensor arrays and its control logic has not been dealt with by DSS. The ongoing design and testing in this area is being done at NASA by Mr. Dennis Wells. DSS has, however, identified several areas which need to be developed. These areas include the command logic, the proximity detection and response logic, and contingency situation responses. The command and response logic would probably involve artificial intelligence, which will need to be carefully tested. Responses to contingency situations such as loss of communications, loss of proximity sensors, or loss of any major system also need to be developed. Another area of concern is the command hierarchy, i.e. who or which computer has final command authority over the satellite.

4.2 Energy Attenuation

The energy attenuation device that is outlined in this report has been left at a conceptual stage. Recommendations for future work in this area include more detailed studies on the following:

- the size of the bumpers and the internal gas pressure needed
- protecting the bumpers from puncture by micrometeorites and other space debris
- the maintenance costs of the energy attenuating device.

A study of impact velocities and structural stiffness of Station components needs to be conducted in order to determine the size and internal pressure of the gas bumpers that will be required. Also, since the Station is in an orbit containing many micrometeorites, the bumpers will need some degree of protection from puncture. It is recommended that kevlar and mylar be studied for this purpose. Finally, since some punctures will be unavoidable, the cost of maintaining the bumper system will need to be studied to finally determine the feasibility of the concept.

In addition, further studies regarding the use of an energy attenuating skin for the satellite might be undertaken. Although this method of energy attenuation was not chosen by DSS for the AERCAM baseline, if a material which meets the requirements in section 3.6 could be found, it might have considerable advantages.

4.3 Camera System

There are several areas in need of further research in the camera system. These areas are the onboard lighting, the flash shield, and the infrared system. In order to raise the existing light levels on the Station to a suitable level for AERCAM's camera, an onboard light source is recommended. DSS has been unable to find a suitable off-the-shelf, low-powered lighting system for this purpose. Therefore, a system will have to be designed to meet this need. The requirements for an onboard lighting system are twofold: 1) the light source should be of high intensity so that suitable light levels can be achieved even at a reasonably long distance from the Station and 2) it should have low power needs so that AERCAM's batteries are not unduly stressed. Also, in the event that the camera should accidentally point at the Sun, a flash shield over the camera window has been baselined and further research into this is required. More specifically, research into the coating required for the flash shield and its suitability to the harsh environment of low earth orbit could be undertaken. Finally, the infrared system needs to be more thoroughly researched, and a suitable IR cooling system still needs to be selected.

4.4 Communication System

Whether or not the Station can telemeter our data with its own data through TDRSS link needs to be investigated. If it cannot, then AERCAM would need to develop a dedicated antenna for transmission of data and commands through TDRSS to the ground. Also, locations for the conformal arrays (antennas) on the Station truss need to be selected based on availability of the truss for scarring and interference caused by nearby objects.

4.5 Station Waste Gases

AERCAM has been designed to utilize Station waste gas as a propellant. Unfortunately, the design of the waste gas disposal system onboard the Station has not yet been finalized. The two methods currently being considered by NASA are a low pressure venting of carbon dioxide and the storage of a mix of carbon dioxide and methane, to be used as an aid in reboost. DSS has fully investigated the use of carbon dioxide as a propellant. However, using the carbon dioxide/methane mixture as a propellant has not been studied and it is recommended that this study be conducted so as to be prepared for either design.

4.6 Thermal Control

Thermal controls on the satellite bus are essential for the proper operation of almost every subsystem. The two subsystems requiring the tightest thermal controls are the camera and computer systems. These systems will need to kept within very small temperature ranges. To achieve this DSS recommends investigating the use of both active and passive heat dissipating systems.

4.7 Station Scarring

The AERCAM concept requires two satellites to be stored and serviced onboard the Station. AERCAM also needs communications support from the Station. This makes an analysis of Station truss scarring essential to the success of the project. If low impact scarring cannot be achieved, the AERCAM project could be jeopardized. Therefore, an analysis of Station scarring is highly recommended.

4.8 Spacecraft Sizing

One final area of investigation which could significantly benefit the AERCAM project involves spacecraft sizing. The DSS design attempted, where possible, to identify small components which would not require significant power to operate. Unfortunately, miniaturizing a spacecraft requires intense research, which could consume an entire semester in itself. Therefore, the DSS team recommends future research into miniaturizing such key components of the spacecraft as the transponders, reaction wheels, batteries, propulsion system, sensor clusters, mass storage units, and computer processors.

5.0 Project Management

This section of the report outlines how the AERCAM team of Degobah Satellite Systems was organized. It reviews the management and task structure used for project operations up through PDR I, and it also presents the modified project structure and task assignments for the design effort between PDR I and PDR II. Finally, this section provides the schedule which has been used since the proposal to monitor completion of the project.

5.1 Design Team Organization

The organizational structure of the design team changed very little since the initial proposal. The management was structured to provide a reasonable span of control and tracking of responsibilities and to facilitate effective communication. The organizational structure of the AERCAM team is shown in Figure 5.1-1.

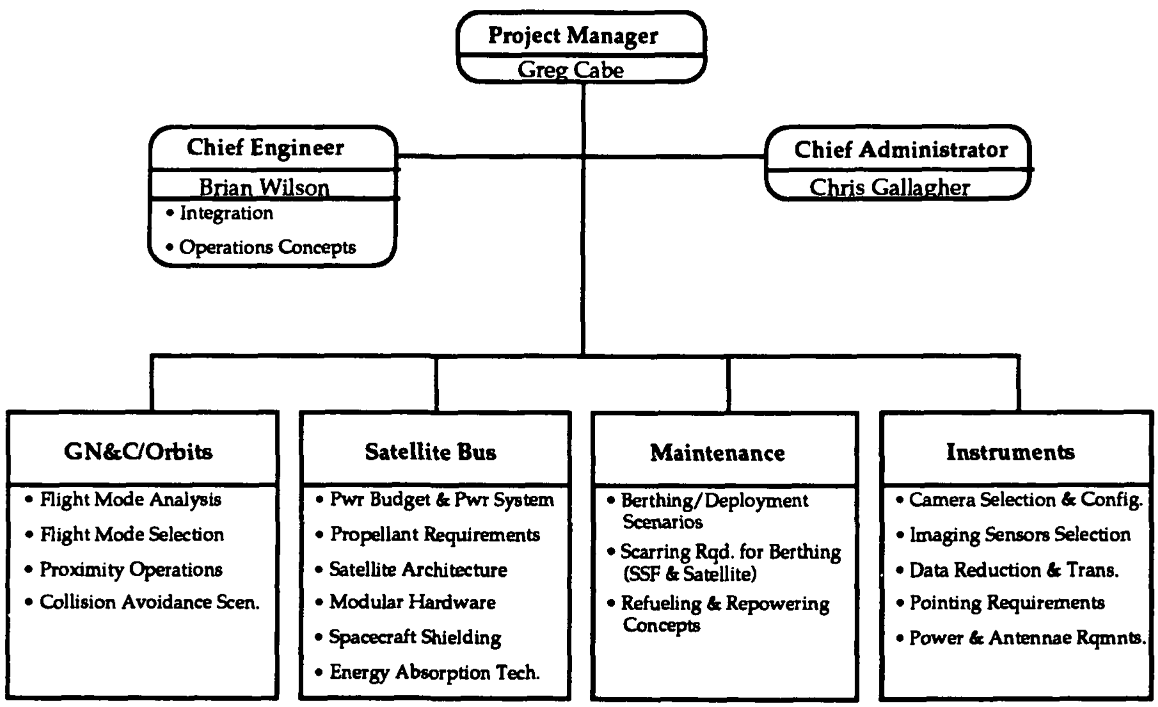


Figure 5.1-1: Project Organizational Chart

The design team was led by a Project Manager, with assistance from a Chief Administrator. The Project Manager managed and monitored the overall design effort and coordinated scheduling with the technical tasks identified by the Chief Engineer and Department Heads.

The design team was divided into four major technical departments, each with a Department Head and engineers. These four groups were responsible for all of the design tasks required to fulfill the contract with the customer. The Chief Engineer coordinated the design efforts of each technical department and integrated the technical design. He also oversaw the technical decisions which were made by the team.

This organizational structure was effective and was used throughout the design phases. This structure permitted rapid communication and coordination of design decisions, and it was easily modified to meet new task requirements when necessary. Engineers that worked in more than one department were assigned to tasks which overlapped from one department to another. This organization aided in the integration effort and also allowed for easy reapportioning of manpower without having to reassign an engineer or create a new department.

To further facilitate communication and problem identification and resolution, a team meeting was held weekly. The meeting was structured around status updates for each of the major tasks within the various departments. In addition, each department met as required to complete its tasks. The Project Manager, Chief Administrator, and Chief Engineer met the day before each team meeting to review project organization, schedules, and design status; to discuss resolution to problems impacting the project; and to plan activities required to meet the project schedules.

5.2 Task Structure

As mentioned previously, the AERCAM design was divided among four departments. This organizational structure was the same as presented in the proposal except that the public relations effort was concluded and the engineering resources dedicated to that department were redistributed to the Instruments and GN&C/Orbits departments.

Figure 5.2-1 shows the most important links of communication and interaction among the various design tasks. At the hub of the design effort, the Chief Engineer integrated and coordinated all of the departmental work and monitored departmental activities as they related to the project schedule. Although the work of the Instruments Package department was only loosely related to the integrated effort, the other departments required highly coordinated work and constant communication, as depicted in the figure. In order to facilitate this interrelation, each of the other three groups, though led by a single department head, contained an engineer who worked in at least one other group. In addition, one engineer was a member of all three of the departments, assisting the Chief Engineer in integration through his distributed work.

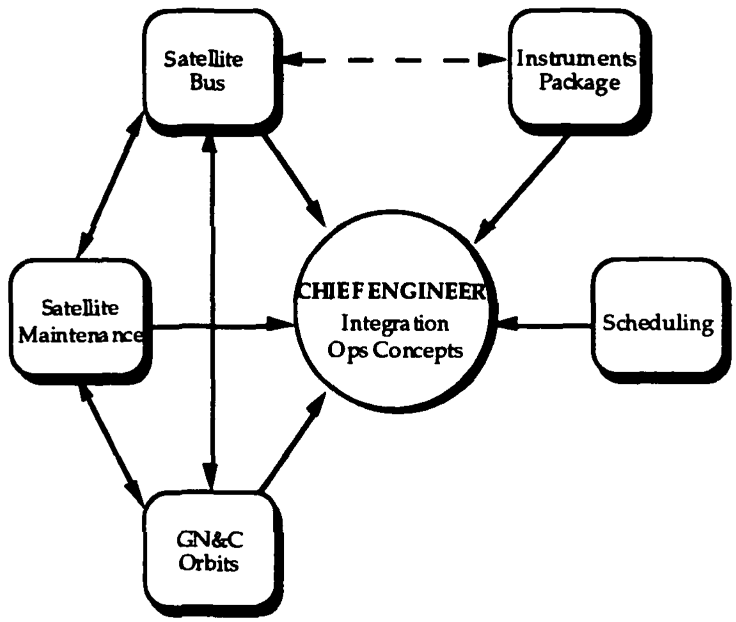


Figure 5.2-1: Task Integration Chart

The major tasks for each of the four departments are shown in Figure 5.1-1. These design activities represent a complete list of the work which Degobah Satellite Systems provided for the customer, based on project analysis after the preliminary design report and discussions with the customer. A number of these activities, including the power and propellant recommendations, are presented in their final form as studies, comparing the relative advantages and disadvantages of several options. Although DSS includes its recommendations for the best options, the final product is formatted as a trade study. Degobah Satellite Systems believes this wide range of information best satisfies the requirements of our customer at NASA-JSC.

5.3 Schedules

The schedule in Figure 5.3-1 shows the milestones for Degobah Satellite Systems. This schedule was used to keep track of tasks and to ensure that DSS remained on schedule. The Chief Administrator was responsible for making sure the schedule was followed and tasks were completed on time. The activities presented on the schedule were completed in conjunction with the Engineering Pert Chart, shown in Figure 5.3-2. The team has just completed its design briefing at NASA, and this report concludes DSS's work on the AERCAM project.

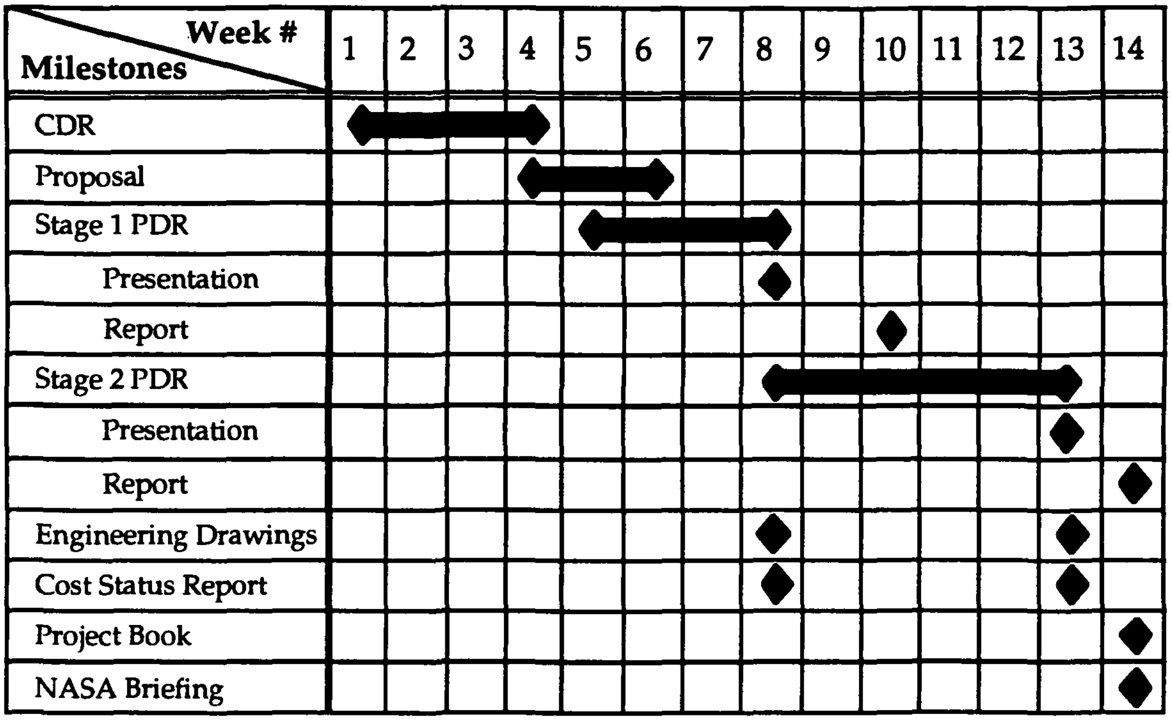


Figure 5.3-1: Milestones Schedule For Degobah Satellite Systems

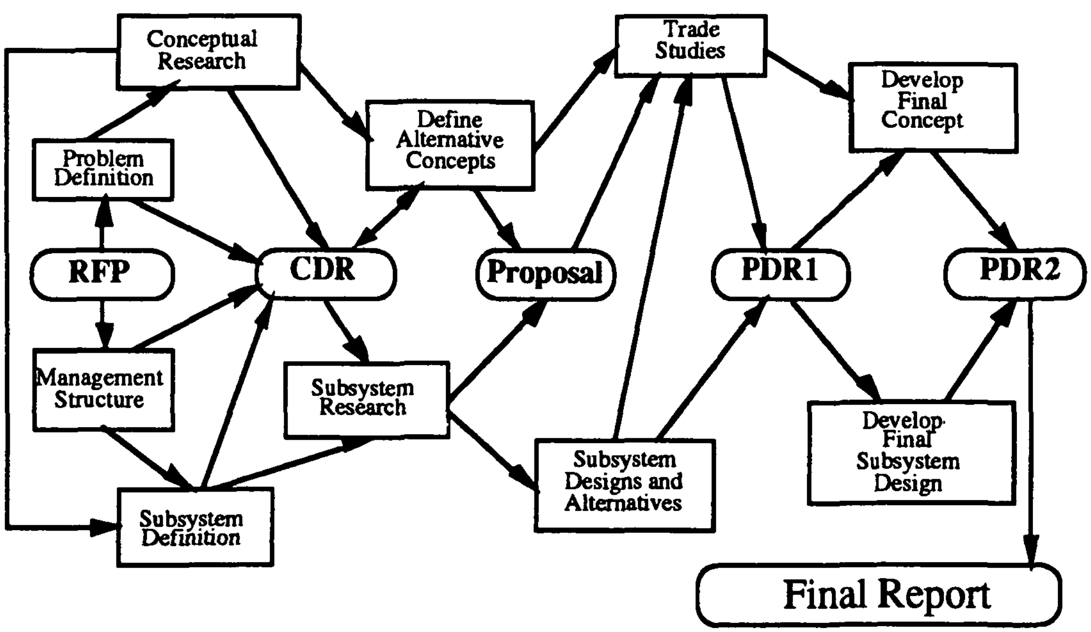


Figure 5.3-2: Engineering PERT Chart

6.0 Cost Analysis

The costs for the AERCAM project, undertaken by DSS, were divided into two parts: personnel costs and material/preparation costs. Tables 6.1-1 and 6.2-1 itemize the projected costs and estimate the overall cost for the project, and Tables 6.1-2 and 6.2-2 show the actual costs for the project. The total proposed and actual costs for the project were $48,580.00 and $47,742.00, respectively. Thus, the project was completed within the proposed budget.

6.1 Personnel Costs

The salaries of each individual were determined from the Request for Proposal packet and previous project reports. The actual manhours were averaged from the personnel timecards, which divided the manhours into various activities. The actual personnel costs for the project were only $185 higher than the proposed costs, but the division of the workload was different than proposed. Table 6.1-1 shows the proposed hourly, weekly, and overall costs for the personnel, as well as the number of personnel on the project. Table 6.1-2 shows the actual costs for the personnel.

Table 6.1-1: Proposed Personnel Costs

Title	Hourly Costs	Number on Staff	Average Hours/Week	Weekly Costs	Total Estimated Personnel Costs
Project Manager	$25.00	1	10	$250.00	$3,500.00
Chief Engineer	$22.00	1	10	$220.00	$3,080.00
Chief Administrator	$22.00	1	8	$132.00	$2,464.00
Department Head	$20.00	5	6	$600.00	$8,400.00
Engineer	$15.00	8	8	$960.00	$13,440.00
Consultants	$75.00	-	8	$600.00	$8,400.00
Subtotal:				$2,762.00	$39,284.00
10% Error				$276.00	$3,928.00
TOTAL PERSONNEL COSTS:				$3,038.00	**$43,212.00**

C-2

Table 6.1-2: Actual Personnel Costs

Title	Hourly Costs	Number on Staff	Average Hours/Week	Weekly Costs	Total Personnel Costs
Project Manager	$25.00	1	6	$150.00	$2,100.00
Chief Engineer	$22.00	1	5	$110.00	$1,540.00
Chief Administrator	$22.00	1	4	$88.00	$1,232.00
Department Head	$20.00	5	5.5	$550.00	$7,700.00
Engineer	$15.00	8	11	$1320.00	$18,480.00
Consultants	$75.00	-	8	$600.00	$8,400.00
Subtotal:				$2,818.00	$39,452.00
10% Error				$282.00	$3,945.00
TOTAL PERSONNEL COSTS:				$3,100.00	**$43,397.00**

Figure 6.1-1 shows a comparison of the cumulative proposed and actual personnel hours for each week. The personnel worked approximately 108 hours more than proposed. Figure 6.1-2 shows the breakdown of the weekly manhours between engineering, administration, and presentation development.

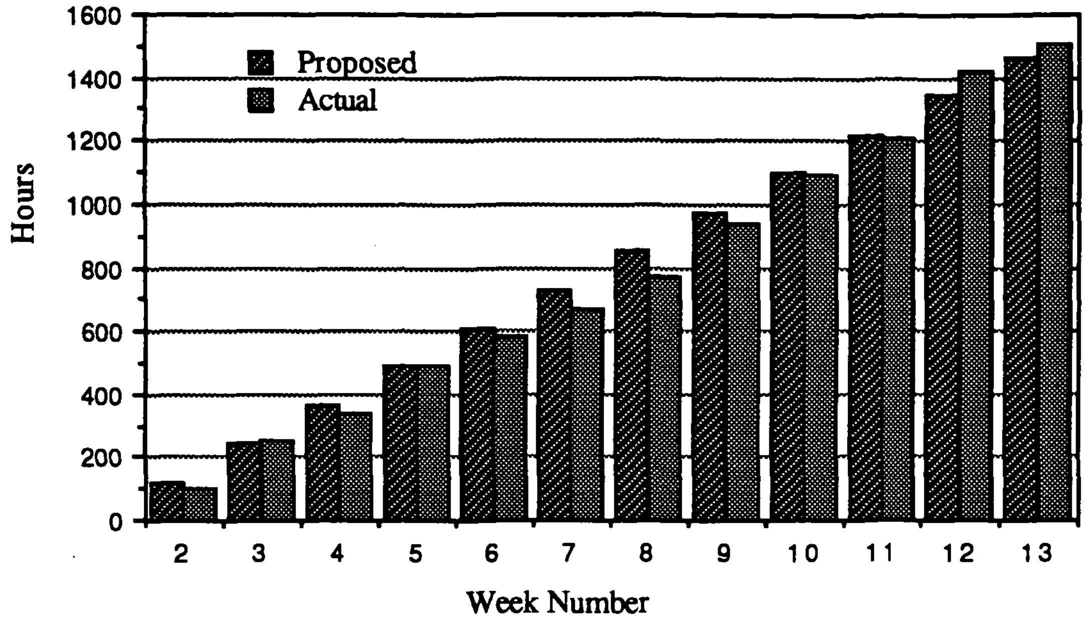


Figure 6.1-1: Cumulative Proposed and Actual Manhours

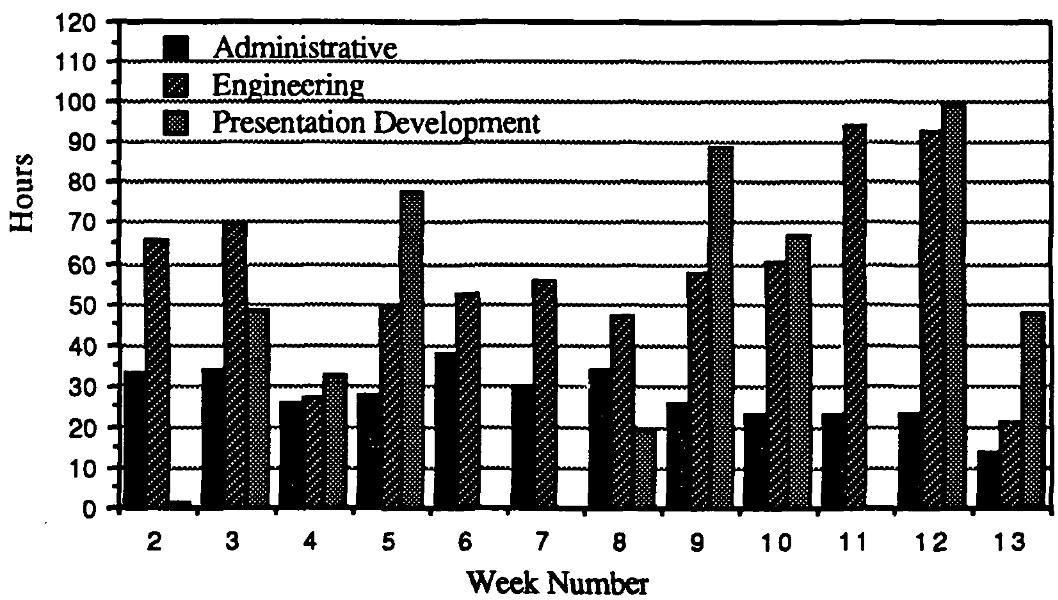


Figure 6.1-2: Weekly Manhours

6.2 Material Costs

The materials used for the project are listed in Tables 6.2-1 and 6.2-2 along with the proposed and actual costs for each material. The proposed costs were based on previous project reports, and the actual costs were determined as the project progressed. The actual costs for the materials were $1,023 less than the proposed costs.

Table 6.2-1: Proposed Material Costs

Material	Cost per Unit	Number of Units Needed	Total Cost
Apple Macintosh Rental	$750.00	4	$3,000.00
UNIX Mainframe Time	$300.00	1	$300.00
Software	$500.00	1	$500.00
Photocopies	$0.08	1000	$80.00
Transparencies	$0.50	200	$100.00
Model	$400.00	1	$400.00
Miscellaneous (travel, etc.)	$500.00	-	$500.00
Subtotal			$4,880.00
10% error			$488.00
TOTAL MATERIAL COST:			**$5,368.00**

Table 6.2-2: Actual Material Costs

Material	Cost per Unit	Number of Units Needed	Total Cost
Apple Macintosh Rental	$750.00	4	$3,000.00
UNIX Mainframe Time	$300.00	0	$0.00
Software	$500.00	1	$500.00
Photocopies	$0.08	1875	$150.00
Transparencies	$0.50	100	$50.00
Model	$50.00	1	$50.00
Miscellaneous (travel, etc.)	$200.00	-	$200.00
Subtotal			$3,950.00
10% error			$395.00
TOTAL MATERIAL COST:			**$4,345.00**

6.3 Total Project Cost

Based on the values listed in Tables 6.1-1 and 6.2-1, the total project cost was proposed to be:

Total Proposed Personnel Cost:	$43,212.00
Total Proposed Material Cost:	$5,368.00
Total Proposed Project Cost:	**$48,580.00**

Based on the values listed in Tables 6.1-2 and 6.2-2, the total project cost was actually:

Total Estimated Personnel Cost:	$43,397.00
Total Estimated Material Cost:	$4,345.00
Total Actual Project Cost:	**$47,742.00**

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Appendix A
Sample Calculations of Camera Features

Six cameras were researched to locate an off the shelf camera that would meet the requirements of the AERCAM project. Calculations which were used to compare each of the cameras are shown in Table A-1. These cameras are listed in Table A-2, along with the features that were calculated for each camera. The decision on what camera to use was based on the camera specifications, as described in Appendix B, and on the data shown in Table A-2.

Table A-2 was developed by using two driving requirements: a 1mm/pixel resolution at a 10 meter distance and a 128 meter field of view at a 500 meter distance. A summary of the equations used to calculate the data in Table A-2 are shown in Table A-1 and explained in the sample calculation that follows. It is important to note that the variables used in Table A-1 are defined in Table A-2.

Table A-1: Summary of Camera Calculations

Part 1:	Part 2:
FOV = RR * EN	$X_{desired}$ = WD = 500 meters
RR = 1mm	$\theta_{new} = \arctan\left(\frac{64}{WD}\right)$
EN = Dependent of camera used.	$FOV_{new} = 2 * 10 * \tan(\theta_{new})$
$MAG = \frac{FOV}{AL}$	WD = 10 meters
AL = EN * (Pixel Spacing)	$MAG_{new} = \frac{FOV_{new}}{AL}$
WD = 10 meters	FL_{new} = Obtained from graph
FL = Obtained from graph	$RR = \frac{FOV}{EN}$
$\theta = \arctan\left(\frac{FOV}{20}\right)$	
$X_{actual} = \frac{64}{\tan(\theta)}$	

Table A-2: Camera Features

Feature \ Camera	Units	EG&G Reticon MC 9000	EG&G Reticon MC 4020	Kodak DCS 200	Cohu 8210	Sony CCD-V801	EG&G Reticon LC 1902
Field of View (FOV)	meters	0.512 / 0.512	2.048 / 2.048	1.5 / 1	0.768 / 0.493	0.64 / 0.64	2.048/0.001
Required Resolution (RR)	mm/pixel	1	1	1	1	1	1
Element Number (EN)	(none)	512 / 512	2048 / 2048	1524 / 1012	768 / 493	640 / 640	2048 / 1
Array Length (AL)	mm	12.8 / 12.8	27.6 / 27.6	14 / 9.3	6.4 / 4.8	12.7 / 12.7	26.62 / 0.013
Magnification (MAG)	(none)	40	75	110	120	50	80
Working Distance (WD)	meters	10	10	10	10	10	10
Focal Length (FL)	mm	~210	135	90	75	~ 180	105
Distance from station to view the entire Station with the calculated FL (Xactual)	meters	2044	624	851	1665	2036	624
Field of View Angle (theta)	degrees	1.5°	5.85°	4.3°	2.2°	1.8°	5.85°

Desired distance from the Station (Xdesired)	meters	500	500	500	500	500	500
Theta new @ 10 meters	degrees	7.3°	7.3°	7.3°	7.3°	7.3°	7.3°
FOV new @ 10 meters	meters	2.56	2.56	2.56	2.56	2.56	2.56
Mag new @ 10 meters	(none)	200	93	183	400	202	96
FL new @ 10 meters	mm	50	100	55	25	50	90
RR @ 10 meters	mm/pixel	5	1.25	1.7	3.33	4	1.25
RR @ 500 meters	mm/pixel	250	62.5	84	166.7	200	62.5

Part 1 of Table A-2 was calculated using the first driving requirement of a 1mm/pixel resolution at a 10 meter distance from the Station. The 10 meter distance was chosen as a reasonable, minimum safe distance to orbit the Station while both providing the necessary resolution and ensuring Station safety. The specifications in part 1 were used to calculate the lens focal length necessary to meet the first requirement. All the values listed represent data calculated for the 10 meter orbit around the Station.

In part 2 of Table A-2, the driving requirement was not the resolution, but rather the field of view. Part 2 uses a field of view of 128 meters (i.e. the entire Station) at a 500 meter distance from the Station to calculate the features for the camera at this distance. The 500 meter distance was chosen because it provided a reasonable resolution, minimum propellant usage, and maximum Station safety. To be able to compare parts 1 and 2 of Table A-2, the data in part 2 was converted to equivalent values at a 10 meter distance. In other words, the angular field of view, θ_{new}, was calculated at 500 meters and used to calculate the other values at a distance from 10 meters. These calculations result in the focal length necessary to meet the field of view requirement at 500 meters. This focal length, combined with the focal length calculated in part 1, yields the necessary zooming requirement of the lens.

Since the EG&G Reticon MC 4020 camera was chosen for the AERCAM design, this camera data will be used and shown in the sample calculation.

Sample Calculation:

Part 1:

For part 1 of Table A-2, the driving requirement was a 1mm/pixel resolution at a distance of 10 meters from the Station. Immediately this identifies two features of the camera: RR = 1mm and WD = 10 meters.

Based on the camera specification sheet, the number of pixels in the array (EN) is 2048 pixels. The first calculation is the field of view (FOV). The field of view is given by

$$FOV = RR * EN = 2.048 \text{ meters.}$$

The center to center spacing of the pixels in the matrix array is given in the camera data sheet to be 0.0135 mm. This value is used to calculate the array length from the equation

$$AL = EN * 0.0135mm = 27.6 \text{ mm}.$$

After acquiring the array length, the image magnification is obtained by the relation

$$MAG = \frac{FOV}{AL} = 75.$$

This magnification is plotted against the working distance, as shown in Figure 1. The magnification of 74 and the working distance of 1000 centimeters were evaluated on the graph and yielded the necessary focal length of 135 mm.

Optical Calculation Worksheet

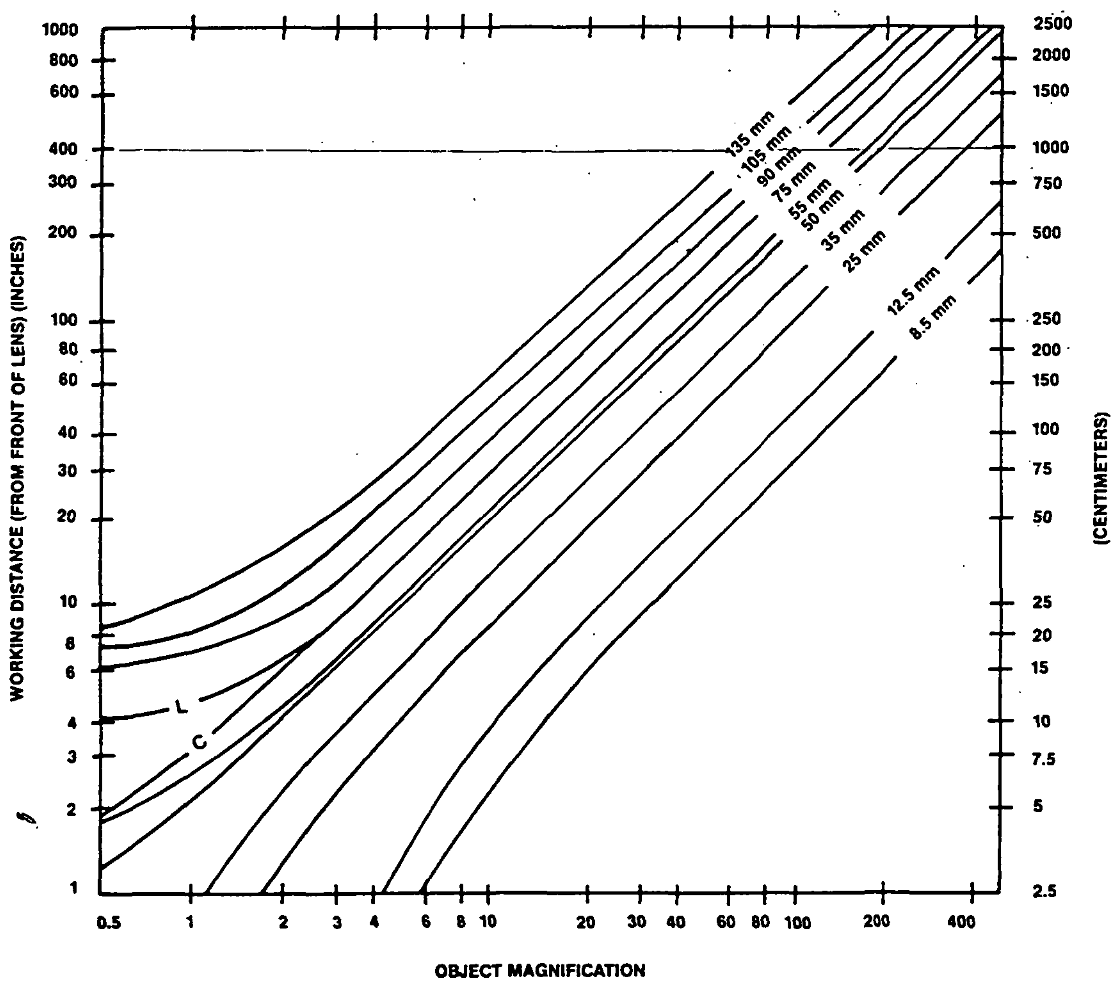

Figure A-1: Relationship between Focal Length, Magnification, and Working Distance

Part 2:

In part 2 of Table A-1, the second requirement is used. The satellite is positioned at a 500 meter distance from the station and has a field of view of 128 meters. This yields the value for two variables: WD = X desired = 500 meters and FOV = 128 meters.

Using simple geometry, as shown in Figure A-2, half of the angular field of view is calculated using the equation

$$\theta_{new} = \tan^{-1}\left(\frac{64}{500}\right) = 7.3°$$

This yields a total angular field of view of 14.6°. This value becomes a characteristic of the lens at the 500 meter distance and hence remains constant in the rest of the calculations.

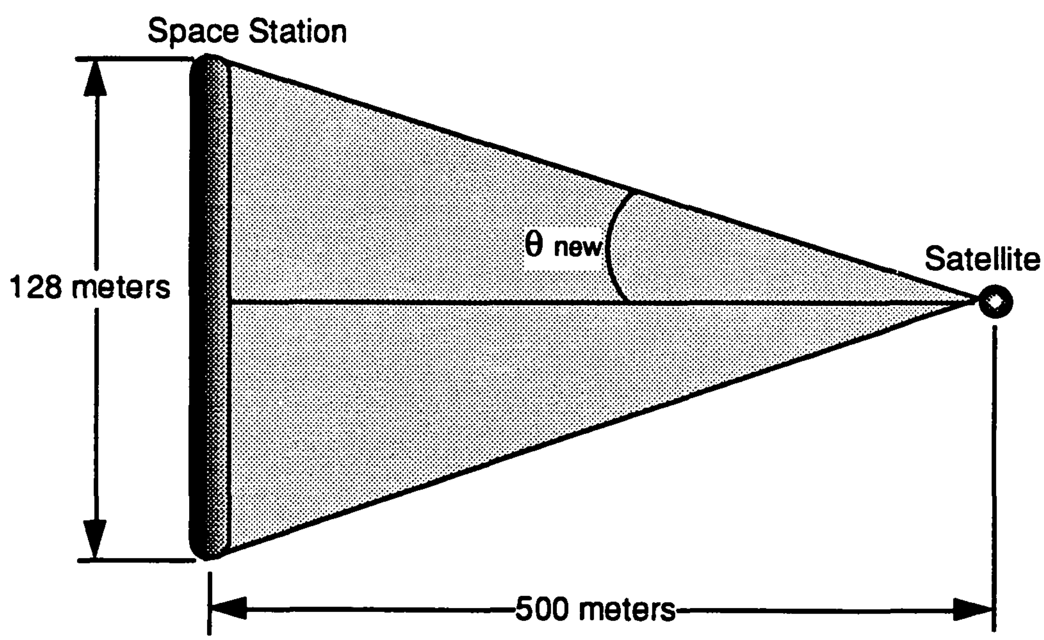

Figure A-2: Angular Field of View Geometry

As the satellite changes orbits, the linear field of view changes since the angular field of view must remain constant. Therefore, if the satellite gets closer to the Station, the linear field of view decreases to only a section of the Station as shown in Figure A-3. The new linear field of view, FOV_{new}, is larger than the initial field of view at 10 meters because θ_{new} is larger. This procedure allows us to calculate lens specifications for the 500 meter distance, while calculating corresponding data at 10 meters to use as a comparison between parts 1 and 2 of Table A-1.

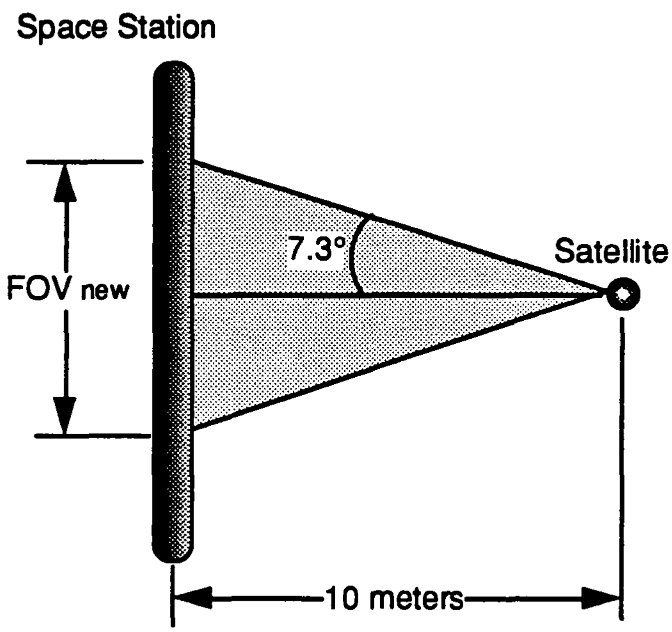

Figure A-3: New Field of View Geometry

After positioning the satellite 10 meters from the Station with the following known values:

$$WD = 10 \text{ meters}$$
$$\theta_{new} = 7.3°$$

the FOV_{new} can then be calculated for the 10 meter orbit. The FOV_{new} is given by the relation

$$FOV_{new} = 2 * WD * \tan(\theta_{new}) = 2.56 \text{ meters.}$$

Using this value with the same array length as calculated earlier, the new image magnification is found by using

$$MAG_{new} = \frac{FOV_{new}}{AL} = 93.$$

This new magnification, along with the working distance are then compared in Figure A-1 to yield a new focal length (FL_{new}) of 100 mm. With this new focal length, the new resolution is calculated using the relation

$$RR = \frac{FOV_{new}}{EN} = 1.25 \text{ mm/pixel.}$$

Using this focal length, and the FOV of 128 meters at 500 meters, the resolution obtained at 500 meters is

$$RR = \frac{FOV}{EN} = 62.5 \text{ mm/pixel.}$$

What these calculations show is that for the MC 4020 camera, the system can resolve 62.5 mm/pixel with the 100 mm focal length at the 500 meter distance. This results in an image magnification of 93. Similarly, the system can resolve 1.25 mm/pixel with the 100 mm focal length or 1mm/pixel with the 135 mm focal length at the 10 meters distance. This results in an image magnification of 75.

Appendix B
Camera Trade Study and Specifications

The camera specifications for the six cameras that were researched are shown in Table B-1. Multiple companies were contacted for information on their digital cameras, and information sheets were obtained from these companies. The specifications for each camera were obtained from the camera information sheet.

The decision of what camera to use onboard the AERCAM was dependent on the data shown in Table B-1 and the information calculated in Appendix A. The EG&G Reticon MC 4020 camera was chosen because it provides high resolution and meets the other requirements for the mission. Also, to provide zooming, the MC 4020 required one of the smallest zooming ranges, and hence required one of the smallest lenses. Furthermore, since the range is small, it minimizes the distance that any moving part has to move. Although we recommend using the MC 4020, the other cameras shown in Table B-1 are viable options for the mission, each offering unique advantages.

The EG&G Reticon MC 9000 camera is small in size and weight, requires low operating power, is very durable under both high shock and vibration conditions. The drawbacks of the camera are that it has a low resolution, it requires a significantly longer lens and zooming range, and it requires a 2 kilometer distance to view the entire Station with the smallest focal length. The camera in general meets most of the initially defined requirements; however it does not meet the second driving requirement of being able to view the entire Station from a 500 meter orbit.

The EG&G Reticon MC 4020 camera is a moderate size, has high resolution, and is durable under both high shock and vibration conditions. The camera also requires a small focal length and zooming range lens (100 mm - 135 mm) to meet the requirements of the mission. The drawbacks of the camera are that is weighs about 2 lbs and requires 8 - 9 Watts of power.

The Kodak DCS 200 camera provides high resolution, includes its own processor, and requires a small focal length lens (50 mm - 90 mm) to meet the mission requirements. The drawbacks of the camera are that it requires 13 watts of operating power, it is large in size, it weighs almost 4 lbs., and it has never been shock tested. Although the camera would provide the required resolution, it would require over 1/3 of the total satellite power to operate.

The Cohu 8210 camera provides medium resolution, requires low operating power, weighs 2 lbs., requires a small focal length lens, and provides continuous video coverage (i.e. > 30 fps). The drawbacks of this camera are that it is not very sturdy to shock or vibration conditions, it is very long, it has never been considered for space application, and its primary usage is as a security camera.

The Sony CCD-V801 camera is primarily a home video camera; however, it does provide television quality resolution and produces greater than 30 fps. The drawbacks of this camera are that it is very large (14.1 inches in length), it weighs almost 7 lbs., it requires a large focal length zooming range (50 mm - 180 mm), and it requires over a 1.5 kilometer distance to view the entire Station using the 180 mm focal length.

Finally, the EG&G Reticon LC 1902 camera provides high resolution, is very small in size, requires low power consumption, and requires a small focal length zooming range (90 mm - 105 mm). The major drawback of this camera is that it is a line scan camera, which adds complexity, whereas the previous cameras were matrix cameras. What this means is that this camera uses one row of pixels to scan the object. A line scan camera actually results in higher resolution images than a matrix camera because it is able to improve the image quality each scan. The camera is capable of scanning 70,000 lines per second, but would require constant moving of the imaging system. This would increase the complexity of the system and affect the stability of the satellite. Furthermore, since the imaging system would constantly be moving, generating a stable, consistent picture at an acceptable rate might be difficult.

Table B-1: Camera Specifications

Characteristic\Camera	EG&G Reticon MC 9000	EG&G Reticon MC 4020	Kodak DCS 200	Cohu 8210	Sony CCD-V801	EG&G Reticon LC 1902
Dimensions (inches)	2.5 x 2.5 x 1.7	3.8 x 3.8 x 6.3	6.7 x 4.5 x 8.2	8.6 x 2.6 x 2.5	4.3 x 5.5 x 14.1	2.5 x 2.5 x 1.9
Pixel Array Size	512 x 512	2048 x 2048	1524 x 1012	768 x 493	640 x 640	1 x 2048
Array Length (mm)	12.8 x 12.8	27.6 x 27.6	14 x 9.3	6.4 x 4.8	N/A	0.013 x 27
Operating Power (Watts)	< 4	< 8	13	5	2.60	4.00
Weight (lbs.)	0.75	2.20	3.70	2.00	6.70	0.75
Speed (Frames/second)	~ 2	0.90	0.33	~ 30	~ 30	~4
Computer Interface	Line Sequential Sends data line by line	SCSI Connector Ports, JPEG compression	SCSI Connector Ports, 50 image storage, 200 MB Harddrive	BNV Video Out R.C. Connector	NTSC Standard	Line transmission Video Formatter Interface and controller card
Ruggedness Shock: Vibration:	100 G's 20 G's	50 G's 150 Hz	N/A N/A	15 G's 15 G's @ 60 Hz	N/A (Low)	300 G's 30 G's
Processing Requirements	Needs entire Processor	Needs entire Processor	Compresses Images, Limited Has a processor	Needs entire Processor	Needs alot of processor implementation	Contains its own processor
Operating Temperature (°C)	0° - 50°	-20° - 50°	4° - 54°	-20° - 50°	Ambient	0 - 55°
Storage Temperature (°C)	-40° - 80°	-40° - 80°	N/A	-30° - 70°	N/A	-40° - 80°

Appendix C
Alternate Imaging System Architecture

New Architectural Design

The baseline design for the imaging system architecture has the entire instruments package housed in a rotating sphere. This design offers a few disadvantages. The mass that has to be panned by the pointing actuators is significant, making pointing difficult, power intensive, and less precise. Removing the imaging sphere from its gimbals maybe a difficult maneuver, especially for a suited astronaut; and cables supplying data and power must be routed through the rotating gimbals, requiring a specially defined interface for the sphere and perhaps reducing the performance of the gimbals. Finally, servicing the imaging system components would require opening and later reassembling the imaging sphere, and the components are not well-positioned for easy removal.

With these concerns in mind, the DSS design team investigated other architectures for the imaging system. One architecture met all of the requirements for the imaging system while solving some of the problems encountered with the baseline design. Therefore, this alternate architecture is presented in its conceptual design in this appendix.

The alternate architecture is shown in a side and a rear view in Figures C-1 and C-2, respectively. As the rear view shows, both of the cameras and the image processing computers are placed side by side and are attached directly to the satellite panel. For cooling purposes, the IR camera and cooling unit have been moved farther away from the other components than in the other configuration. Instead of images being directed immediately through the flash shield and primary lensing systems, as in the baseline configuration, images are bounced off a scanning mirror, through the flash shield, and into the same mirror and lens configuration used in the baseline architecture. The transparent viewing port in the spacecraft panel will still be used, as before.

The lighting system is also shown in the figures, above the mirror. Although the light could be directly attached to the mirror for gimbaling, this is not recommended, since the mirror requires extremely precise pointing and the addition of an off-center mass might impair that pointing or even warp the mirror. Therefore, the light has been placed at the center of the large gimbal for rotation about the vertical axis, and it is equipped with its own gimbal and actuator for rotation about the horizontal axis. Not only does independent pointing reduce interference with the mirror pointing but it allows increased versatility in lighting.

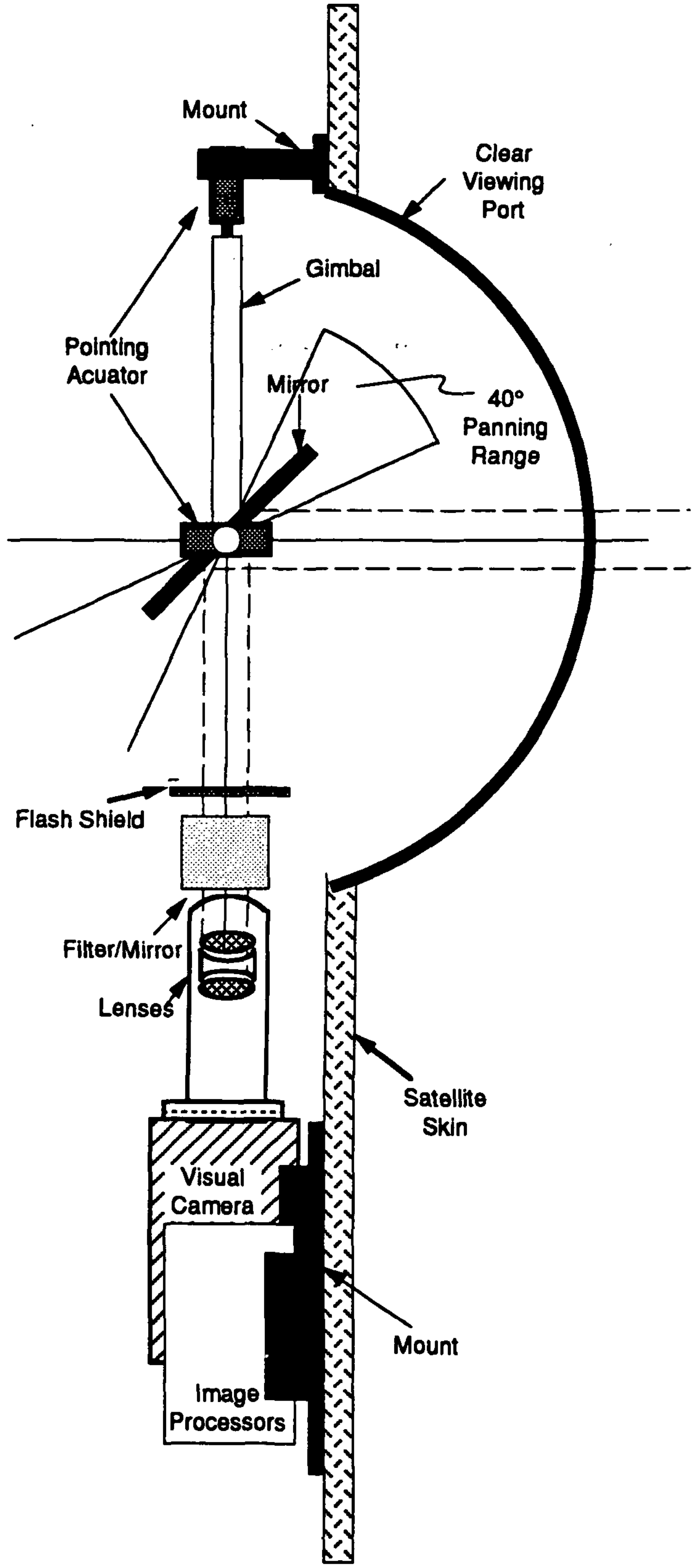

Figure C-1: Side View of the Imaging System

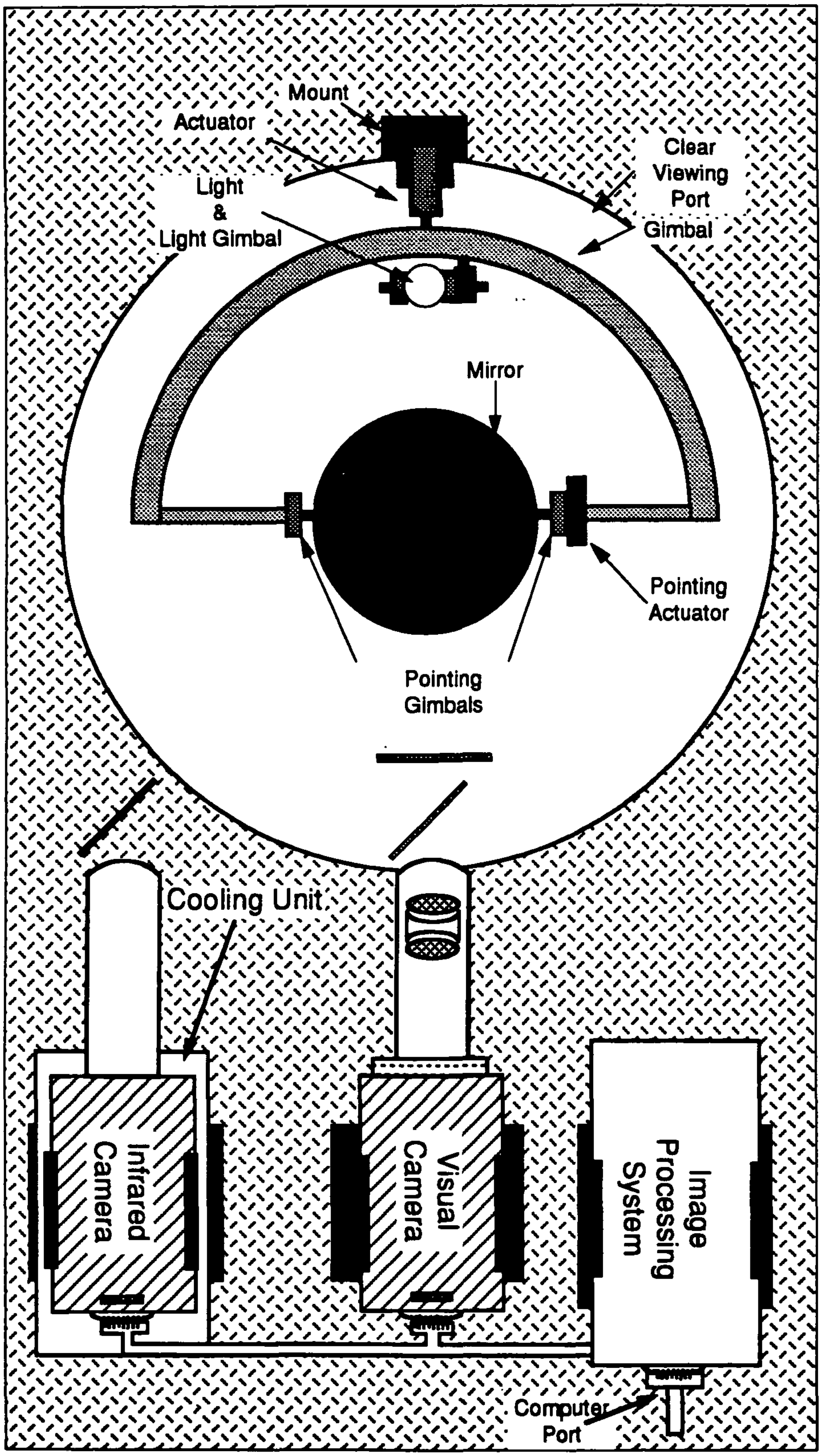

Figure C-2: Rear View of the Imaging System

Mirror Rotation

It is important to note that while this architecture represents a different configuration for the imaging system components, none of those components have changed. The only significant difference is the addition of a mirror. This high precision replaces the gimballed sphere to meet the AERCAM pointing requirements. This is obviously a less massive, less power intensive, and more precise pointing system.

The mirror is nominally positioned at 45° above the horizontal and 45° above the normal to the flash shield. Two gimbals and pointing actuators allow the mirror to rotate at least 20° about both the horizontal and vertical axes. The range of pointing is dependent on the size of the mirror. The farther "up" the mirror rotates, the smaller the profile the camera sees. Therefore, a greater range of pointing would require a larger mirror to compensate for the reduced profile.

Currently, the ±20° of mirror rotation provides an image scan of ±40°. Since the angle of reflection for an image bouncing off a flat mirror is equal to the angle of incidence, as the incident angle changes, the reflected angle changes an equal amount. Therefore, changing the angle of the mirror by 20°, changes the angle of incidence by 20° and the angle of reflection by 20° as well. The result is that the camera is effectively panned by a total of 40°. The range of mirror panning, and associated incident and reflected angles are shown in Figure C-3. The first drawing represents the mirror in its maximum positive rotation. In this position, the camera view an image above the normal to the viewing port. The second drawing represents the mirror in its nominal +45° position. In this position, the camera views an image directly in front of the satellite. The final drawing represents the mirror in its maximum negative rotation. In this position, the camera views an image below the normal to the viewing port.

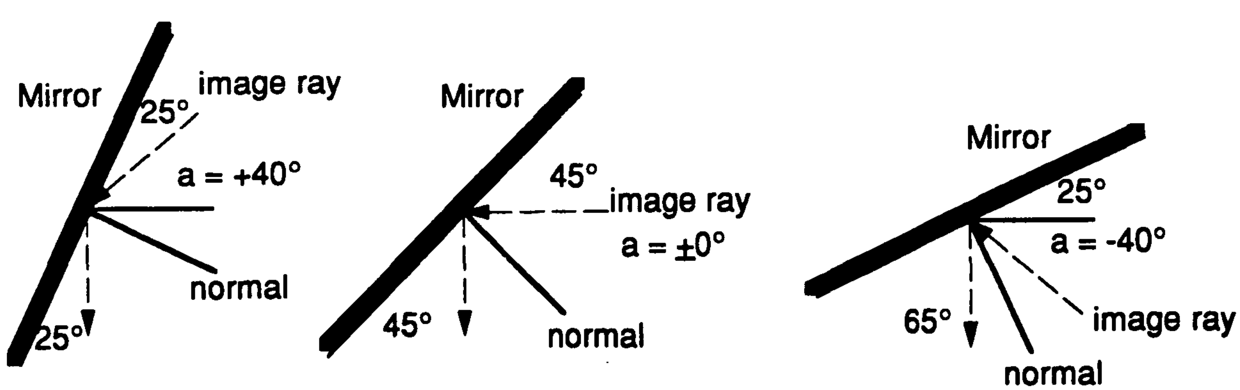

Figure C-3: Mirror Pointing

Advantages and Drawbacks

This architecture offers a number of advantages over the system currently proposed by the DSS design team. Since less mass must be rotated, it requires less power. The system is also smaller and less massive, since the gimbals, actuators and housing are smaller. In conjunction with being smaller, the new system intrudes little more than one half as deep into the satellite as the old system does. This allows for more efficient use of the satellite space.

The new architecture offers easier maintenance. There are no special latches to remove the system from its gimbals, nor is there a sphere which has to be opened to access the components. Although the instruments would still require some type of box housing to provide thermal and electromagnetic shielding and to eliminate contamination of the system by out gasing from other satellite components, this housing can easily be unlatched or unscrewed from the surface of the panel; and the housing will not have to extend any farther into the satellite than the imaging system components already do. By spreading the major components out on the flat surface of the satellite panel, the components can be individually accessed much more easily. The flat architecture also allows for a simpler data and power interface for the instruments package. A single detachable plug can supply power data for the system, as opposed to cables which must run through a moving gimbal and must be detachable at two locations.

The smaller mass of the mirror significantly improves the pointing of the image. It also requires smaller gimbals and actuators. Because of space constraints, the old architecture required the lens to be slightly off-center of the spherical viewing port. The mirror, however, is placed directly on center, thus eliminating any possible distortion.

The new architecture does have a few drawbacks that are worth mentioning, however. It relies on an additional component - a precision mirror that must not be subjected to warping. This mirror must also have extremely high reflectivity over a spectral range from 0.45 μm to 15 μm. If the mirror were to move off center by settling or warping of the gimbals or the mirror itself, this could adversely affect the images produced by the cameras. Any particulate contamination of the image path or the mirror would also degrade the image, but this risk is present in the old architecture as well.

Although this architecture is only represented in its conceptual form here, it offers a number of advantages that recommend it over the old system. Only further study can verify its feasibility and its relative advantages and disadvantages, however.

Appendix D
Computer System Throughput

Table D-1 represents a preliminary estimate of the throughput required for the computer system controlling the AERCAM satellite. It was developed using information in the computer systems chapter of Space Mission Analysis and Design, by J. R. Wertz. The estimates are based on a 1750A-class Instruction Set Architecture using 16-bit words. A typical number of 200 tasks per second and 1000 I/O words per second is assumed for the operating system. "K" represents 1000, and KIPS represents 1000 instructions per second. It should be noted that these estimates are for a standard computer system not employing artificial intelligence.

Table D-1: Computer System Throughput

Component	Frequency Hz	Code Kwords	Data Kwords	Required Throughput (KIPS)
Application Functions				
Thruster Control	8	2.4	1.6	4.8
Reaction Wheel Control	8	4.0	1.2	20.0
Rate Gyros	20	1.6	1.0	18.0
Suns Sensor	4	2.0	0.4	4.0
Kinematic Integration	20	4.0	0.4	30.0
Error Determination	20	2.0	0.2	24.0
Ephemeris Propagation	2	4.0	0.6	4.0
Orbit Propagation	2	26.0	8.0	40.0
Complex Autonomy	20	30.0	20.0	40.0
Fault Detection	10	8.0	2.0	30.0
Fault Correction	10	4.0	20.0	10.0
Power Management	1.0	1.2	0.5	5.0
Thermal Control	0.2	1.6	3.0	6.0
Imaging System Control	10	5.0	1.5	20.0
Collision Avoidance Sensor Control and Integration	20	6.6	3.0	60.0
Operating System				
Local Executive		3.5	2.0	60.0
Runtime Kernal		8.0	4.0	N/A
I/O handlers		2.0	0.7	50.0
Built-in-Test and Diagnostics	1 Hz	7.0	4.0	5.0
Utilities		1.2	0.2	N/A
Total Size and Throughput Estimate		**124.1**	**74.3**	**430.8**

Appendix E
Data Reduction Techniques

This appendix gives detailed definitions of each of the data reduction techniques discussed in section 3.2 of the report.

Frame Grabbing means that the computer system on the satellite selects only a portion of the entire image to transmit. This portion of the frame would be selected by the user or by the computer system based on some predefined criteria. Therefore, Frame Grabbing eliminates some of the image, but it does not reduce the image resolution. Since spacecraft pointing will not be perfect, Frame Grabbing may employ image recognition techniques to ensure that the correct image is continuously transmitted to the user. Frame Grabbing would typically be employed when the user needs to actively view a particular object in great detail to the exclusion of other objects.

Pixel Averaging is another method of image reduction. This method is accomplished simply by averaging the values of groups of pixels. Hence, the quality of the image is reduced. To prevent some loss of resolution, logic could be employed to identify similar groups of pixels to select for averaging, while also identifying "boundaries" between pixels of significantly different values, over which data would not be averaged. In another method of preventing resolution loss, pixels of similar intensity and values could be identified, their values averaged, and then this "representative pixel" could be sent as data along with the location within the pixel array of each of the "represented pixels." These techniques of Pixel Averaging would typically be employed when the user is either not actively using the system, or when the users wishes to see a large frame without great detail.

Intensity Filtering is another method of image reduction. The computer employs this type of data reduction by sending only the values of pixels which have values equal to or greater than a specified intensity. Other pixels would remain black or some other user-defined color. This type of reduction might be employed as alternative to Pixel Averaging, especially when the Space Station is being viewed from a great distance.

An advanced and experimental form of data reduction is based on image recognition and requires complex imaging processing before the data may be transmitted by the satellite. For this method, referred to by DSS as **Pixel Memorization,** the computer maintains a stored representation of a previous image which is periodically updated.

Overlapped frames (from the memorized and current images) are compared, and only the pixels that have changed by some specified amount are transmitted to update the image at the user station. This method would be best suited to use when the AERCAM remains at a fixed distance from its target and when the user is not actively viewing a specified object. An adaptation of Pixel Memorization could also be used to assist in identifying anomalies (especially thermal anomalies) on the Station.

Reduction in the number of pixels scanned by the CCD matrix would reduce data without requiring computer processing of the image. If a variable speed scanner can be used for the camera, then this method could also be used to increase the scan rate of the camera. Therefore, more images could be produced each second, without exceeding the capabilities of the scanner and image processor, by reducing the number of pixels scanned.

Reducing the scan rate, that is simply reducing the number of frames per second that are transmitted, is the final method of data reduction. This less frequent update of the user station image would most likely be employed when the user is not actively using the AERCAM imaging platform.

Appendix F
Communications Link Budget

The link budget calculated for the AERCAM communications system was developed using formulas from the communications chapter in Space Mission Analysis and Design, by J. R. Wertz. Table F-1, below, shows each of the parameters used in calculating the AERCAM link budget. Specifically, the table shows the margin for a number of different power inputs. It is evident that 0.5 watts is more than sufficient to support the communications system. This table was developed with the satellite housing the transmitting antenna and the station housing the receiving antenna. The tables which follow the primary link budget were used in determining the optimal specifications for the communications system.

The following are notes and assumptions which clarify a number of the inputs used in the link budget:

- The frequency was specified according to the old SSF space-to-space communications protocol.
- The power range of up to 2 watts should not significantly affect the instruments package, since the antennas were allocated 10 watts according to original estimates.
- The transmitter line loss is actually 2 times the typical value. ATHENA™, a spacecraft similar to AERCAM, actually has line losses of 0.2 dB. The maximum loss possible for AERCAM, using waveguide transmission, would be less than 5 dB.
- The transmitting antenna is one wavelength in diameter.
- The system noise temperature is an extremely conservative estimate, considering that there is no atmosphere or rain and very little loss so close to the Station.
- The antenna efficiency is a typical value.
- The pointing offset is also a typical value.

Table F-1: AERCAM Primary Link Budget

Item					Units
Transmit Parameters					
Frequency	14.00	14.00	14.00	14.00	GHz
Transmitter Power	2.00	1.50	1.00	0.50	Watts
Transmitter Power	3.01	1.76	0.00	-3.01	dBW
Transmitter Line Loss	-1.00	-1.00	-1.00	-1.00	dB
Transmit Antenna Diameter	0.02	0.02	0.02	0.02	m
Transmit Antenna Beamwidth	75.00	75.00	75.00	75.00	deg
Peak Transmit Antenna Gain	6.80	6.80	6.80	6.80	dB
Transmit Antenna Pointing offset	2.00	2.00	2.00	2.00	deg
Transmit Antenna Pointing Loss	-0.01	-0.01	-0.01	-0.01	dB
Transmit Antenna Gain	6.79	6.79	6.79	6.79	dB
Equiv. Isotropic Radiated Power	8.80	7.55	5.79	2.78	dBW
Propagation Parameters					
Propagation Path Length	1.00	1.00	1.00	1.00	km
Space Loss	-115.36	-115.36	-115.36	-115.36	dB
Propagation & Polarization Loss	0.00	0.00	0.00	0.00	dB
System Noise Temperature	100.00	100.00	100.00	100.00	K
Receiving Station Parameters					
Receive Antenna Diameter	0.02	0.02	0.02	0.02	m
Receive Antenna Efficiency	0.55	0.55	0.55	0.55	-
Peak Receive Antenna Gain	6.75	6.75	6.75	6.75	dB
Receive Antenna Beamwidth	75.00	75.00	75.00	75.00	deg
Receive Antenna Pointing Error	2.00	2.00	2.00	2.00	deg
Receive Antenna Pointing Loss	-0.24	-0.24	-0.24	-0.24	dB
Receive Antenna Gain	6.50	6.50	6.50	6.50	dB
Final Calculations					
Data Rate	64	64	64	64	Mbps
Eb/No (1)	30.48	29.23	27.47	24.46	dB
Carrier-to-Noise Density Ratio	122.10	120.85	119.09	116.08	dB-Hz
Bit Error Rate	0.00	0.00	0.00	0.00	-
Required Eb/No (2)	4.40	4.40	4.40	4.40	dB-Hz
Implementation Loss (3)	-2.00	-2.00	-2.00	-2.00	dB
Margin	24.08	22.83	21.07	18.06	dB

Item	Symbol	Source	Diameters		Power Input (1 lambda)			
			1 lambda	1/2 Lambda				
Transmit Parameters								
Frequency	f	Input	14.00	14.00	14.00	14.00	14.00	14.00
Transmitter Power	P	Input	2.00	2.00	2.00	1.50	1.00	0.50
Transmitter Power	P	10 log(P)	3.01	3.01	3.01	1.76	0.00	-3.01
Transmitter Line Loss	Ll	Input	-1.00	-1.00	-1.00	-1.00	-1.00	-1.00
Transmit Antenna Diameter	Dt	Eqn 13-17	0.02	0.01	0.02	0.02	0.02	0.02
Transmit Antenna Beamwidth	0t	Eqn 13-17	75.00	150.00	75.00	75.00	75.00	75.00
Peak Transmit Antenna Gain	Gtp	Eqn 13-18	6.80	0.78	6.80	6.80	6.80	6.80
Transmit Antenna Pointing offset	et	Input	2.00	2.00	2.00	2.00	2.00	2.00
Transmit Antenna Pointing Loss	Lpt	Eqn 13-19	-0.01	0.00	-0.01	-0.01	-0.01	-0.01
Transmit Antenna Gain	Gt	Gpt+Lpt	6.79	0.78	6.79	6.79	6.79	6.79
Equiv. Isotropc Radiated Power	EIRP	P+Ll+Gt	8.80	2.79	8.80	7.55	5.79	2.78
Propogation Parameters								
Propogation Path Length	S	Input	1.00	1.00	1.00	1.00	1.00	1.00
Space Loss	Ls	Eqn 13-21	-115.36	-115.36	-115.36	-115.36	-115.36	-115.36
Propagation & Polarization Loss	La	Figure 13-10	0.00	0.00	0.00	0.00	0.00	0.00
System Noise Temperature	Ts	Table 13-9	100.00	100.00	100.00	100.00	100.00	100.00
Receiving Station Parameters								
Receive Antenna Diameter	Dr	Input	0.02	0.01	0.02	0.02	0.02	0.02
Receive Antenna Efficiency	n	Input	0.55	0.55	0.55	0.55	0.55	0.55
Peak Receive Antenna Gain	Grp	Eqn 13-16	6.75	0.73	6.75	6.75	6.75	6.75
Receive Antenna Beamwidth	0r	Eqn 13-17	75.00	150.00	75.00	75.00	75.00	75.00
Receive Antenna Pointing Error	er	Input	2.00	2.00	2.00	2.00	2.00	2.00
Receive Antenna Pointing Loss	Lpr	Eqn 13-19	-0.24	-0.24	-0.24	-0.24	-0.24	-0.24
Receive Antenna Gain	Gr	Grp+Lpr	6.50	0.48	6.50	6.50	6.50	6.50
Final Calculations								
Data Rate	R	Input	####	####	####	####	####	####
Eb/No (1)	Eb/No	Eqn 13-11	30.48	18.44	30.48	29.23	27.47	24.46
Carrier-to-Noise Densiy Ratio	C/No	Eqn 13-13	122.10	116.03	122.10	120.85	119.09	116.08
Bit Error Rate	BER	Input	0.00	0.00	0.00	0.00	0.00	0.00
Required Eb/No (2)	Req Eb/No	Figure 13-9	4.40	4.40	4.40	4.40	4.40	4.40
Implementation Loss (3)	-	Estimate	-2.00	-2.00	-2.00	-2.00	-2.00	-2.00
Margin	-	(1)-(2)+(3)	24.08	12.04	24.08	22.83	21.07	18.06

11/17/92

Item	Symbol	Source	Comparison of Different Frequencies							Units
Transmit Parameters										
Frequency	f	Input	**1.00**	**2.00**	**5.00**	**7.00**	**10.00**	**13.00**	**15.03**	GHz
Transmitter Power	P	Input	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	Watts
Transmitter Power	P	10 log(P)	10.00	10.00	10.00	10.00	10.00	10.00	10.00	dBW
Transmitter Line Loss	Ll	Input	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	dB
Transmit Antenna Diameter	Dt	Eqn 13-17	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	m
Transmit Antenna Beamwidth	Θt	Eqn 13-17	300.00	150.00	60.00	42.86	30.00	23.08	19.96	deg
Peak Transmit Antenna Gain	Gtp	Eqn 13-18	-5.24	0.78	8.74	11.66	14.76	17.04	18.30	dB
Transmit Antenna Pointing offset	et	Input	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	deg
Transmit Antenna Pointing Loss	Lpt	Eqn 13-19	-0.01	-0.05	-0.33	-0.65	-1.33	-2.25	-3.01	dB
Transmit Antenna Gain	Gt	Gpt+Lpt	-5.26	0.72	8.40	11.01	13.42	14.78	15.28	dB
Equiv. Isotropc Radiated Power	EIRP	P+Ll+Gt	3.74	9.72	17.40	20.01	22.42	23.78	24.28	dBW
Propogation Parameters										
Propogation Path Length	S	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	km
Space Loss	Ls	Eqn 13-21	-92.44	-98.46	-106.42	-109.34	-112.44	-114.72	-115.98	dB
Propagation & Polarization Loss	La	Figure 13-10	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dB
System Noise Temperature	Ts	Table 13-9	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	K
Receiving Station Parameters										
Receive Antenna Diameter	Dr	Input	**1.50**	**1.50**	**1.50**	**1.50**	**1.50**	**1.50**	**1.50**	m
Receive Antenna Efficiency	n	Input	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	-
Peak Receive Antenna Gain	Grp	Eqn 13-16	21.33	27.35	35.30	38.23	41.33	43.60	44.86	dB
Receive Antenna Beamwidth	Θr	Eqn 13-17	14.00	7.00	2.80	2.00	1.40	1.08	0.93	deg
Receive Antenna Pointing Error	er	Input	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	deg
Receive Antenna Pointing Loss	Lpr	Eqn 13-19	-0.48	-0.12	-0.02	-0.01	0.00	0.00	0.00	dB
Receive Antenna Gain	Gr	Grp+Lpr	20.85	27.23	35.29	38.22	41.32	43.60	44.86	dB
Final Calculations										
Data Rate	R	Input	**#####**	**#####**	**#####**	**#####**	**#####**	**#####**	**#####**	bps
Eb/No (1)	Eb/No	Eqn 13-11	62.69	69.03	76.81	79.42	81.84	83.20	83.71	dB
Carrier-to-Noise Densiy Ratio	C/No	Eqn 13-13	140.11	140.14	139.94	139.65	139.00	138.10	137.35	dB-Hz
Bit Error Rate	BER	Input	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	-
Required Eb/No (2)	Req Eb/No	Figure 13-9	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	dB-Hz
Implementation Loss (3)	-	Estimate	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	dB
Margin	-	(1)-(2)+(3)	56.29	62.63	70.41	73.02	75.44	76.80	77.31	dB

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Item	Symbol	Source	Comparison of Different Receiving Antenna Diameter							Units
Transmit Parameters										
Frequency	f	Input	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	GHz
Transmitter Power	P	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	Watts
Transmitter Power	P	10 log(P)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dBW
Transmitter Line Loss	Ll	Input	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	dB
Transmit Antenna Diameter	Dt	Eqn 13-17	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	m
Transmit Antenna Beamwidth	Ot	Eqn 13-17	150.00	150.00	150.00	150.00	150.00	150.00	150.00	deg
Peak Transmit Antenna Gain	Gtp	Eqn 13-18	0.78	0.78	0.78	0.78	0.78	0.78	0.78	dB
Transmit Antenna Pointing offset	et	Input	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	deg
Transmit Antenna Pointing Loss	Lpt	Eqn 13-19	-0.05	-0.05	-0.05	-0.05	-0.05	-0.05	-0.05	dB
Transmit Antenna Gain	Gt	Gpt+Lpt	0.72	0.72	0.72	0.72	0.72	0.72	0.72	dB
Equiv. Isotropc Radiated Power	EIRP	P+Ll+Gt	-0.28	-0.28	-0.28	-0.28	-0.28	-0.28	-0.28	dBW
Propogation Parameters										
Propogation Path Length	S	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	km
Space Loss	Ls	Eqn 13-21	-98.46	-98.46	-98.46	-98.46	-98.46	-98.46	-98.46	dB
Propagation & Polarization Loss	La	Figure 13-10	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dB
System Noise Temperature	Ts	Table 13-9	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	K
Receiving Station Parameters										
Receive Antenna Diameter	Dr	Input	**0.25**	**0.50**	**1.00**	**1.50**	**2.00**	**2.50**	**3.00**	m
Receive Antenna Efficiency	n	Input	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	-
Peak Receive Antenna Gain	Grp	Eqn 13-16	11.78	17.80	23.82	27.35	29.84	31.78	33.37	dB
Receive Antenna Beamwidth	Or	Eqn 13-17	42.00	21.00	10.50	7.00	5.25	4.20	3.50	deg
Receive Antenna Pointing Error	er	Input	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	deg
Receive Antenna Pointing Loss	Lpr	Eqn 13-19	-0.12	-0.12	-0.12	-0.12	-0.12	-0.12	-0.12	dB
Receive Antenna Gain	Gr	Grp+Lpr	11.66	17.68	23.70	27.23	29.72	31.66	33.25	dB
Final Calculations										
Data Rate	R	Input	*****	*****	*****	*****	*****	*****	*****	bps
Eb/No (1)	Eb/No	Eqn 13-11	43.47	49.49	55.51	59.03	61.53	63.47	65.05	dB
Carrier-to-Noise Densiy Ratio	C/No	Eqn 13-13	129.98	130.04	130.10	130.14	130.16	130.18	130.20	dB-Hz
Bit Error Rate	BER	Input	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	-
Required Eb/No (2)	Req Eb/No	Figure 13-9	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	dB-Hz
Implementation Loss (3)	-	Estimate	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	dB
Margin	- -	(1)-(2)+(3)	37.07	43.09	49.11	52.63	55.13	57.07	58.65	dB

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Item	Symbol	Source	Comparison of Different Transmitting Antenna Diamet							Units
Transmit Parameters										
Frequency	f	Input	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	GHz
Transmitter Power	P	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	Watts
Transmitter Power	P	10 log(P)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dBW
Transmitter Line Loss	Ll	Input	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	dB
Transmit Antenna Diameter	Dt	Eqn 13-17	**0.05**	**0.10**	**0.50**	**1.00**	**1.50**	**2.00**	**0.07**	m
Transmit Antenna Beamwidth	Θt	Eqn 13-17	210.00	105.00	21.00	10.50	7.00	5.25	150.00	deg
Peak Transmit Antenna Gain	Gtp	Eqn 13-18	-2.14	3.88	17.86	23.88	27.40	29.90	0.78	dB
Transmit Antenna Pointing offset	et	Input	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	deg
Transmit Antenna Pointing Loss	Lpt	Eqn 13-19	-0.03	-0.11	-2.72	-10.88	-24.49	-43.54	-0.05	dB
Transmit Antenna Gain	Gt	Gpt+Lpt	-2.17	3.77	15.13	12.99	2.91	-13.64	0.72	dB
Equiv. Isotropc Radiated Power	EIRP	P+Ll+Gt	-3.17	2.77	14.13	11.99	1.91	-14.64	-0.28	dBW
Propogation Parameters										
Propogation Path Length	S	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	km
Space Loss	Ls	Eqn 13-21	-98.46	-98.46	-98.46	-98.46	-98.46	-98.46	-98.46	dB
Propagation & Polarization Loss	La	Figure 13-10	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dB
System Noise Temperature	Ts	Table 13-9	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	K
Receiving Station Parameters										
Receive Antenna Diameter	Dr	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	m
Receive Antenna Efficiency	n	Input	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	-
Peak Receive Antenna Gain	Grp	Eqn 13-16	23.82	23.82	23.82	23.82	23.82	23.82	23.82	dB
Receive Antenna Beamwidth	Θr	Eqn 13-17	10.50	10.50	10.50	10.50	10.50	10.50	10.50	deg
Receive Antenna Pointing Error	er	Input	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	deg
Receive Antenna Pointing Loss	Lpr	Eqn 13-19	-0.12	-0.12	-0.12	-0.12	-0.12	-0.12	-0.12	dB
Receive Antenna Gain	Gr	Grp+Lpr	23.70	23.70	23.70	23.70	23.70	23.70	23.70	dB
Final Calculations										
Data Rate	R	Input	#####	#####	#####	#####	#####	#####	#####	bps
Eb/No (1)	Eb/No	Eqn 13-11	52.61	58.55	69.92	67.77	57.69	41.14	55.51	dB
Carrier-to-Noise Densiy Ratio	C/No	Eqn 13-13	127.20	133.14	144.51	142.37	132.28	115.74	130.10	dB-Hz
Bit Error Rate	BER	Input	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	-
Required Eb/No (2)	Req Eb/No	Figure 13-9	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	dB-Hz
Implementation Loss (3)	-	Estimate	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	dB
Margin	-	(1)-(2)+(3)	46.21	52.15	63.52	61.37	51.29	34.74	49.11	dB

Item	Symbol	Source	Comparison of Different Transmitting Power							Units
Transmit Parameters										
Frequency	f	Input	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	GHz
Transmitter Power	P	Input	**0.50**	**0.75**	**1.00**	**2.00**	**5.00**	**7.00**	**10.00**	Watts
Transmitter Power	P	10 log(P)	-3.01	-1.25	0.00	3.01	6.99	8.45	10.00	dBW
Transmitter Line Loss	Ll	Input	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	dB
Transmit Antenna Diameter	Dt	Eqn 13-17	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	**0.07**	m
Transmit Antenna Beamwidth	Ot	Eqn 13-17	150.00	150.00	150.00	150.00	150.00	150.00	150.00	deg
Peak Transmit Antenna Gain	Gtp	Eqn 13-18	0.78	0.78	0.78	0.78	0.78	0.78	0.78	dB
Transmit Antenna Pointing offset	et	Input	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	**10.00**	deg
Transmit Antenna Pointing Loss	Lpt	Eqn 13-19	-0.05	-0.05	-0.05	-0.05	-0.05	-0.05	-0.05	dB
Transmit Antenna Gain	Gt	Gpt+Lpt	0.72	0.72	0.72	0.72	0.72	0.72	0.72	dB
Equiv. Isotropc Radiated Power	EIRP	P+Ll+Gt	-3.29	-1.52	-0.28	2.74	6.71	8.18	9.72	dBW
Propogation Parameters										
Propogation Path Length	S	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	km
Space Loss	Ls	Eqn 13-21	-98.46	-98.46	-98.46	-98.46	-98.46	-98.46	-98.46	dB
Propogation & Polarization Loss	La	Figure 13-10	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dB
System Noise Temperature	Ts	Table 13-9	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	K
Receiving Station Parameters										
Receive Antenna Diameter	Dr	Input	**1.50**	**1.50**	**1.50**	**1.50**	**1.50**	**1.50**	**1.50**	m
Receive Antenna Efficiency	n	Input	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	-
Peak Receive Antenna Gain	Grp	Eqn 13-16	27.35	27.35	27.35	27.35	27.35	27.35	27.35	dB
Receive Antenna Beamwidth	Or	Eqn 13-17	7.00	7.00	7.00	7.00	7.00	7.00	7.00	deg
Receive Antenna Pointing Error	er	Input	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	**0.20**	deg
Receive Antenna Pointing Loss	Lpr	Eqn 13-19	-0.12	-0.12	-0.12	-0.12	-0.12	-0.12	-0.12	dB
Receive Antenna Gain	Gr	Grp+Lpr	27.23	27.23	27.23	27.23	27.23	27.23	27.23	dB
Final Calculations										
Data Rate	R	Input	#####	#####	#####	#####	#####	#####	#####	bps
Eb/No (1)	Eb/No	Eqn 13-11	56.02	57.78	59.03	62.04	66.02	67.48	69.03	dB
Carrier-to-Noise Densiy Ratio	C/No	Eqn 13-13	127.13	128.89	130.14	133.15	137.13	138.59	140.14	dB-Hz
Bit Error Rate	BER	Input	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	-
Required Eb/No (2)	Req Eb/No	Figure 13-9	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	dB-Hz
Implementation Loss (3)	-	Estimate	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	dB
Margin	-	(1)-(2)+(3)	49.62	51.38	52.63	55.64	59.62	61.08	62.63	dB

2 Watts is probably ok

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Item	Symbol	Source	Comparison of	Different	Transmitting	Antenna	Diamet			Units
Transmit Parameters										
Frequency	f	Input	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	GHz
Transmitter Power	P	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	Watts
Transmitter Power	P	10 log(P)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dBW
Transmitter Line Loss	L1	Input	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	**-1.00**	dB
Transmit Antenna Diameter	Dt	Eqn 13-17	**0.10**	**0.10**	**0.10**	**0.10**	**0.10**	**0.10**	**0.10**	m
Transmit Antenna Beamwidth	Θt	Eqn 13-17	105.00	105.00	105.00	105.00	105.00	105.00	105.00	deg
Peak Transmit Antenna Gain	Gtp	Eqn 13-18	3.88	3.88	3.88	3.88	3.88	3.88	3.88	dB
Transmit Antenna Pointing offset	et	Input	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	deg
Transmit Antenna Pointing Loss	Lpt	Eqn 13-19	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dB
Transmit Antenna Gain	Gt	Gpt+Lpt	3.87	3.87	3.87	3.87	3.87	3.87	3.87	dB
Equiv. Isotropc Radiated Power	EIRP	P+L1+Gt	2.87	2.87	2.87	2.87	2.87	2.87	2.87	dBW
Propogation Parameters										
Propogation Path Length	S	Input	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	**1.00**	km
Space Loss	Ls	Eqn 13-21	-98.46	-98.46	-98.46	-98.46	-98.46	-98.46	-98.46	dB
Propagation & Polarization Loss	La	Figure 13-10	0.00	0.00	0.00	0.00	0.00	0.00	0.00	dB
System Noise Temperature	Ts	Table 13-9	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	**100.00**	K
Receiving Station Parameters										
Receive Antenna Diameter	Dr	Input	**0.20**	**0.30**	**0.40**	**0.50**	**0.60**	**0.70**	**0.80**	m
Receive Antenna Efficiency	n	Input	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	**0.55**	-
Peak Receive Antenna Gain	Grp	Eqn 13-16	9.84	13.37	15.87	17.80	19.39	20.73	21.89	dB
Receive Antenna Beamwidth	Θr	Eqn 13-17	52.50	35.00	26.25	21.00	17.50	15.00	13.13	deg
Receive Antenna Pointing Error	er	Input	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	**2.00**	deg
Receive Antenna Pointing Loss	Lpr	Eqn 13-19	-12.00	-12.00	-12.00	-12.00	-12.00	-12.00	-12.00	dB
Receive Antenna Gain	Gr	Grp+Lpr	-2.16	1.37	3.87	5.80	7.39	8.73	9.89	dB
Final Calculations										
Data Rate	R	Input	#####	#####	#####	#####	#####	#####	#####	bps
Eb/No (1)	Eb/No	Eqn 13-11	32.79	36.32	38.81	40.75	42.34	43.68	44.84	dB
Carrier-to-Noise Densiy Ratio	C/No	Eqn 13-13	132.99	133.02	133.05	133.07	133.09	133.10	133.11	dB-Hz
Bit Error Rate	BER	Input	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	-
Required Eb/No (2)	Req Eb/No	Figure 13-9	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	**4.40**	dB-Hz
Implementation Loss (3)	-	Estimate	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	**-2.00**	dB
Margin	-	(1)-(2)+(3)	26.39	29.92	32.41	34.35	35.94	37.28	38.44	dB

low pointing error = lots of antennas
on both satellite & Station

11/7/92

Appendix G
Battery Sizing and Selection

Preliminary sizing of the battery began with the establishment of the maximum power requirement of 40 watts. The two types of rechargeable batteries which were considered are Nickel Hydrogen (NiH_2) and Nickel Cadmium (NiCd).

The NiCd has been proven to be reliable, but the NIH_2 has a longer lifetime and the capability of much larger energy densities. The lifetime of a battery depends on the number of charge-discharge cycles and the depth of discharge (DOD). DOD is the percentage of the total battery power used during discharge. The NiH_2 can undergo more charge-discharge cycles at a larger DOD than the NiCd can at a smaller DOD. For an eight hour discharge time at 40% DOD, the NiH_2 battery would have a mass of 14 kilograms while the NiCd would have a mass of 28 kilograms. The NiH_2's performance is due to its higher energy density of the NiH_2, 60 watt-hours per kilogram. The energy density of the NiCd is only 30 watt-hours per kilogram. Figure G-1 shows a graphical representation of this information.

The preliminary data show a clear superiority of the NiH_2 battery as compared to the NiCd. Although the difference in cost of the two batteries is not expected to be substantial, this factor was considered for the final recommendation. Table G-1 shows the comparison of the NiH_2 and the NiCd batteries for a 40 watt battery with 24 cells. The superiority of the NiH_2 battery is clearly shown for the lifetime, DOD, and mass.

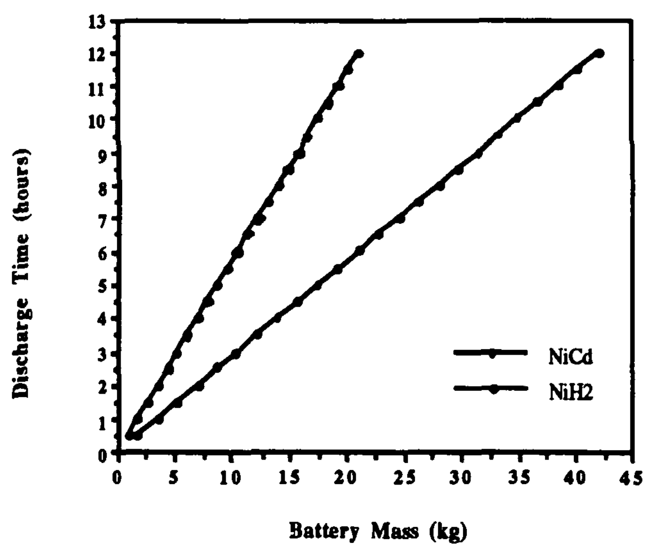

Figure G-1: Capacity(W-hrs/Kg) and Mass(KG) Compared to Discharge Time

Table G-1: Battery Comparison for 40 Watts

	NiH_2	NiCd
Maximum DOD	40%	20%
Energy Density	60 W-hr/kg	30 W-hr/kg
Lifetime	> 20000 cycles	<10000 cycles
4 hr. battery's mass	7.0 kg	28.1 kg
6 hr. battery's mass	10.5 kg	42.1 kg
8 hr. battery's mass	14.0 kg	56.1 kg

The following tables list the data obtained from a TK Solver routine that compares battery mass to discharge times. The TK Solver routine follows the tables.

Battery Mass vs. Discharge Times

```
MMMMMMMMMMMMMMMMMMMM VARIABLE SHEET MMMMMMMMMMMMMMMMMMMMM For Academic Use Only
St InputDDDD NameDDD OutputDDD UnitDDDDD CommentDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDD
L            Cr      1.8274854 A-hrs     capacity
   .4        Cd                          depth of discharge
             Vd      28.8      Volts     minimum batt voltage
   .95       n                           inefficiencies
   40        power             W         power output needed
L  .5        time              hrs       time of discharge
L            Mtotal  .87719298 Kg        total mass of battery
   60        SE                          specific energy
   24        ncell
   1.2       Vcell

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S RuleDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDD
* Vd = ncell*Vcell
* time = Mtotal*Cd*n*SE/power
* Cr = power*time/(Vd*n*Cd)
```

Mass in Kg and time in hours for NiH2

time	Mtotal	Cr, Capacity
.5	.877192982	1.82748538
1	1.75438596	3.65497076
1.5	2.63157895	5.48245614
2	3.50877193	7.30994152
2.5	4.38596491	9.1374269
3	5.26315789	10.9649123
3.5	6.14035088	12.7923977
4	7.01754386	14.619883
4.5	7.89473684	16.4473684
5	8.77192982	18.2748538
5.5	9.64912281	20.1023392
6	10.5263158	21.9298246
6.5	11.4035088	23.7573099
7	12.2807018	25.5847953
7.5	13.1578947	27.4122807
8	14.0350877	29.2397661
8.5	14.9122807	31.0672515
9	15.7894737	32.8947368
9.5	16.6666667	34.7222222
10	17.5438596	36.5497076

Mass in Kg and time in hours for NiCd

time	Mtotal	Cr, Capacity
.5	1.75438596	1.82748538
1	3.50877193	3.65497076
1.5	5.26315789	5.48245614
2	7.01754386	7.30994152
2.5	8.77192982	9.1374269
3	10.5263158	10.9649123
3.5	12.2807018	12.7923977
4	14.0350877	14.619883
4.5	15.7894737	16.4473684
5	17.5438596	18.2748538
5.5	19.2982456	20.1023392
6	21.0526316	21.9298246
6.5	22.8070175	23.7573099
7	24.5614035	25.5847953
7.5	26.3157895	27.4122807
8	28.0701754	29.2397661
8.5	29.8245614	31.0672515
9	31.5789474	32.8947368
9.5	33.3333333	34.7222222
10	35.0877193	36.5497076

Appendix H
Propellant Study

The performance study of cold gases has been achieved through a comparison of the mass of propellant required for a satellite of a given mass to maneuver at a given delta-V. Figure H-1 shows the general relationship between the mass of the satellite, mass of propellant, and delta-V for freon. The other gases behave similarly. The data show that the higher the specific impulse the less propellant is needed. For example, freon would use 3.0 pounds of propellant for a delta-V of 21 feet per second if the initial satellite mass is 200 pounds.

As important as mass, the volume required to hold the propellant has been studied. At 3500 psia and 273 Kelvin, the densities of the cold gases were computed to estimate the volume of propellant for a given delta-V. Figure H-2 shows the results of these estimates. Carbon dioxide and freon clearly out perform the other gases when considering volume. Freon and carbon dioxide volume estimates result in 0.19 and 0.31 cubic feet of propellant, respectively, for a delta-V of 20 feet per second and satellite mass of 100 pounds. Carbon dioxide is a waste gases from SSF, thus it would be available with no transportation or purchase cost. Freon might be inexpensive and available due to the current debate over detrimental effect on the atmosphere.

The following information includes data for the leading cold gases in a similar comparison as Figure H-2 and a TK Solver routine which was used in the calculations of the propellant. mass.

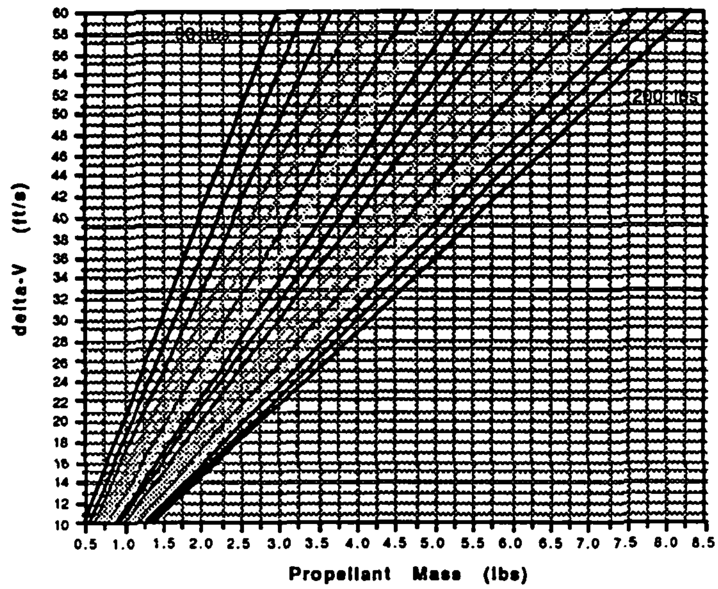

Figure H-1: Freon's Mass and Delta-V Relationships

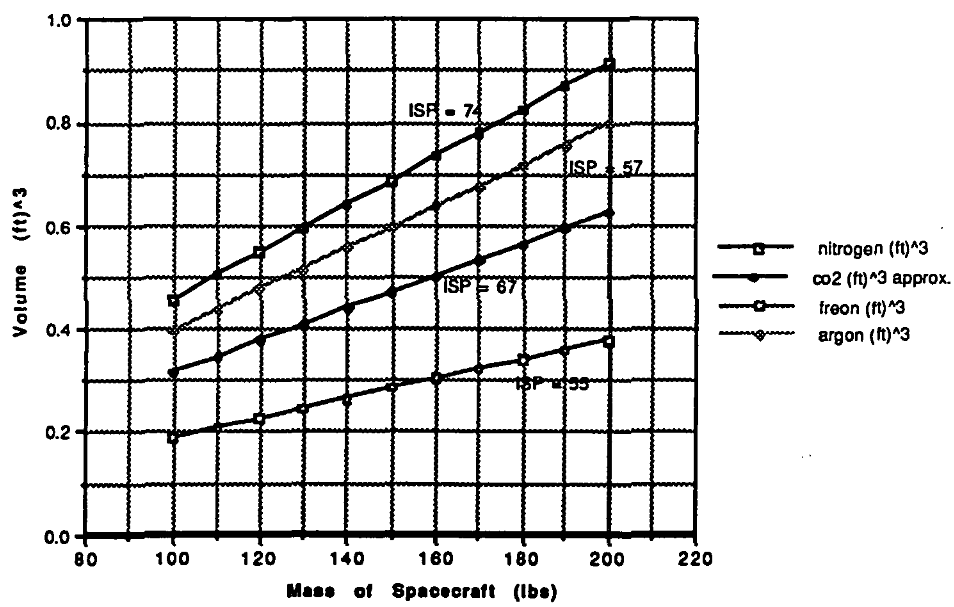

Figure H-2: Cold Gas Volume Estimates for a Delta-V of 10 ft/sec

Cold Gas Data

dV= 5 ft/s for CO2(density=31.2 lb/ft^3)

iter	Mp	Mo	Mf
1	.370205646	150	149.629794
2	.369291965	149.629794	149.260502
3	.368380538	149.260502	148.892122
4	.367471361	148.892122	148.52465
5	.366564428	148.52465	148.158086
6	.365659733	148.158086	147.792426
7	.364757271	147.792426	147.427669
8	.363857037	147.427669	147.063812
9	.362959024	147.063812	146.700853
10	.362063227	146.700853	146.33879
11	.361169642	146.33879	145.97762
12	.360278261	145.97762	145.617342
13	.359389081	145.617342	145.257953
14	.358502095	145.257953	144.899451
15	.357617299	144.899451	144.541833
16	.356734686	144.541833	144.185099
17	.355854251	144.185099	143.829244
18	.354975989	143.829244	143.474268
19	.354099895	143.474268	143.120169
20	.353225963	143.120169	142.766943
21	.352354188	142.766943	142.414588
22	.351484565	142.414588	142.063104
23	.350617088	142.063104	141.712487
24	.349751752	141.712487	141.362735
25	.348888551	141.362735	141.013846
26	.348027481	141.013846	140.665819
27	.347168536	140.665819	140.31865

dV= 10 ft/s for CO2(density=31.2 lb/ft^3)

iter	Mp	Mo	Mf
1	.739497611	150	149.260502
2	.7358519	149.260502	148.52465
3	.732224162	148.52465	147.792426
4	.728614308	147.792426	147.063812
5	.725022251	147.063812	146.33879
6	.721447903	146.33879	145.617342
7	.717891176	145.617342	144.899451
8	.714351984	144.899451	144.185099
9	.71083024	144.185099	143.474268
10	.707325859	143.474268	142.766943
11	.703838753	142.766943	142.063104
12	.70036884	142.063104	141.362735
13	.696916032	141.362735	140.665819
14	.693480247	140.665819	139.972339
15	.690061401	139.972339	139.282277
16	.686659409	139.282277	138.595618
17	.683274189	138.595618	137.912344
18	.679905658	137.912344	137.232438
19	.676553734	137.232438	136.555884
20	.673218335	136.555884	135.882666
21	.669899379	135.882666	135.212767

dV= 20 ft/s for CO2(density=31.2 lb/ft^3)

iter	Mp	Mo	Mf
1	1.47534951	150	148.52465
2	1.46083847	148.52465	147.063812
3	1.44647015	147.063812	145.617342
4	1.43224316	145.617342	144.185099
5	1.4181561	144.185099	142.766943
6	1.40420759	142.766943	141.362735
7	1.39039628	141.362735	139.972339
8	1.37672081	139.972339	138.595618
9	1.36317985	138.595618	137.232438
10	1.34977207	137.232438	135.882666
11	1.33649617	135.882666	134.54617
12	1.32335084	134.54617	133.222819
13	1.31033481	133.222819	131.912484
14	1.29744679	131.912484	130.615037
15	1.28468554	130.615037	129.330352
16	1.27204981	129.330352	128.058302
17	1.25953836	128.058302	126.798764
18	1.24714996	126.798764	125.551614
19	1.23488341	125.551614	124.31673
20	1.22273752	124.31673	123.093993
21	1.21071108	123.093993	121.883282
22	1.19880293	121.883282	120.684479
23	1.18701191	120.684479	119.497467
24	1.17533686	119.497467	118.32213
25	1.16377664	118.32213	117.158353
26	1.15233013	117.158353	116.006023
27	1.1409962	116.006023	114.865027

dV= 100 ft/s for CO2(density=31.2 lb/ft^3)

iter	Mp	Mo	Mf
1	7.23305739	150	142.766943
2	6.8842766	142.766943	135.882666
3	6.55231415	135.882666	129.330352
4	6.23635905	129.330352	123.093993
5	5.93563943	123.093993	117.158353
6	5.64942063	117.158353	111.508933
7	5.3770034	111.508933	106.131929
8	5.11772224	106.131929	101.014207
9	4.87094372	101.014207	96.1432634
10	4.63606495	96.1432634	91.5071984
11	4.41251212	91.5071984	87.0946863
12	4.1997391	87.0946863	82.8949472
13	3.99722607	82.8949472	78.8977211
14	3.8044783	78.8977211	75.0932428
15	3.6210249	75.0932428	71.4722179
16	3.4464177	71.4722179	68.0258002
17	3.28023012	68.0258002	64.7455701
18	3.12205616	64.7455701	61.623514
19	2.97150942	61.623514	58.6520045
20	2.8282221	58.6520045	55.8237824
21	2.69184415	55.8237824	53.1319383
22	2.56204239	53.1319383	50.5698959
23	2.43849973	50.5698959	48.1313962
24	2.32091434	48.1313962	45.8104818
25	2.20899896	45.8104818	43.6014829
26	2.10248019	43.6014829	41.4990027
27	2.00109779	41.4990027	39.4979049

dV= 5 ft/s for Freon(density=60.01 lb/ft^3)

iter	Mp	Mo	Mf
1	.450856288	150	149.549144
2	.449501145	149.549144	149.099643
3	.448150076	149.099643	148.651492
4	.446803067	148.651492	148.204689
5	.445460107	148.204689	147.759229
6	.444121184	147.759229	147.315108
7	.442786285	147.315108	146.872322
8	.441455399	146.872322	146.430866
9	.440128512	146.430866	145.990738
10	.438805614	145.990738	145.551932
11	.437486692	145.551932	145.114446
12	.436171735	145.114446	144.678274
13	.43486073	144.678274	144.243413
14	.433553665	144.243413	143.80986
15	.432250529	143.80986	143.377609
16	.43095131	143.377609	142.946658
17	.429655996	142.946658	142.517002
18	.428364575	142.517002	142.088637
19	.427077036	142.088637	141.66156
20	.425793367	141.66156	141.235767
21	.424513556	141.235767	140.811253

dV= 10 ft/s for Freon(density=60.01 lb/ft^3)

iter	Mp	Mo	Mf
1	.900357433	150	149.099643
2	.894953143	149.099643	148.204689
3	.889581291	148.204689	147.315108
4	.884241684	147.315108	146.430866
5	.878934127	146.430866	145.551932
6	.873658427	145.551932	144.678274
7	.868414395	144.678274	143.80986
8	.863201839	143.80986	142.946658
9	.858020571	142.946658	142.088637
10	.852870403	142.088637	141.235767
11	.847751149	141.235767	140.388016
12	.842662622	140.388016	139.545353
13	.837604638	139.545353	138.707748
14	.832577014	138.707748	137.875171
15	.827579568	137.875171	137.047592
16	.822612119	137.047592	136.22498
17	.817674486	136.22498	135.407305
18	.812766491	135.407305	134.594539
19	.807887955	134.594539	133.786651
20	.803038702	133.786651	132.983612
21	.798218556	132.983612	132.185393

dV= 20 ft/s for Freon(density=60.01 lb/ft^3)

iter	Mp	Mo	Mf
1	1.79531058	150	148.204689
2	1.77382297	148.204689	146.430866
3	1.75259255	146.430866	144.678274
4	1.73161623	144.678274	142.946658
5	1.71089097	142.946658	141.235767
6	1.69041377	141.235767	139.545353
7	1.67018165	139.545353	137.875171
8	1.65019169	137.875171	136.22498
9	1.63044098	136.22498	134.594539
10	1.61092666	134.594539	132.983612
11	1.5916459	132.983612	131.391966
12	1.57259591	131.391966	129.81937
13	1.55377392	129.81937	128.265596
14	1.53517721	128.265596	126.730419
15	1.51680308	126.730419	125.213616
16	1.49864886	125.213616	123.714967
17	1.48071192	123.714967	122.234255
18	1.46298967	122.234255	120.771265
19	1.44547953	120.771265	119.325786
20	1.42817897	119.325786	117.897607
21	1.41108547	117.897607	116.486522

dV= 10 ft/s for Methane(density= 12.10 lb/ft^3)

iter	Mp	Mo	Mf
1	.434757857	150	149.565242
2	.433497761	149.565242	149.131744
3	.432241317	149.131744	148.699503
4	.430988515	148.699503	148.268515
5	.429739344	148.268515	147.838775
6	.428493794	147.838775	147.410281
7	.427251853	147.410281	146.98303
8	.426013513	146.98303	146.557016
9	.424778761	146.557016	146.132237
10	.423547588	146.132237	145.70869
11	.422319984	145.70869	145.28637
12	.421095938	145.28637	144.865274
13	.41987544	144.865274	144.445398
14	.418658479	144.445398	144.02674
15	.417445045	144.02674	143.609295
16	.416235128	143.609295	143.19306
17	.415028718	143.19306	142.778031
18	.413825805	142.778031	142.364205
19	.412626378	142.364205	141.951579
20	.411430428	141.951579	141.540148

dV= 100 ft/s for Methane(density= 12.10 lb/ft^3)

iter	Mp	Mo	Mf
1	4.2913103	150	145.70869
2	4.16854134	145.70869	141.540148
3	4.04928465	141.540148	137.490864
4	3.93343973	137.490864	133.557424
5	3.820909	133.557424	129.736515
6	3.71159762	129.736515	126.024917
7	3.60541351	126.024917	122.419504
8	3.50226719	122.419504	118.917237
9	3.40207175	118.917237	115.515165
10	3.30474278	115.515165	112.210422
11	3.21019827	112.210422	109.000224
12	3.11835856	109.000224	105.881865
13	3.02914626	105.881865	102.852719
14	2.94248622	102.852719	99.9102328
15	2.85830541	99.9102328	97.0519274
16	2.77653291	97.0519274	94.2753945
17	2.69709981	94.2753945	91.5782947
18	2.6199392	91.5782947	88.9583555
19	2.54498605	88.9583555	86.4133694
20	2.47217722	86.4133694	83.9411922

dV= 100 ft/s for Freon(density=60.01 lb/ft^3)

iter	Mp	Mo	Mf
1	8.76423331	150	141.235767
2	8.25215474	141.235767	132.983612
3	7.76999601	132.983612	125.213616
4	7.31600896	125.213616	117.897607
5	6.88854756	117.897607	111.009059
6	6.48606198	111.009059	104.522997
7	6.10709291	104.522997	98.4159045
8	5.75026633	98.4159045	92.6656382
9	5.41428849	92.6656382	87.2513497
10	5.09794124	87.2513497	82.1534085
11	4.8000776	82.1534085	77.3533309
12	4.5196176	77.3533309	72.8337133
13	4.25554437	72.8337133	68.5781689
14	4.00690048	68.5781689	64.5712684
15	3.77278441	64.5712684	60.798484
16	3.55234733	60.798484	57.2461367
17	3.34478999	57.2461367	53.9013467
18	3.14935986	53.9013467	50.7519868
19	2.96534836	50.7519868	47.7866385
20	2.79208833	47.7866385	44.9945502
21	2.62895157	44.9945502	42.3655986

dV= 5 ft/s for Methane(density= 12.10 lb/ft^3)

iter	Mp	Mo	Mf
1	.217536669	150	149.782463
2	.217221188	149.782463	149.565242
3	.216906164	149.565242	149.348336
4	.216591597	149.348336	149.131744
5	.216277486	149.131744	148.915467
6	.215963831	148.915467	148.699503
7	.215650631	148.699503	148.483852
8	.215337884	148.483852	148.268515
9	.215025592	148.268515	148.053489
10	.214713752	148.053489	147.838775
11	.214402365	147.838775	147.624373
12	.214091429	147.624373	147.410281
13	.213780944	147.410281	147.1965
14	.213470909	147.1965	146.98303
15	.213161324	146.98303	146.769868
16	.212852188	146.769868	146.557016
17	.212543501	146.557016	146.344473
18	.212235261	146.344473	146.132237
19	.211927468	146.132237	145.92031
20	.211620121	145.92031	145.70869

dV= 5 ft/s for Methane(density= 12.10 lb/ft^3)

iter	Mp	Mo	Mf
21	.21131322	145.70869	145.497376
22	.211006764	145.497376	145.28637
23	.210700753	145.28637	145.075669
24	.210395185	145.075669	144.865274
25	.210090061	144.865274	144.655184
26	.209785379	144.655184	144.445398
27	.209481139	144.445398	144.235917
28	.20917734	144.235917	144.02674
29	.208873982	144.02674	143.817866
30	.208571063	143.817866	143.609295
31	.208268584	143.609295	143.401026
		143.401026	

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(3i) Input: 100                                                        253 /F9

MMMMMMMMMMMMMMMMMMMMM VARIABLE SHEET MMMMMMMMMMMMMMMMMMMMM For Academic Use Only
St InputDDDD NameDDD OutputDDD UnitDDDDD CommentDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDD
L             Mp      8.7583246 lb        Propellant mass for one burn
L  150        Mo                lb        Initial mass at the burn
   100        deltaV            ft/s      Change in velocity
   67         Isp               s         Specific Impulse
   30.2       g                 ft/s^2    Acceleration of gravity
L             Mf      141.24168 lb        Final mass after the burn
L             iter

MMMMMMMMMMMMMMMMMMMMM RULE SHEET MMMMMMMMMMMMMMMMMMMMMMMMM For Academic Use Only
S RuleDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDDD
  iter = elt()
  Mp = Mo*(1- exp(-deltaV/(Isp*g)))
  Mf = Mo - Mp
  Mf = place('Mo,iter + 1)



 F1 Help  F2 Cancel  F5 Edit  F9 Solve  / Commands  = Sheets  ; Window switch
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Appendix I
Satellite Health and Status Data

The following is a list of spacecraft data that will need to be monitored by the AERCAM satellite and a command and control station. This data will need to be telemetered with the Satellite health and status data through the Space Station communication link. This list was compiled from information in Space Mission Analysis and Design, by J. R. Wertz. It presents a number of suggested data for monitoring, but is in no way a comprehensive list. The data has been divided into groups according to subsystem.

Power Functions

Battery on-line status
Battery power disconnect status
Unregulated bus voltage
Unregulated bus current
Battery current
Battery temperature
Other temperatures
Bus voltage
Bus current
Each subsystem current
Redundancy status

Telemetry Functions

Telemetry clock
Telemetry ready signal
Power from Power system
Temperature
Redundancy status
Redundant system status

Subsystem Functions

Redundancy status
Redundat system status

Telecommand Functions

Power from power system
Temperature
Redundancy Status
Status of various command relays

Attitude Control System Functions

Power from power system
Heater power
Fine error sensor, Roll, (+)
Fine error sensor, Roll, (-)
Fine error sensor, Pitch, (+)
Fine error sensor, Pitch, (-)
Fine error sensor, Yaw, (+)
Fine error sensor, Yaw, (-)
Direct control output, Roll, (+)
Direct control output, Roll, (-)
Direct control output, Pitch, (+)
Direct control output, Pitch, (-)
Direct control output, Yaw, (+)
Direct control output, Yaw, (-)
Gyro heater power
Gyro signal
Gyro temperature
Sun sensor, Pitch
Sun sensor, Yaw
Horizon Sensor
Pulse valve
Temperature
Attitude Integration error signal
State vector
Command Storage Registers
Equipment voltages
Redundant systems status

Propulsion Functions

Power from power system
Heater power
Direct control, Roll
Direct control, Pitch
Direct control, Yaw
Propellant Tank temperature
Propellant Tank pressure
Propellant Latch Valve status
Temperature
Thruster status for each thruster
Redundancy status
Redundant system status

Antennae Functions

Temperature
Strain gauge
Receiver signal power
Receiver AGC
Transmitter output power

Tracking Functions

Fine phase
Coarse phase
Reference phase
Analog phase
Receiver signal power

Imaging System Functions

Power from power system
Heater power
Temperature (IR and visual cameras)
Pitch, Roll, and Yaw data (see Attitude Control System)
Zoom
Data Reduction Technique
IR camera operational status
Visual camera operational status
Image Processor operational status
Light operational status
Component power status (computer, IR camera, visual camera, light)

Appendix J
Micro Interface Design Specifications

Grappling of an AERCAM satellite by the SPDM will be made possible by the use of a Micro Interface grapple fixture. Detailed design specifications on the Micro Interface grapple fixture are shown in Table J-1. This information is taken from Robotic Interface Standards, page 3-17.

Micro Interface Design Specifications	
Part Number	31459E530
Weight (lb)	0.227 lb.
Footprint (length x width, in.)	1.625 in. x 1.625 in.
Material (Specify)	17–4PH
Design Temperature Range (Deg F)	TBD
Structure Volume (in^3)	3.16 in^3
Max Payload Handling Capability (lb)	500 lb.
Compatible Tools	ORU–Tool Changeout Mechanism (OTCM)
Micro Interface Operational Data	
Maximum Force Along Axes	
X Axis (lb)	3200 lb. (Incorporates Factor of Safety = 3)*
Y Axis (lb)	3200 lb. (Incorporates Factor of Safety = 3)*
Z Axis (lb)	4500 lb. (Incorporates Factor of Safety = 3)*
Maximum Torque Along Axes	
X Axis (ft–lb)	140 ft–lb. (Incorporates Factor of Safety = 3)*
Y Axis (ft–lb)	250 ft–lb. (Incorporates Factor of Safety = 3)*
Z Axis (ft–lb)	170 ft–lb. (Incorporates Factor of Safety = 3)*
Maximum Misalignment Tolerances	
Positional (in)	± 0.3 in. in X and Y, ± 0.25 in. in Z **
Pitch (deg) (around X Axis)	± 10 deg.**
Yaw (deg) (around Y Axis)	± 10 deg.**
Roll (deg) (around Z Axis)	± 10 deg.**

Table J-1: Micro Interface Design Specifications

Appendix K
TK Model of Suborbit and DV Plots

This appendix contains two main things: I) TK Solver Model and II.) Propellant budget figures for stationkeeping.

I.) The first thing is the TK Solver model coding of the Clohessy-Wiltshire equations. The figures are generated in local vertical and local horizontal reference frame, where the Station is the center of the inertial frame, i.e., the motion of the satellite is shown relative to the Station. The following figures were generated from the TK Solver. They are the following:

1) X Z plane positions
2) YZ plane positions
3) XZ plane velocities
4) YZ plane velocities
5) X vs Xdot
6) Positions vs time
7) Velocities vs times.

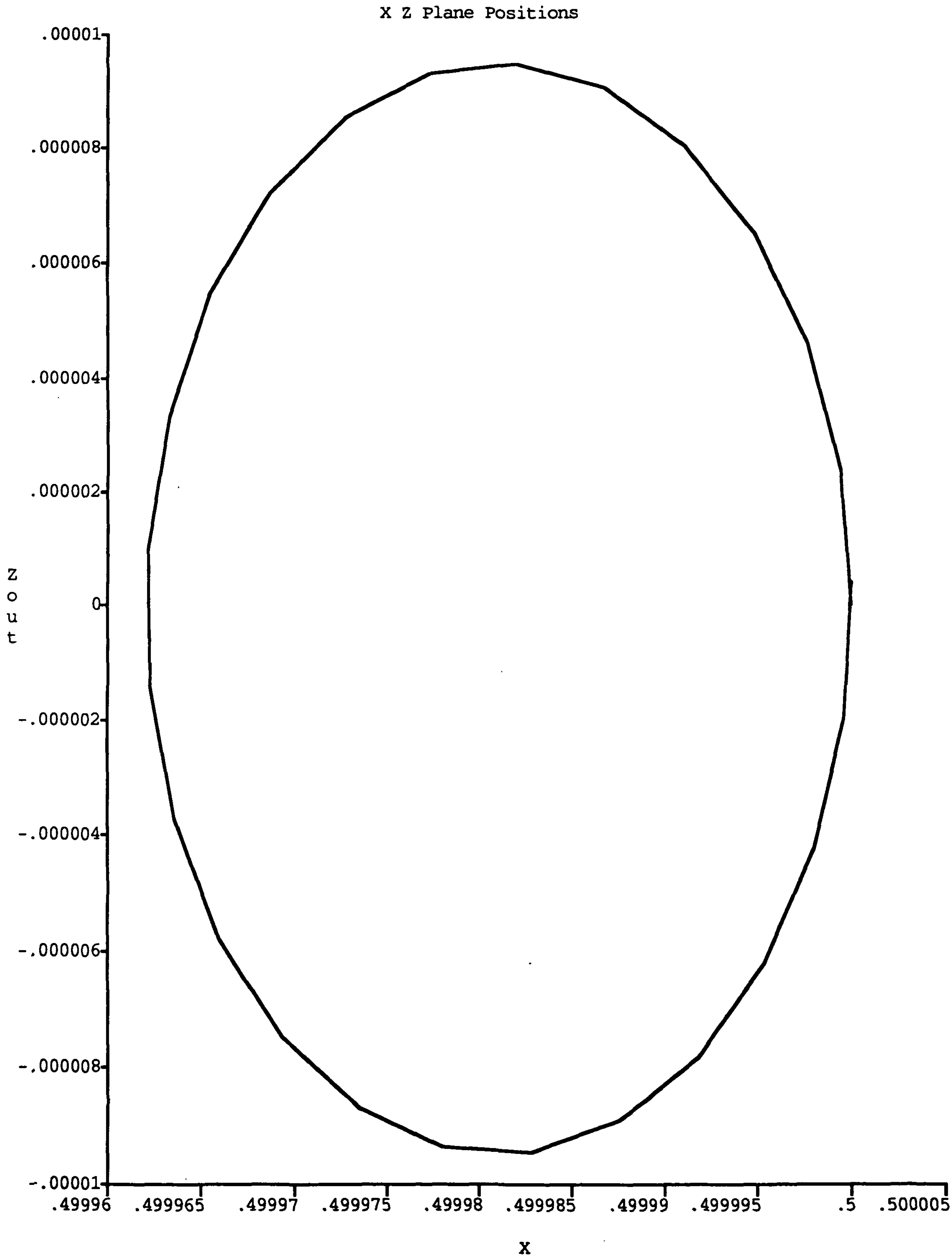
A table generated from TK Solver is also attached. It contains the following parameters:

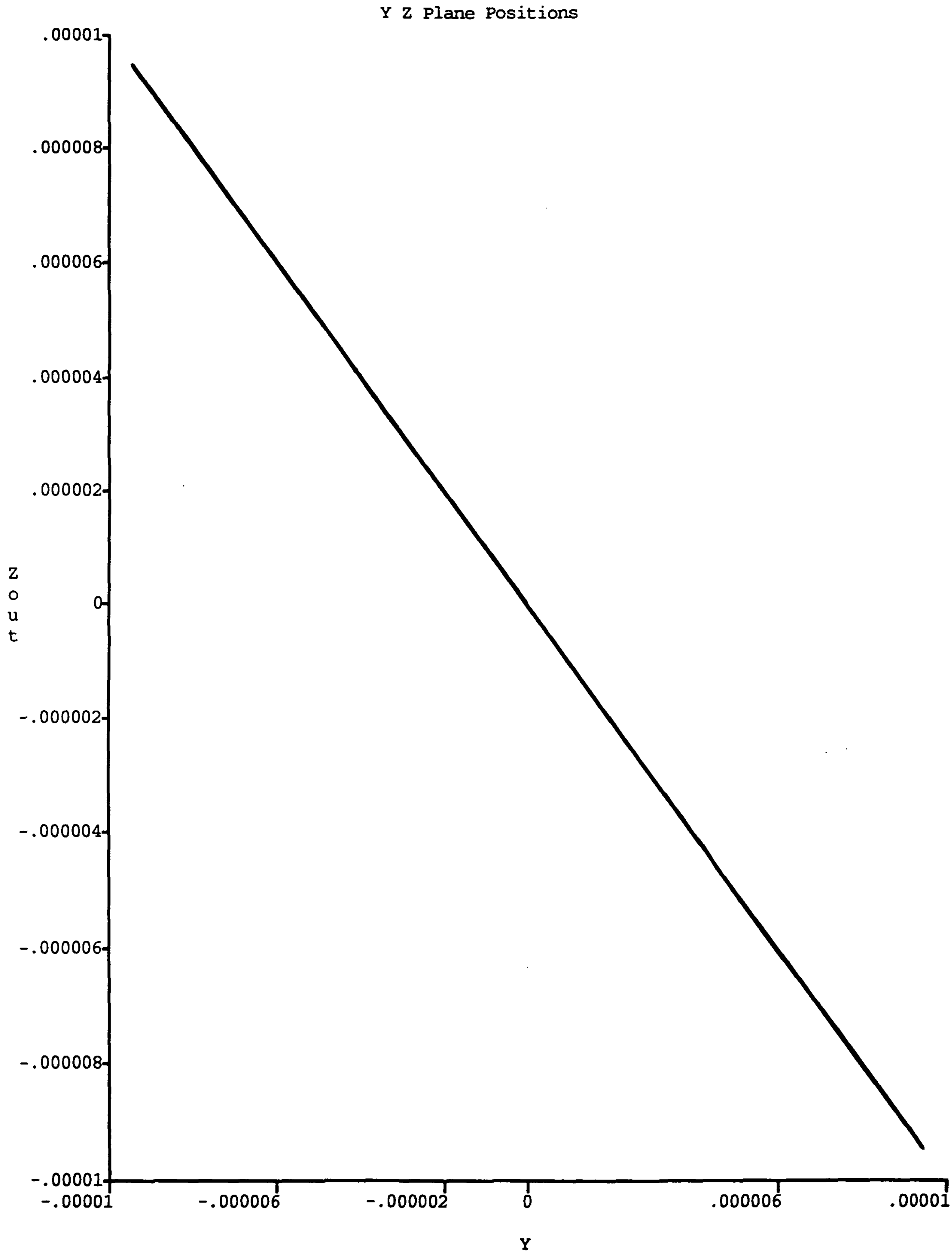
1) time
2) x-position
3) y-position
4) z-position
5) x-velocity
6) y-velocity
7) z-velocity

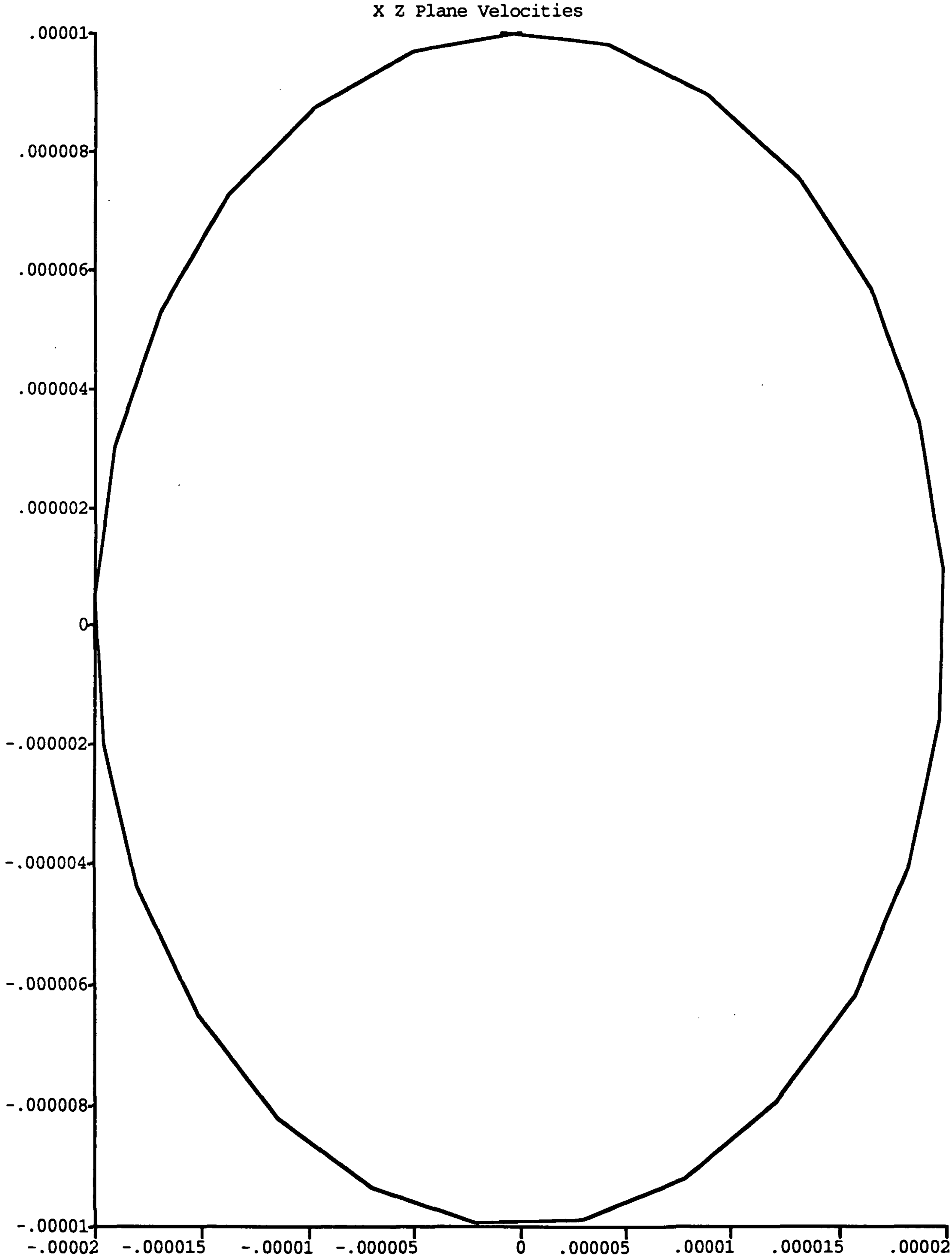
II.) The second thing are the figures that show the delta velocities (dv's) needed to get out at different locations around the Station and then stationkeep. The figures were generated on Deltagraph on the Macintosh and were generated using Clohessy-Wiltshire equations and the software (Clohessy-Wiltshire Propagation) CWPROP that was developed by Don Pearson of Johnson Space Center of NASA. The propellant budget was modeled using the following initial conditions. The satellite was assumed to start from the Station. Thus it had an initial position of 0 feet in the x-direction, 0 feet in the y-direction, and 0 feet in the z-direction and an initial velocity of 0 feet per second in the x-direction, 0 feet per second in the y-direction, and 0 feet per second in the z-direction. Time ranged from 5 minutes to 60 minutes. CWPROP outputted the final positions and velocity for each burn. (CWPROP software is not attached.)

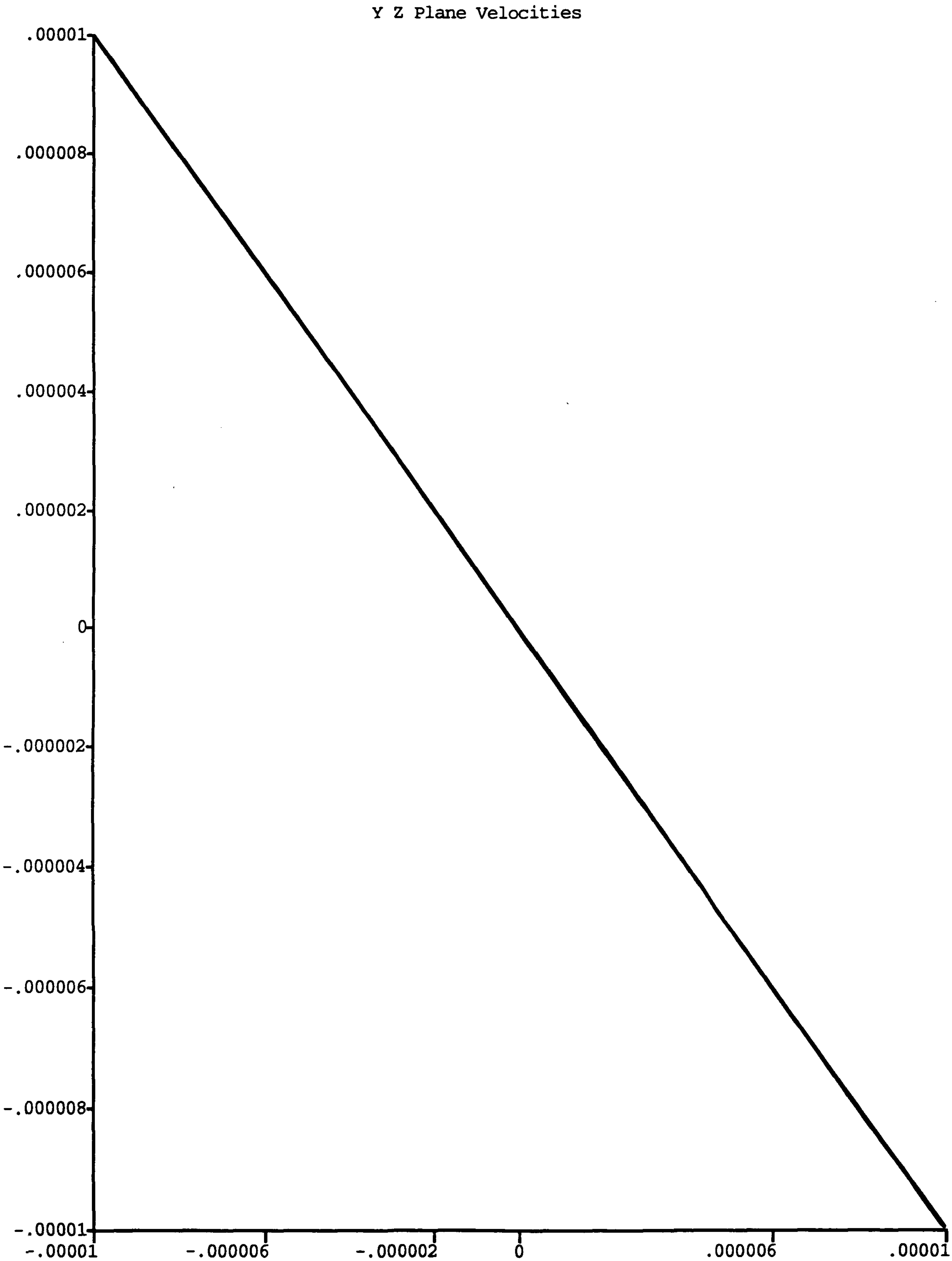
```
S Rule
  "Basic CW Equations - RELATIVE MOTION TRAJECTORIES

  "The equations for the propagation of the relative state are:

* x = -2*S2*cos(w*t)+ 2*S1*sin(w*t)+ 3*S3*w*t+xo+2*S2
* y = yo*cos(w*t)+(vyo/w)*sin(w*t)
* z = S1*cos(w*t)+S2*sin(w*t)+2*S3
* vx = 2*S1*w*cos(w*t)+2*vzo*sin(w*t)+ 3*S3*w
* vy = -yo*w*sin(w*t)+vyo*cos(w*t)
* vz = -S1*w*sin(w*t)+vzo*cos(w*t)

  "where S1, S2, S3, and w are given by:

* S1 = 2*vxo/w-3*zo
* S2 = vzo/w
* S3 = 2*zo-vxo/w
* w = sqrt(Ge*Re^2/ro^3)


  "The distance of the second body from the origin of the LVLH system
  "is given by:

* dist = sqrt( x^2 + y^2 + z^2)

  "The following equations set up new output variables which make the
  " interpretation of plotted output easier ( zout is up).

* zout = -z
* vzout = -vz
* r=sqrt(x*x+y*y+z*z)
```

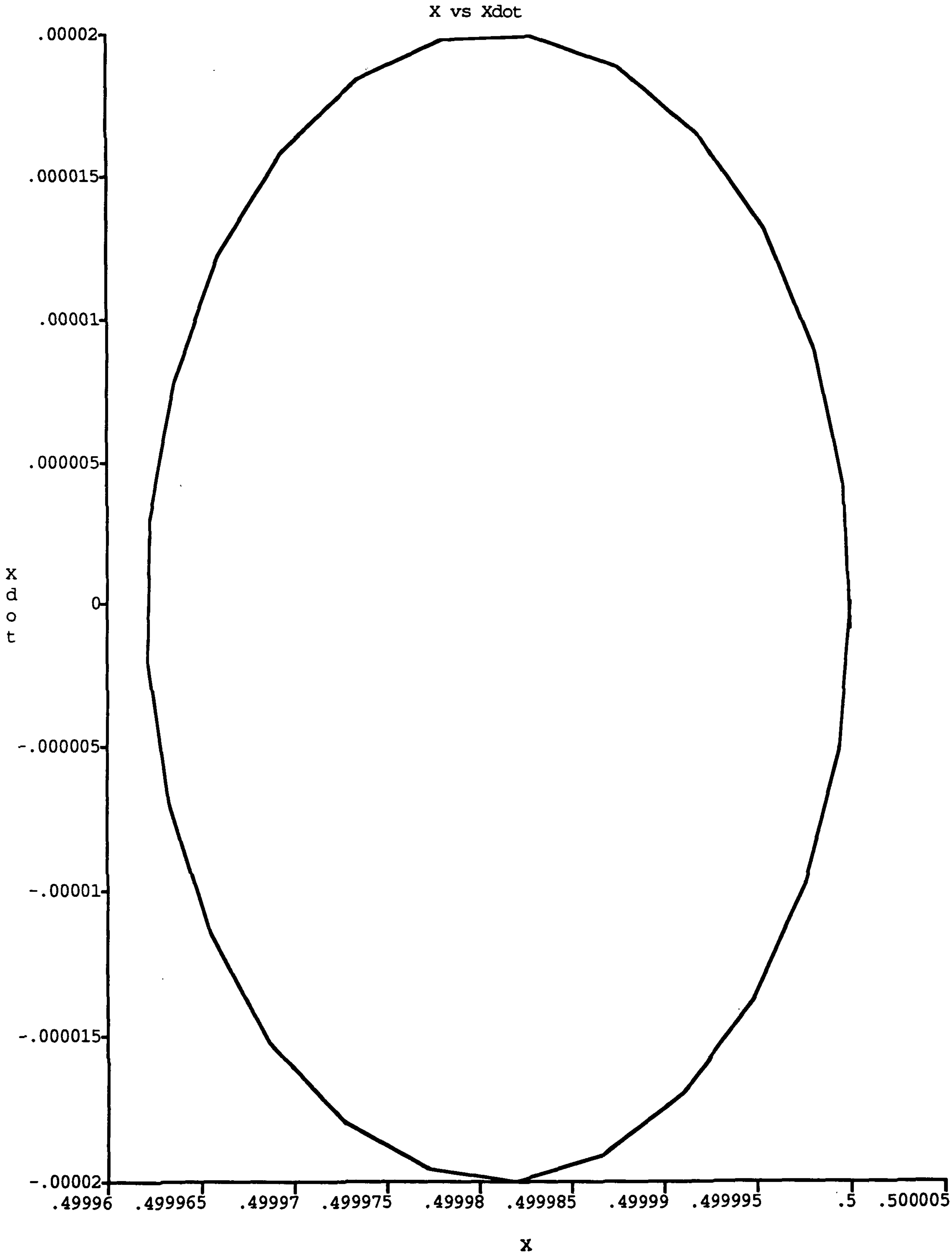
St	Input	Name	Output	Unit	Comment
	6376.436	Re		km	Radius of the Earth
	32.17	Ge		ft/s^2	Earth's Surface Gravitational Accel.
	7104.43	ro		km	Base Orbit Radius (LVLH Origin)
L	100	t		min	Time at which State is desired
	.5	xo		km	Initial Position Vector Component
	0	yo		km	Initial Position Vector Component
	0	zo		km	Initial Position Vector Component
	0	vxo		m/s	Initial Velocity Vector Component
	-.00001	vyo		m/s	Initial Velocity Vector Component
	-.00001	vzo		m/s	Initial Velocity Vector Component
L		x	.49999998	km	Final Position Vector Component
L		y	-4.113E-7	km	Final Position Vector Component
L		z	-4.113E-7	km	Final Position Vector Component
L		vx	-8.674E-7	m/s	Final Velocity Vector Component
L		vy	-9.991E-6	m/s	Final Velocity Vector Component
L		vz	-9.991E-6	m/s	Final Velocity Vector Component
		S1	0		Equation Coefficient
		S2	-.0311149		Equation Coefficient
		S3	0		Equation Coefficient
		w	.00105443		Orbital Angular Rate
L		dist	.49999998	km	Final Distance From Origin
L		zout	.00134953	ft	
L		vzout	3.2778E-5	ft/s	
		r	1640.4199	km	

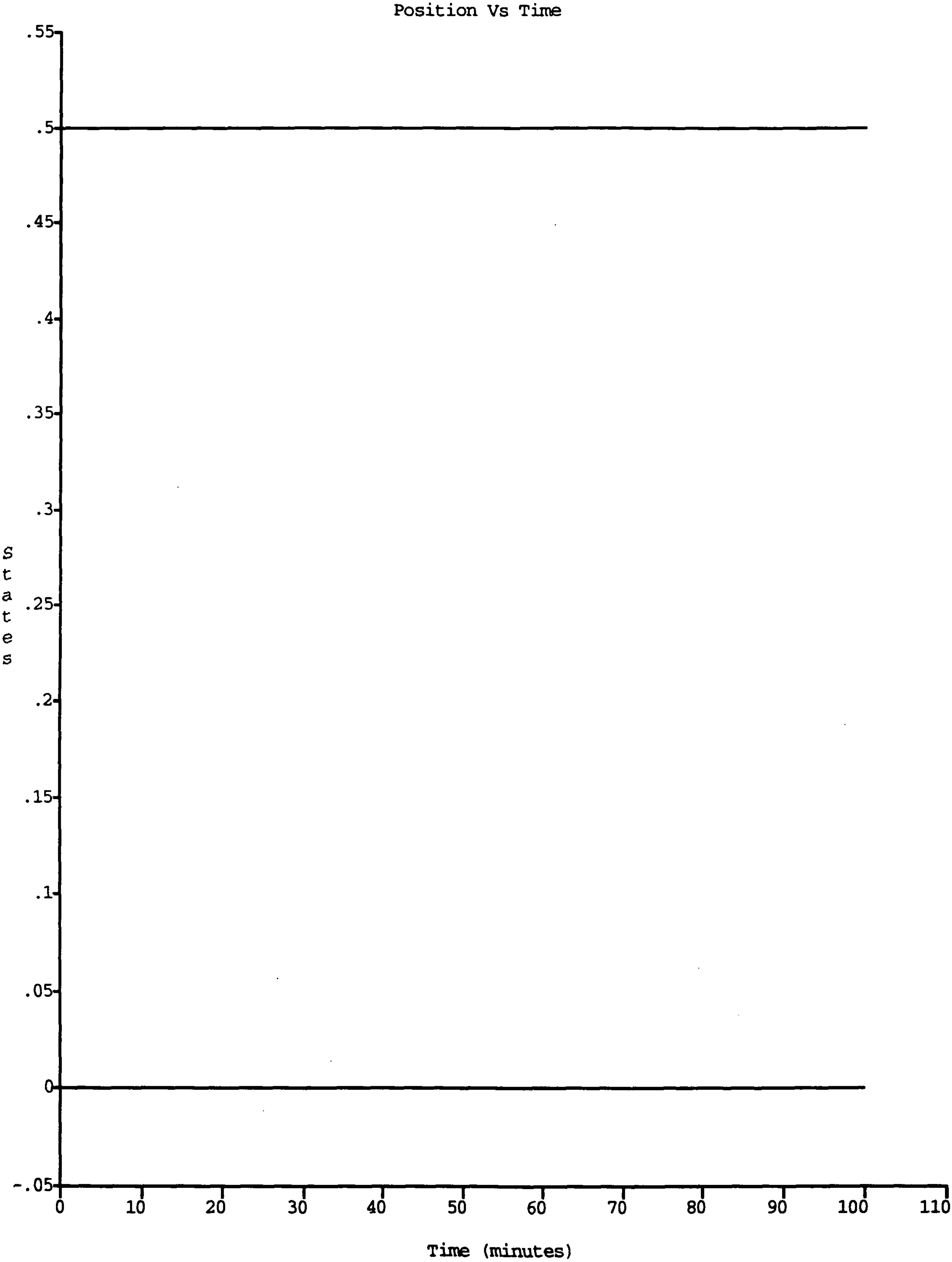

X Z Plane Positions
.00001
.000008
.000006
.000004
.000002
0
-.000002
-.000004
-.000006
-.000008
-.00001
Z o u t
.49996
.499965
.49997
.499975
.49998
.499985
.49999
.499995
.5
.500005
X

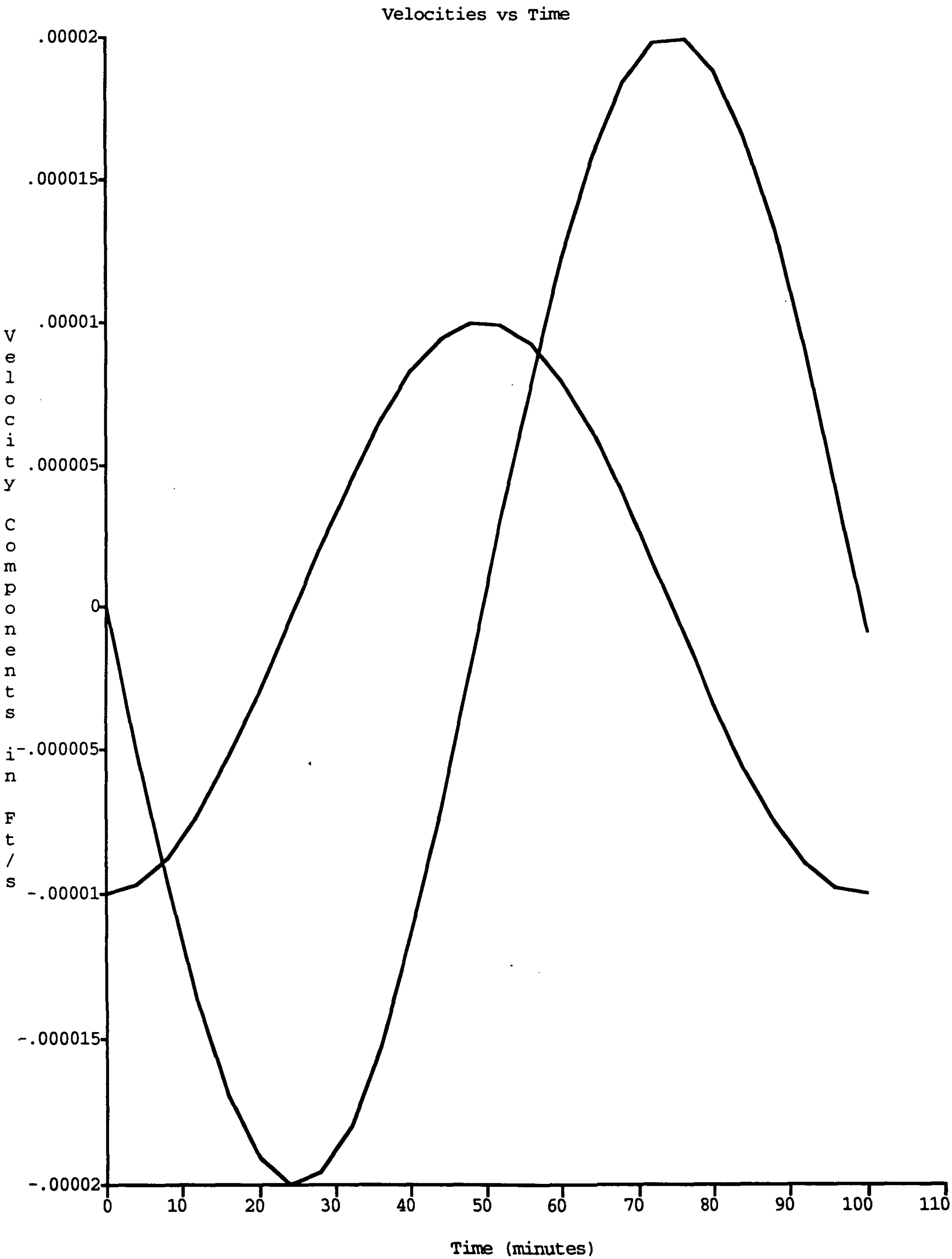

Y Z Plane Positions
.00001
.000008
.000006
.000004
.000002
0
-.000002
-.000004
-.000006
-.000008
-.00001
Z out
-.00001
-.000006
-.000002
0
.000006
.00001
Y


X Z Plane Velocities
.00001
.000008
.000006
.000004
.000002
0
-.000002
-.000004
-.000006
-.000008
-.00001
-.00002
-.000015
-.00001
-.000005
0
.000005
.00001
.000015
.00002


Y Z Plane Velocities
.00001
.000008
.000006
.000004
.000002
0
-.000002
-.000004
-.000006
-.000008
-.00001
-.00001
-.000006
-.000002
0
.000006
.00001


X vs Xdot
.00002
.000015
.00001
.000005
0
-.000005
-.00001
-.000015
-.00002
Xdot
.49996
.499965
.49997
.499975
.49998
.499985
.49999
.499995
.5
.500005
X


Position Vs Time
States
.55
.5
.45
.4
.35
.3
.25
.2
.15
.1
.05
0
-.05
0
10
20
30
40
50
60
70
80
90
100
110
Time (minutes)

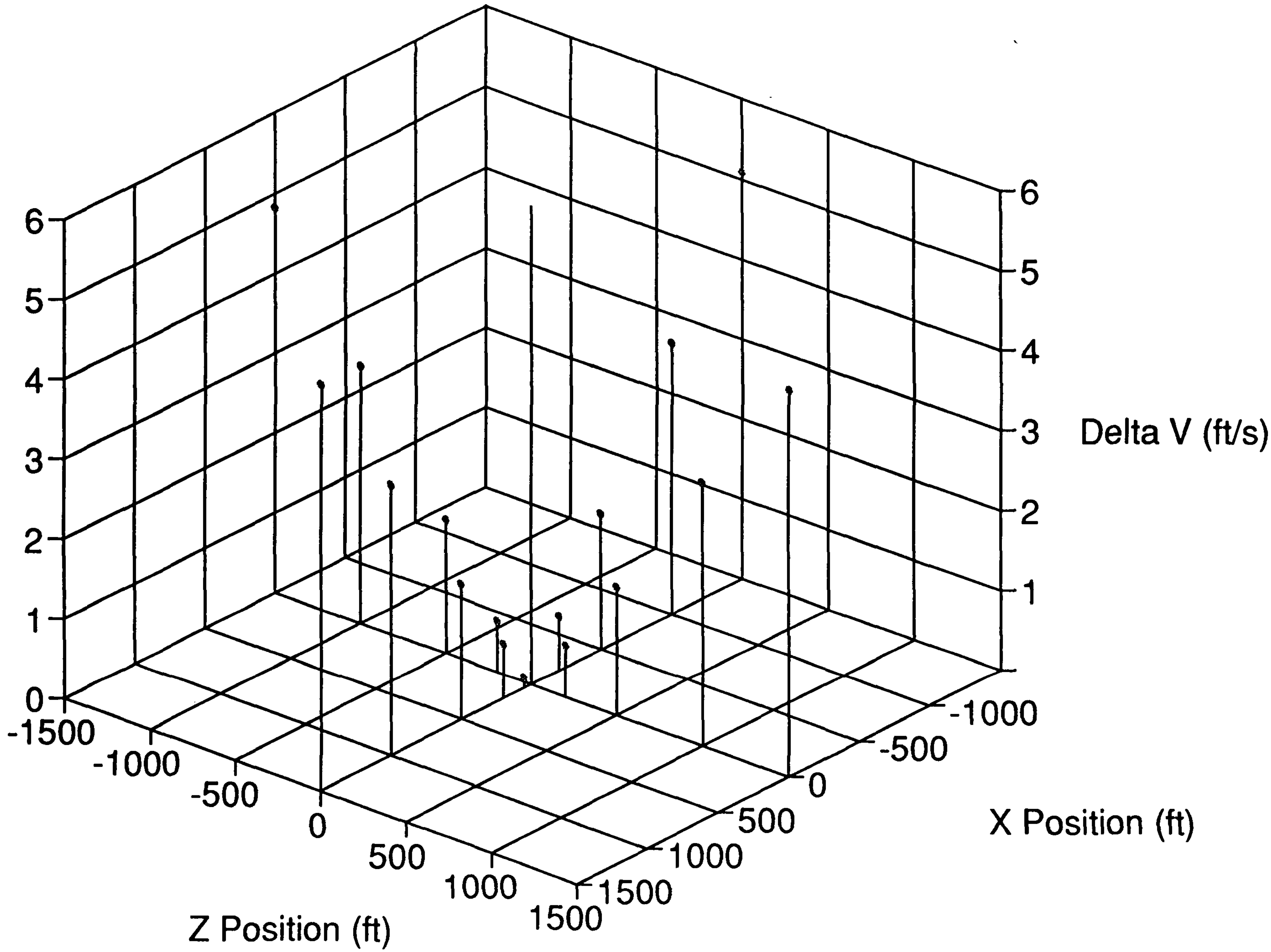


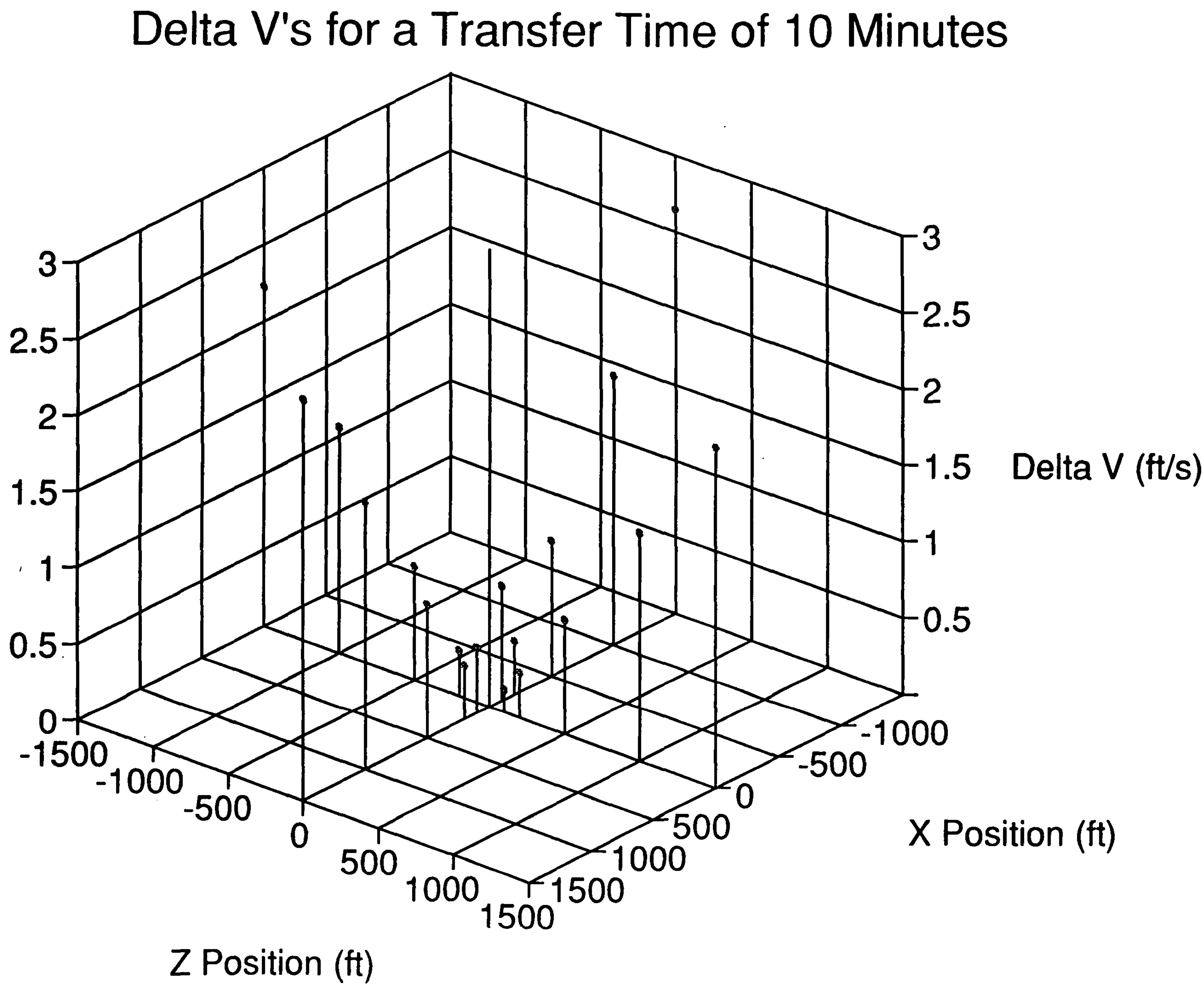
Velocities vs Time
.00002
.000015
.00001
.000005
0
-.000005
-.00001
-.000015
-.00002
Velocity Components in Ft/s
0
10
20
30
40
50
60
70
80
90
100
110
Time (minutes)

Title: CW State Variiables

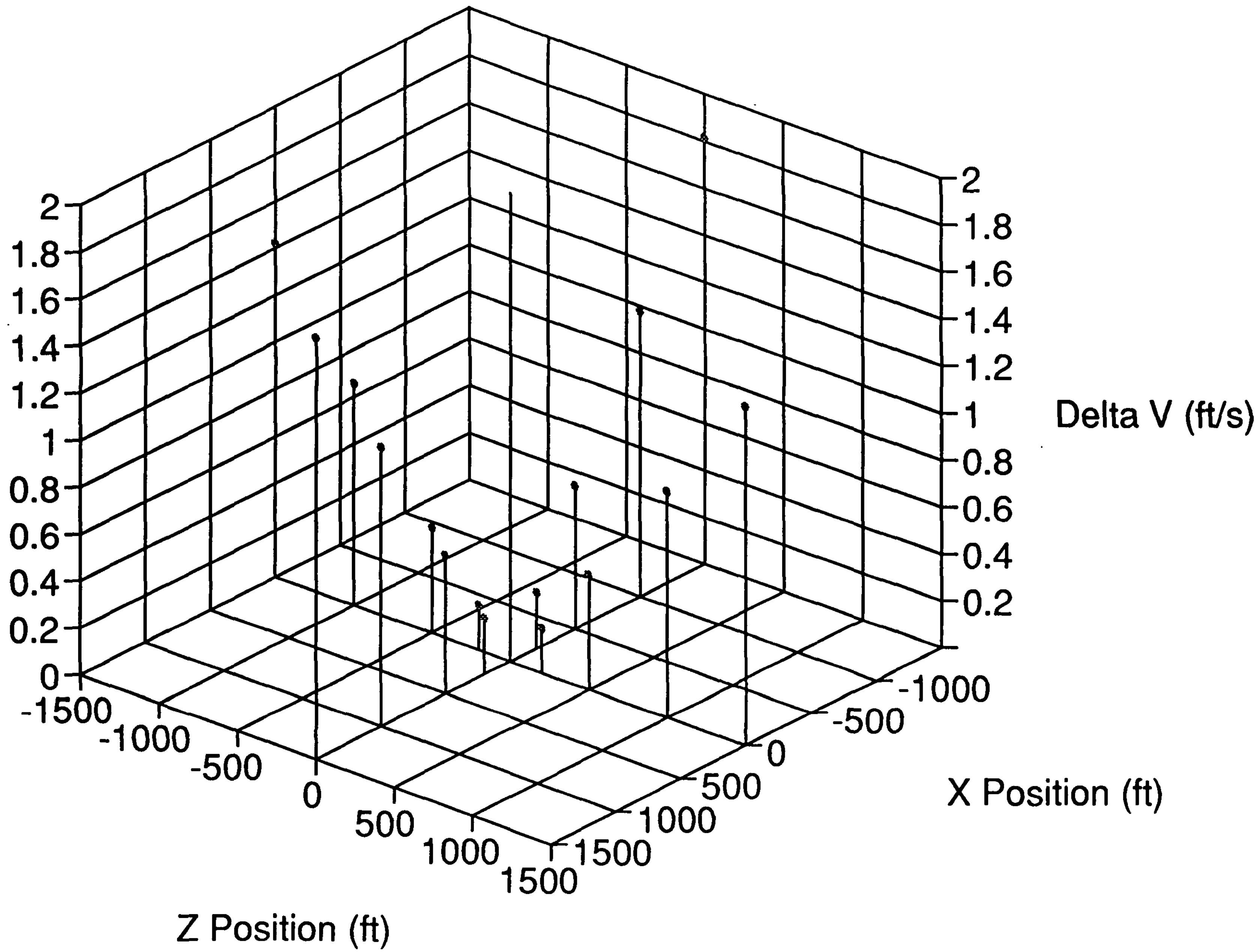
Element	t	x	y	z	vx	vy	vz
1	0	.5	0	0	0	-1E-5	-1E-5
2	4	.5	-2E-6	-2E-6	-5E-6	-1E-5	-1E-5
3	8	.5	-5E-6	-5E-6	-1E-5	-9E-6	-9E-6
4	12	.49999	-7E-6	-7E-6	-1E-5	-7E-6	-7E-6
5	16	.49999	-8E-6	-8E-6	-2E-5	-5E-6	-5E-6
6	20	.49999	-9E-6	-9E-6	-2E-5	-3E-6	-3E-6
7	24	.49998	-9E-6	-9E-6	-2E-5	-5E-7	-5E-7
8	28	.49998	-9E-6	-9E-6	-2E-5	2E-6	2E-6
9	32	.49997	-9E-6	-9E-6	-2E-5	4.4E-6	4.4E-6
10	36	.49997	-7E-6	-7E-6	-2E-5	6.5E-6	6.5E-6
11	40	.49997	-5E-6	-5E-6	-1E-5	8.2E-6	8.2E-6
12	44	.49996	-3E-6	-3E-6	-7E-6	9.4E-6	9.4E-6
13	48	.49996	-1E-6	-1E-6	-2E-6	9.9E-6	9.9E-6
14	52	.49996	1.4E-6	1.4E-6	3E-6	9.9E-6	9.9E-6
15	56	.49996	3.7E-6	3.7E-6	7.8E-6	9.2E-6	9.2E-6
16	60	.49997	5.8E-6	5.8E-6	1.2E-5	7.9E-6	7.9E-6
17	64	.49997	7.5E-6	7.5E-6	1.6E-5	6.2E-6	6.2E-6
18	68	.49997	8.7E-6	8.7E-6	1.8E-5	4E-6	4E-6
19	72	.49998	9.4E-6	9.4E-6	.00002	1.6E-6	1.6E-6
20	76	.49998	9.4E-6	9.4E-6	.00002	-1E-6	-1E-6
21	80	.49999	8.9E-6	8.9E-6	1.9E-5	-3E-6	-3E-6
22	84	.49999	7.8E-6	7.8E-6	1.6E-5	-6E-6	-6E-6
23	88	.5	6.2E-6	6.2E-6	1.3E-5	-8E-6	-8E-6
24	92	.5	4.2E-6	4.2E-6	8.9E-6	-9E-6	-9E-6
25	96	.5	2E-6	2E-6	4.2E-6	-1E-5	-1E-5
26	100	.5	-4E-7	-4E-7	-9E-7	-1E-5	-1E-5

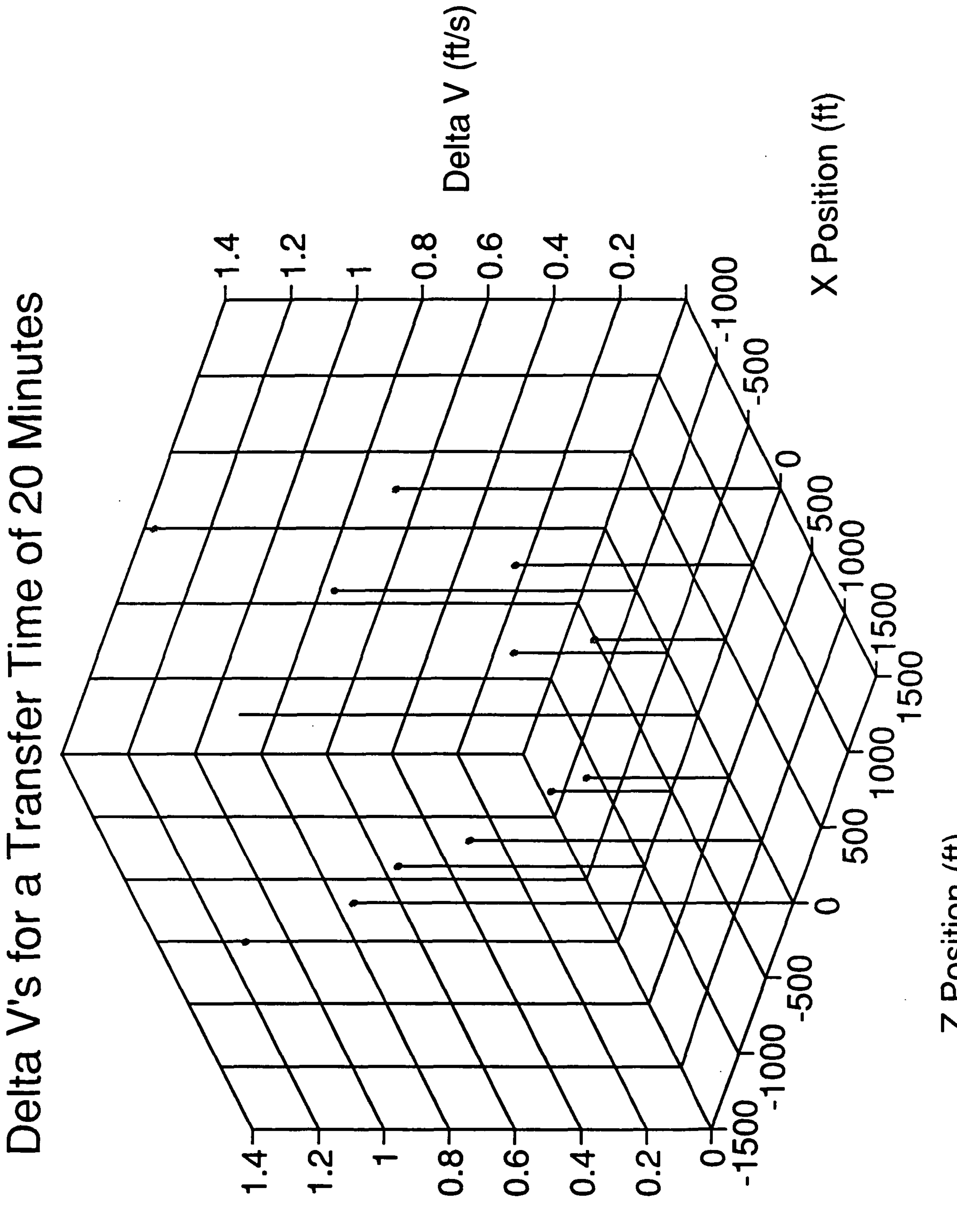
Delta V's for a Transfer Time of 5 Minutes




Delta V's for a Transfer Time of 10 Minutes
Delta V (ft/s)
0
0.5
1
1.5
2
2.5
3
X Position (ft)
-1000
-500
0
500
1000
1500
Z Position (ft)
-1500
-1000
-500
0
500
1000
1500

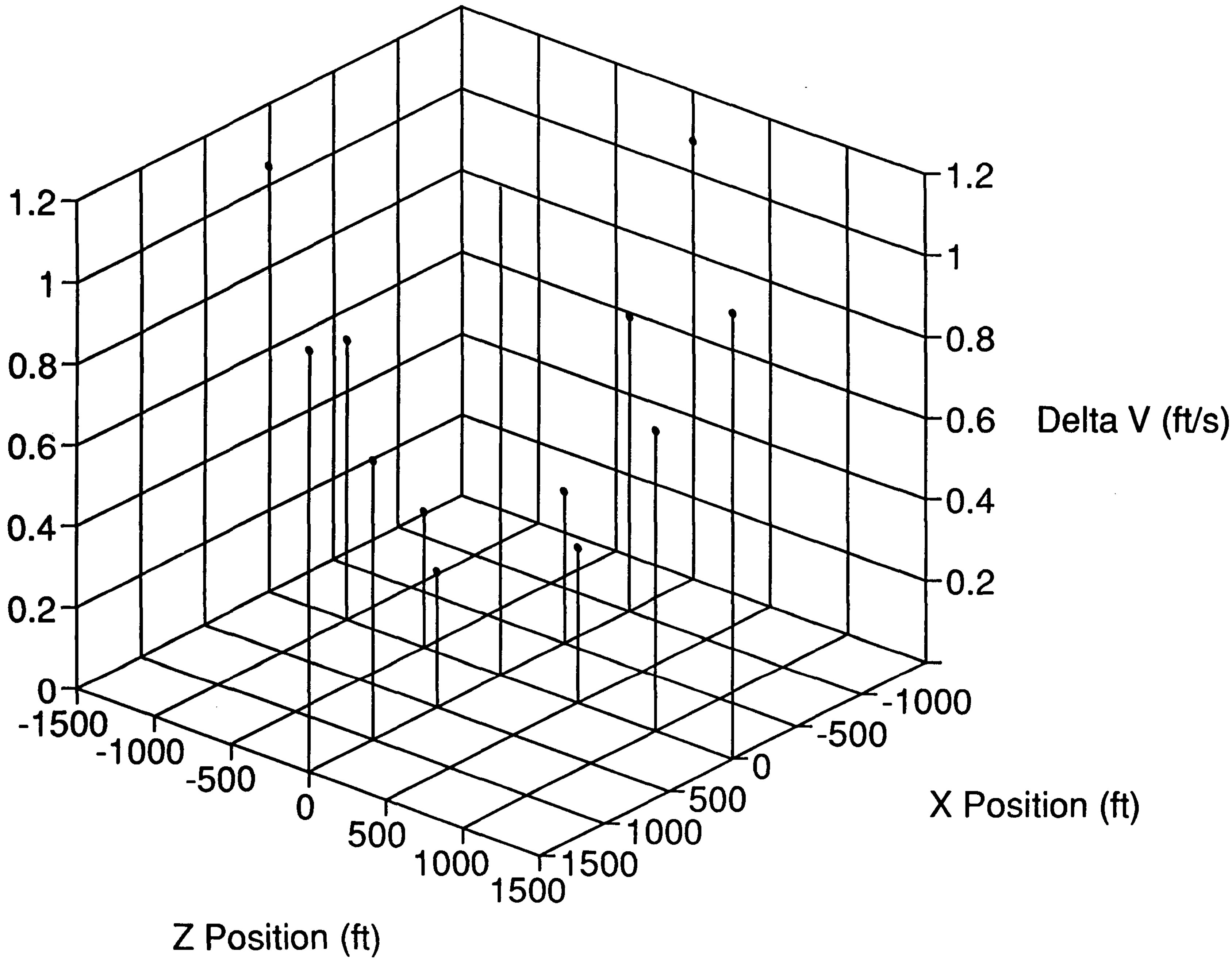
Delta V's for a Transfer Time of 15 Minutes



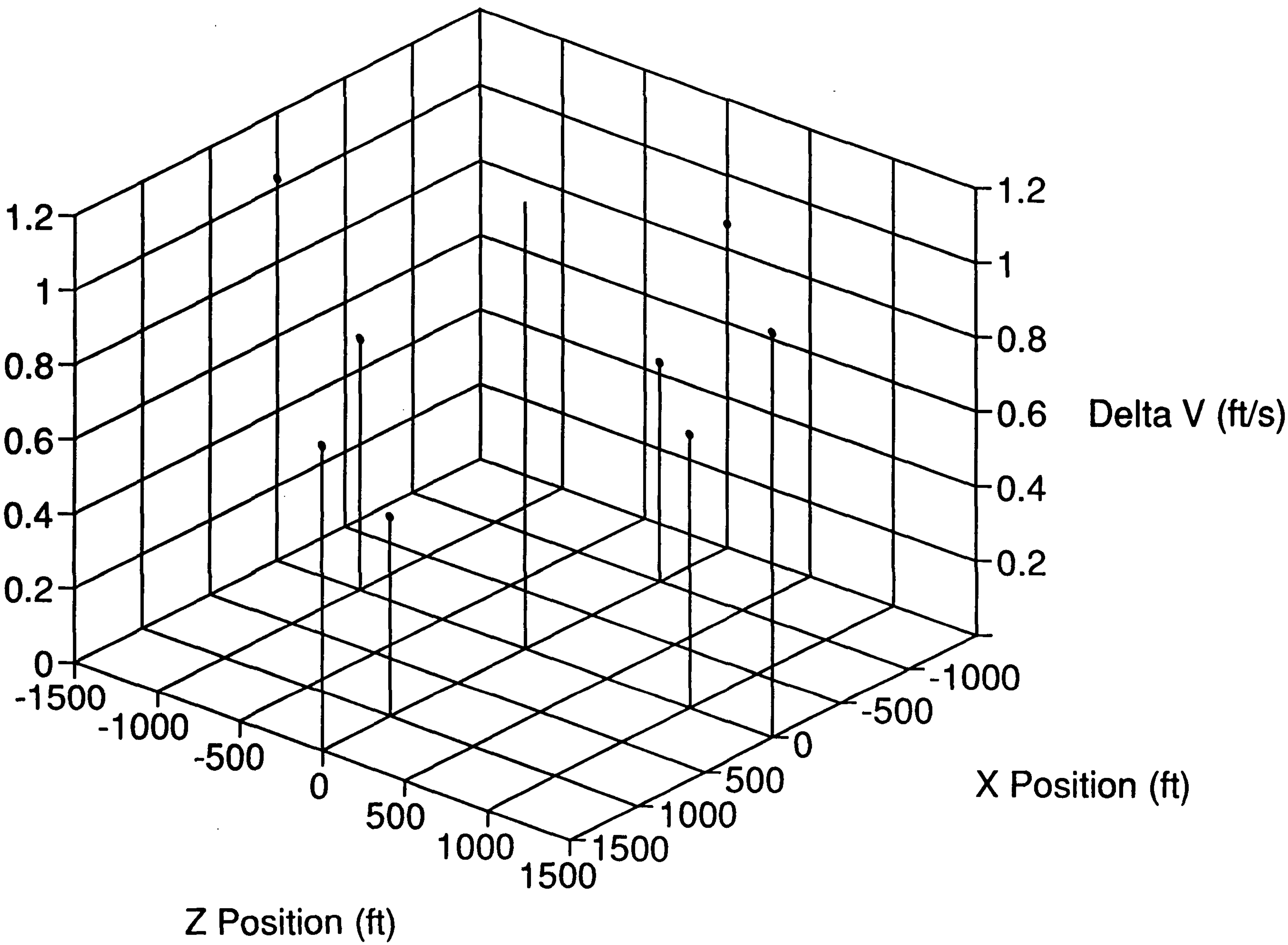


Delta V's for a Transfer Time of 20 Minutes
Delta V (ft/s)
1.4
1.2
1
0.8
0.6
0.4
0.2
0
X Position (ft)
-1000
-500
0
500
1000
1500
Z Position (ft)
-1500
-1000
-500
0
500
1000
1500

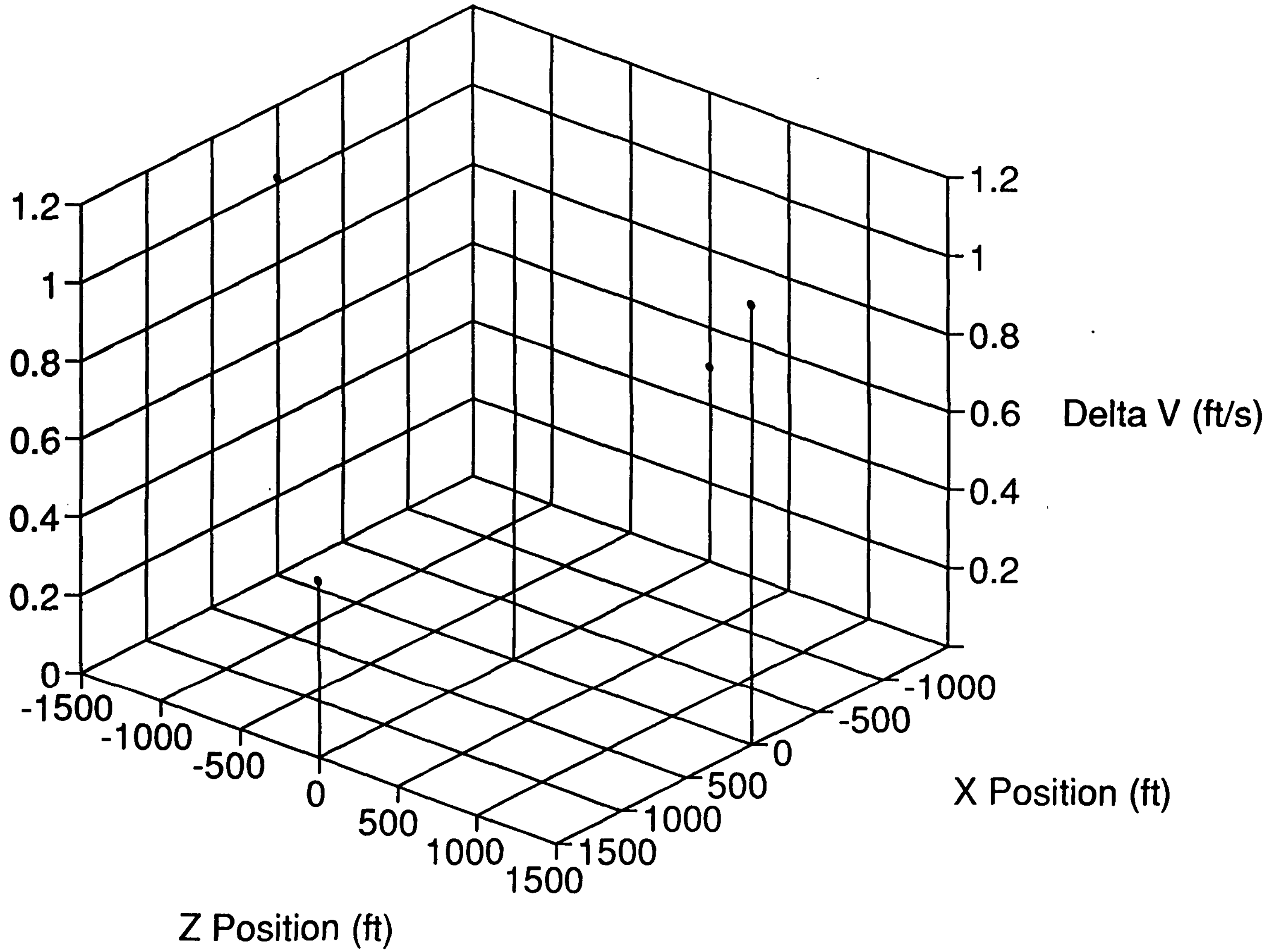
Delta V's for a Transfer Time of 25 Minutes



Delta V's for a Transfer Time of 30 Minutes



Delta V's for a Transfer Time of 45 Minutes



Delta V's for a Transfer Time of 60 Minutes