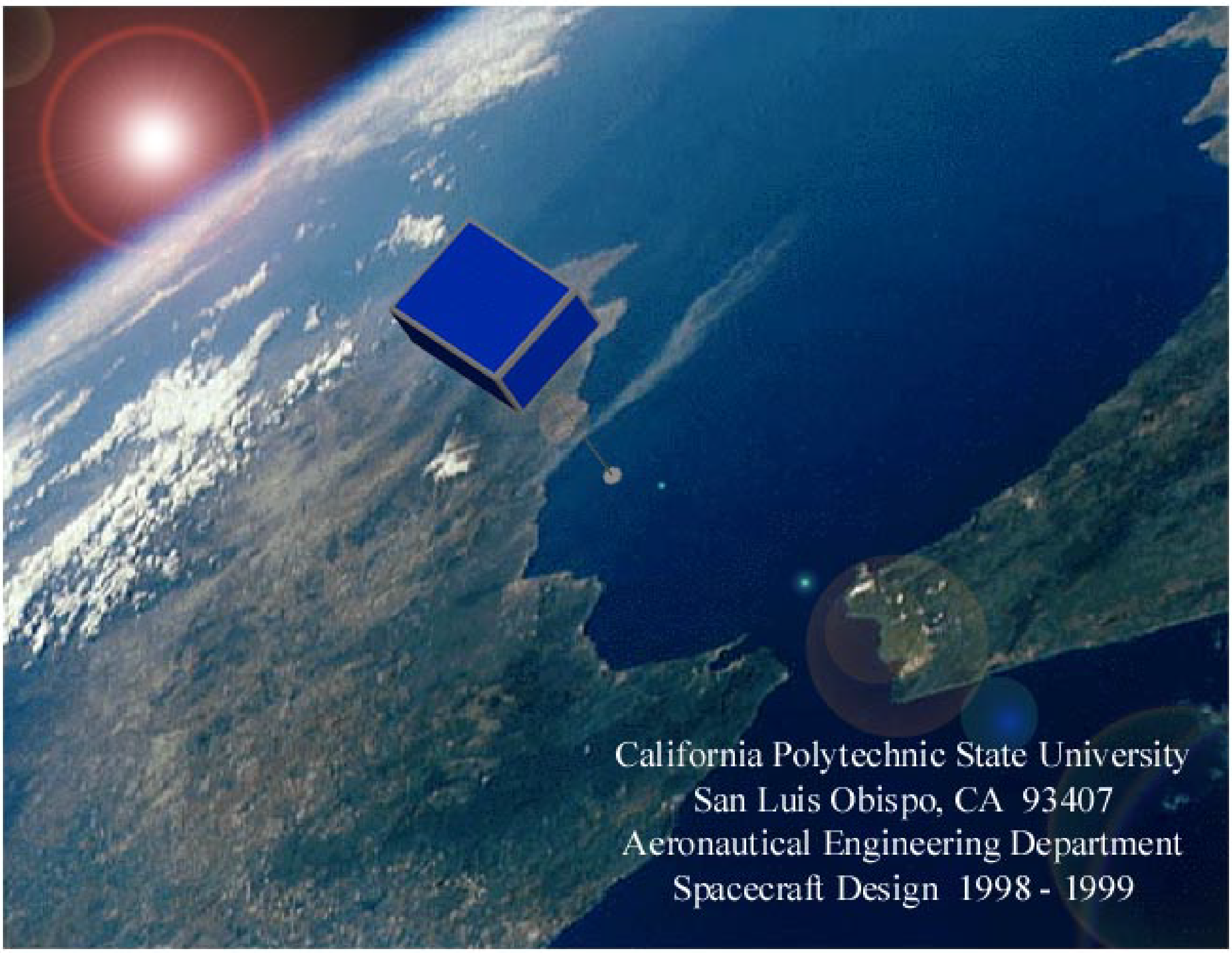
Celestial Concepts

presents

LOCO-1

A multi-mission, commercial micro-satellite



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NOMENCLATURE

Symbol	Name
A	Total cross sectional area
A_s	Surface area (m^2)
C	Circumference (m)
C_d	Coefficient of drag
Cr	Required battery capacity per battery (W-hr)
D	Residual dipole of the satellite (A-m^2)
DOD	Depth-of-discharge limit (%)
E	Young’s Modulus
Eb/No	Energy per bit noise density ratio
EIRP	Equivalent Isotropic Radiated Power (dBW)
F_s	Solar constant (1358 W/m^2)
Gpt	Peak transmit antenna gain (dB)
Gr	Receive antenna gain (dB)
Grp	Peak receive antenna gain (dB)
G_s	Solar constant (1358 W/m^2)
Gt	Transmit antenna gain (dB)
I_d	Inherent degradation
K_a	Factor that counts for incoming solar energy
L	Length (m)
La	Propagation loss (dB)
Ll	Line loss (dB)
Ls	Space loss (dB)
L_{pr}	Receive antenna pointing loss (dB)
L_θ	Transmit antenna pointing loss (dB)
M	Magnetic moment of the Earth (7.96 x 10^{15} tesla-m^3)
N	Number of batteries
P	Transmitter power (Watts)
Pe	Average eclipse load (W)
P_{eq}	load
Q_w	Electrical power dissipation
R	Height from the center of the Earth to the satellite (m)
S	Path length (m)
T_a	Aerodynamic torque (N-m)
Te	Maximum eclipse time (hr)
T_g	Gravity gradient torque (N-m)
T_m	Magnetic torque (N-m)
Ts	System noise temperature (K)
T_{sp}	Solar radiation torque (N-m)
V	Satellite velocity (m/s)
a	Albedo (122 W/m^2)
b	Width
c	Speed of light (m/s)
c_{pa}	Center of aerodynamic pressure (m)
c_{ps}	Center of solar pressure (m)
c_g	Center of gravity (m)
e	Error fraction of beam-width (%)
f	Frequency (Hz)
k	Spring constant
q	Reflectance factor

q_1	Earth IR emission (237 W/m^2)
r	Angular radius of earth
t	Thickness
α	Solar absoptivity
ε	IR emissivity
η_T	Transmission efficiency
η_s	Solar cell efficiency
λ	Wavelength (m)
μ	Earth's gravitational constant (3.986 x 10^{14} m^3/s^2)
ν	Poission's Ratio
θ	Sun incidence angle. The angle between the normal to the surface of the array and the sun line.
θr	Receive antenna beam width
ρ	Atmospheric density (kg/m^3)
σ	Stefan-Boltzman (5.67e-8 W/m^{2}k^4)
σ_{cr}	Critical stress

1. INTRODUCTION

Celestial Concepts presents the LOCO-1: a low-cost micro satellite with multi-mission capabilities. The satellite can be used for technology demonstration, earth observation, or store and forward communications in a LEO environment. The satellite was designed to be able to compete in today's commercial satellite market.

Today's satellite market has shown a tremendous increase in the use of micro and mini satellites. Due to the commercialization of space, there has been a greater demand for access to space. Since these smaller satellites are rapidly produced and are low in cost, payloads have to weigh less and pay less. LOCO-1 will take advantage of this market by incorporating a 50 kg bus with high performance system components and easy payload integration. The satellite is designed specifically for technology demonstration but can carry a variety of other payloads.

The content of this report covers all aspects of satellite design. From market analysis to mission analysis and subsystem design to business objectives, this report covers more than just technical information. As a company, Celestial Concepts is compelled to sell a commercially marketable satellite and be successful.

2. MARKET ANALYSIS

The design project was based on the idea that it would be marketable. We began the year with nothing more than a general intention to start a profitable satellite manufacturing company. With this in mind we began a wide base survey of what kinds of satellites have been launched. These launch trends were then narrowed down to highlight just the small satellites from the year 1993 through 1998. The results were compiled and presented in graphical format to identify the potential of two markets (see **Figure 2.1**). This compilation allowed us to understand the trends of these two niche markets and identify the main competitors in each.

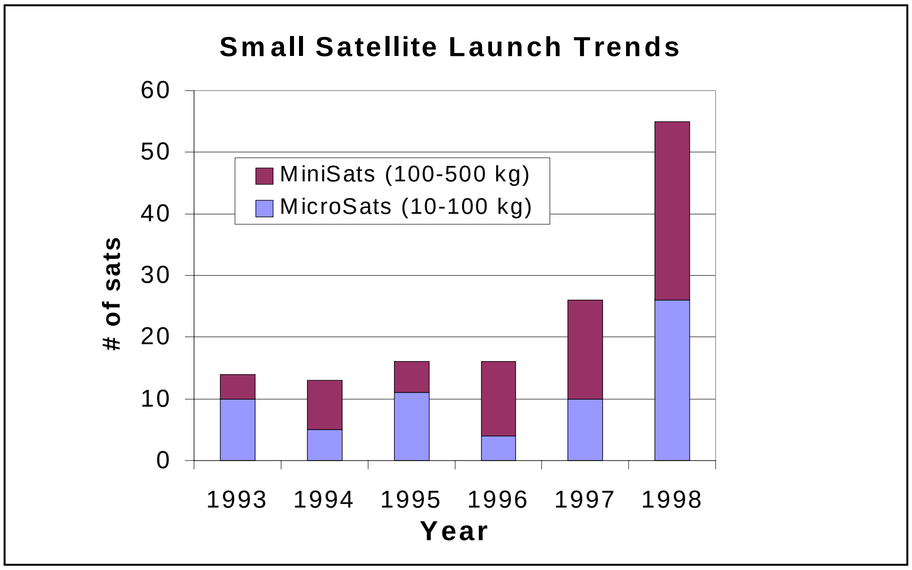


Figure 2.1: Market Trends

Once the markets and competitors were identified, the potential missions of each were studied to determine what customers needs were. Typical missions for mini satellites (100-500kg) included communications, interplanetary, technology tester, and military missions. Communication could be real time as well as store-and-forward. Interplanetary missions could fulfill any number of requirements including scientific experiments to be conducted in other planet's atmospheres or observe naturally occurring cosmic events. Technology testing would allow for new unproven technology to be space rated. For military purposes, LOCO-1 could be used as an observation satellite.

The micro satellite's (10-100kg) possible missions also include those of the mini-satellites such as technology tester and communications. As a result of the micro satellite's limited capabilities, communications would usually be through store-and-forward rather than real-time. Testing new or improved technology seems to be best suited for micro satellites. This does require a more stringent set of procedures to avoid unauthorized transfer of technology. Micro-satellites could also be used for completing limited scientific mission where on board payload instruments could store their data to be forwarded at a later time. Micro satellites could fulfill the need for small-scale start up space programs. Essentially it could be used as a low cost stepping stone leading to further complex satellite manufacturing for emerging companies, universities, or countries.

With first hand information, the decision was made to enter the micro satellite market. This market has easier entry compared to the mini satellite market due to the expected growth potential in the near future. In addition this market is not heavily influenced from larger satellite manufactures. An upstart company, can open up business with relatively low investments needs. The affordability of the product will open up a previously untapped domestic market and provide the customer with a lower overall financial investment risk if failure occurs.

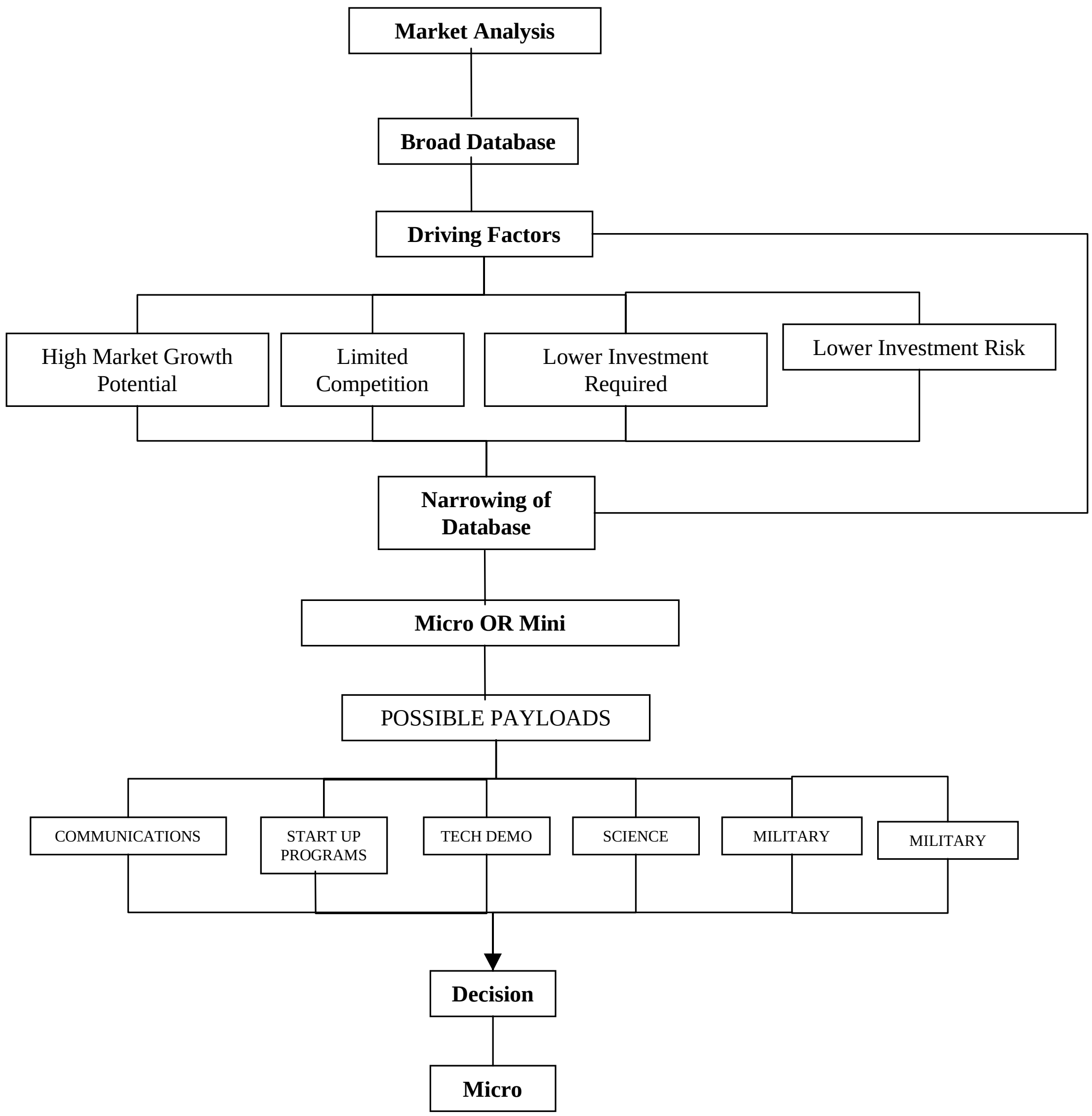


Figure 2.2: Mission Decision Process

3. LAUNCH VEHICLES

3.1 Introduction

The launch vehicle is a key factor in the determination of the sizing of LOCO-1. Since LOCO-1 is to be small and commercial, it needs to ride on the most cost-effective launcher or launchers possible. The launcher determines the structure and configuration of LOCO-1. Primary design drivers include reliability, cost and the frequency of launches available.

3.2 Launch Vehicle Sizing

Flying as a secondary or piggyback payload would significantly reduce the launch cost to the customer. Initially, the Ariane was the primary launch vehicle because it is a proven flight vehicle and has an established secondary launch program. The *Ariane Structure for Auxiliary Payloads* (ASAP) is a circular platform that sits around the main payload adapter. (See **Figure 3.1: ASAP Ring Location**)

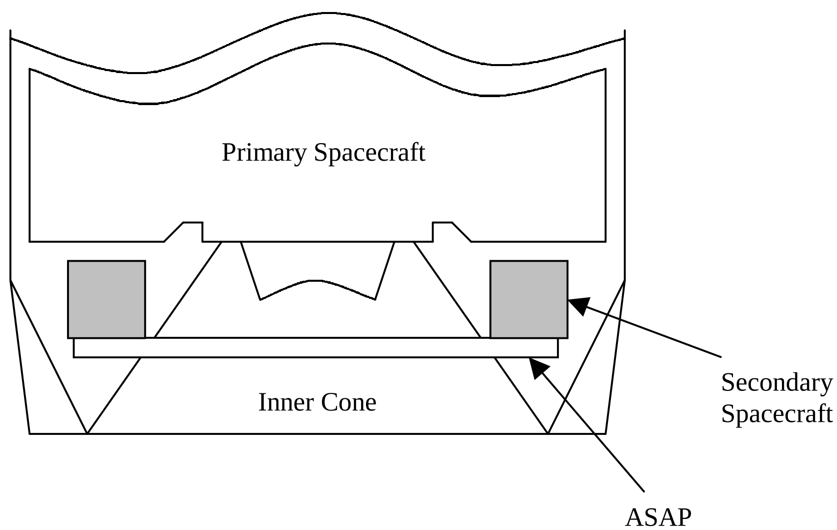


Figure 3.1: ASAP Ring Location

However, as of March 15, 1999, all commercial communication satellites are considered U.S. munitions and therefore require an export license. This involves an application and a review on a case-by-case basis for all commercial satellites travelling overseas. This causes delays and requires added security during launcher integration.

Because of this restriction, the Delta II became the primary launcher for the LOCO-1. It has a proven flight history and also accommodates secondary payloads. A supplied Delta II Secondary Payload Planner's Guide provides all the necessary requirements and restrictions for launching as a secondary payload. Flying on the Delta II is also convenient since one of the primary launch sites is located only one hour away by car at Vandenberg AFB.

3.3 *Delta II Launcher*

3.3.1 Payload Envelope and Satellite Interface

Secondary Payloads on the Delta II are attached to the side of the vehicle's second stage. (See **Appendix A: Technical Drawings**, LOCO SAT, for figure) The payload envelope is approximately 28 inches by 24 inches by 12 inches with the tightest constraint, 12 inches, being the length from the second stage to the fairing. Delta II provides a trickle charge of current to the secondary batteries of the spacecraft while on the launch pad. Delta II also provides a Payload Attach Fitting (PAF) on the vehicle with a nine-inch diameter ring. The spacecraft interfaces with a Payload Adapter Assembly (See **Appendix A: Technical Drawings**, Ring Attachment, for figure) which is then fastened to the PAF using a marmon clamp and secured using two bolts. During separation, two bolt cutters sever the studs and four springs provide a separation velocity of about 2 to 8 ft/sec. The clamp design allows for spacecraft separation even if only one bolt is severed.

3.3.2 Launch Loads Design and Qualification

The Delta II Secondary Payload Planner's Guide gives launch load environments and the qualifications required for launch approval. These launch loads were used as the preliminary and primary design requirements for the LOCO-1. The spacecraft structure has to withstand 10 g loading in all axes simultaneously and have coupled modal frequencies greater than 35 Hz.

Qualification for flight requires a structural test or an acceptable theoretical analysis. The structural test requires survivability up to 1.25 times the maximum flight loads from past flight data. A theoretical analysis for qualification requires a factor of 2.0 times maximum flight loads. Sinusoidal vibration qualification requires the spacecraft to survive a factor of 1.4 times flight levels. Details for stage separation shocks as well as acoustic and thermal environments are also available in the Planner's Guide.

3.4 *Alternative Launchers*

Since LOCO-1 is designed to be a rapid production bus, providing a means to get to orbit quickly is one of the primary concerns. Though Delta II is the primary launch vehicle, LOCO-1 needed to be able to fly on other launchers to provide the customer with an option of other types of orbits and launch opportunities. LOCO-1 can fly on the following launchers as a secondary payload:

Launcher	# LOCO-1s per launch	Launch Site
Delta II	1 or 2	Vandenberg AFB, CA
		Kennedy Space Center, FL
Ariane 5	1 / ASAP spot	Kourou, French Guiana
Pegasus XL	2	Vandenberg AFB, CA
		Kennedy Space Center, FL
		Wallops Flight Facility,VA
		Alcantara, Brazil
Taurus	Up to 12	Vandenberg AFB, CA
		Kennedy Space Center, FL

Table 3.1: Available Launchers for LOCO-1

3.5 Secondary Payload Limitations

The secondary payload of any flight can not affect the orbit that it will be placed in. Since most of LOCO-1's flights will be for technology demonstration or microgravity experiments the orbit plays a minor role. For any customers that require a particular orbit, LOCO-1 can fly on other launchers, pay a higher launch cost to fly as a primary, or wait for a primary payload with a closely matching orbit. The secondary payload can not present any kind of hazard to the primary payload or interfere with the primary payload in any way.

4. SYSTEMS ENGINEERING

4.1 Introduction

Although it was uncertain in the beginning on how the whole system was going to look and how well it would function together, we approached it by breaking it up into manageable segments. Each of the segments was assigned a lead to assess that particular system's concepts and in turn produce specific requirements. The primary purposes for our group was to work together and collectively develop the integration and configuration of all the subsystems compromising LOCO-1. This section of the report will summarize overall approach in accomplishing this task.

4.2 Configuration Development

The process of LOCO-1's physical configuration was debated and at this stage we wanted to review as many concepts as possible, regardless if its merit. Time was spent reviewing and qualitatively analyzing varying spacecraft exterior geometric configurations. The advantages and disadvantages were weighed against each other for each concept. This method of review allowed the narrowing down of certain concepts. Driving factors for this stage of development were the launcher's secondary payload ('piggy back') sizing, subsystem component size, and structural layout complexity.

The search for launch vehicles, which would allow for a low overall mission cost allowed the group to come upon certain trade-off characteristics of each concept design configuration. This trade-off analysis dictated our initial and final generation's spacecraft configuration design.

4.3 Systems Engineering

Without the desirable amount of people power to devote to this task, team members had to take on further responsibility and share in the duties required of a systems engineer's role. While every member's sole requirement was to research their own respective subsystem, little time was spent (at least initially) acquiring an overall understanding on how all the systems were going to interact together.

Throughout the first quarter of LOCO 1's design, keeping all members up to date with the requirements of the independent components (initially considered as such) did not seem a high priority. Each members responsibility was to fully research, developed, and analyzed their subsystem, then present status reports to the rest of the group during in-depth review sessions. As the amount of information gradually increased, this relatively casual style of communication did not appear so well suited. The group needed to be able to communicate back and forth with one another in a manner, which allowed for the latest developments to be known by all, not just that particular system lead.

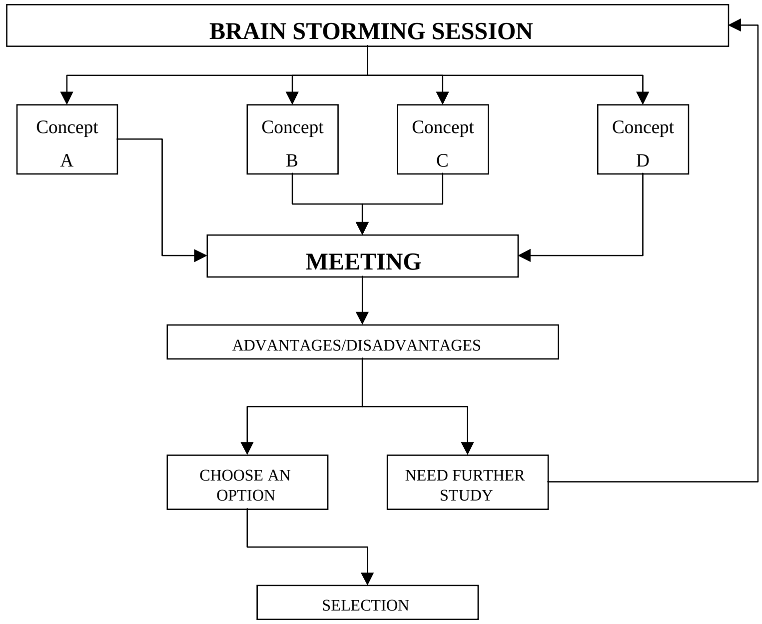


Figure 4.1: Decision-Making Process

4.4 Systems Management System

Searching for an alternative, rather than designating one group member as systems engineer, the group came upon a reasonable solution of handling the systems management. All further changes to the spacecraft configuration and/or subsystems components were reviewed by that subsystem's lead, then posted on a satellite interface wall. This wall's purpose was to keep the rest of the group members informed of the pending alterations of other's subsystems, which possibly required changes on their own subsystem.

4.5 Design Trade-Off

Design trade-offs could be made by analyzing the adverse effects, if any, one design change would have on the rest. We used the trade-off analysis as our primary tool in further developing and refining the overall design process. These trade studies allowed us to consider all available options along with being able to understand the constraints. One example of such a trade analysis came when a momentum wheel was being considered as a cost-effective means of attitude determination and control. It appeared to be the appropriate choice in LOCO-1's cost savings initiative until it was learned of the power requirements needed to operate the momentum wheel. In order to supply the required power, LOCO-1 would need to have larger solar arrays, requiring deployables. The deployables and the increase amounts of already expensive double-junction solar cells dramatically increased the cost of the satellite, not to mention add unwanted complexity to the design.

5. Attitude Determination and Control

5.1 Introduction

The attitude determination and control subsystem for LOCO-1 can achieve accurate pointing and attitude knowledge at a low cost. Designing such a system for low cost means simplicity and minimal redundancy. However, only so many so many options are available when designing such a system.

LOCO-1 is stabilized by gravity-gradient control. This type of control requires no moving actuators (i.e. momentum or reaction wheels, control moment gyros, etc.), only a boom and a tip mass. With gravity-gradient control LOCO-1 can achieve a pointing accuracy within $\pm 5^o$. Further control is accomplished with magnetic torque rods, which can be used for damping and reorienting the satellite.

The attitude determination system for LOCO-1 consists of an earth sensor and a three-axis magnetometer. The earth sensor provides pitch and roll measurements within $\pm 0.06^o$, whereas the three-axis magnetometer can provide pitch, roll, and yaw measurements within $\pm 1^o$. By having two sensors LOCO-1 can have both a primary and a backup sensor so that attitude knowledge can be known at all times.

5.2 Orientation

The orientation of LOCO-1 can be described by its moments of inertia (MOI). Calculation of the satellite's MOI involved calculating the MOI for each internal and external component relative to the center of mass of the entire satellite. Through careful arrangement of the satellite components, the center of mass lies very close to the boom axis. The matrices describing the MOI before and after deployment as well as the centers of mass for each condition are in **Table 5.1**. Complete analyses of the MOI are shown in Appendix B. Notice that the diagonal elements of the matrices are very small. This means that the angles of rotation about the principle axes are so small ($<0.01^o$) that the satellite wants to naturally stay oriented about its principle axes.

1.8542	-7.9937E-05	-2.1152E-05
-7.9937E-05	0.8695	-5.3568E-06
-2.1152E-05	-5.3568E-06	1.4937

I (kg-m^2)

CM_x	0.1686
CM_y	0.3169
CM_z	0.2322

CM (m)

Before Deployment

1.8571	-7.9937E-05	-2.1152E-05
-7.9937E-05	53.3363	-5.3568E-06
-2.1152E-05	-5.3568E-06	53.9607

I (kg-m^2)

CM_x	0.5202
CM_y	0.3169
CM_z	0.2322

CM (m)

After deployment

Table 5.1: Moments of Inertia of LOCO-1 Before and After Deployment

Figure 5.1 is a picture of two orientations. Notice that once LOCO-1 has detached from launcher fairing it will need to go from the first orientation to the second (see **Figure 5.1**). The second orientation is unstable unless the boom is deployed. Therefore there is a boom deployment sequence in which some control maneuvers are to be performed. **Figure 5.2** shows the sequence that LOCO-1 must perform in order to get into its final orientation. The magnetic torque rods will generate an internal torque that will cause the satellite to move in an oscillatory motion about the Y-axis. The boom deploys at some point during the oscillation, depending on the rate at which the satellite is rotating. Once the boom deploys and the satellite is in the upright position as shown (middle of **Figure 5.2**) it will oscillate about the X-axis. Using the magnetic torque rods**,** the oscillatory motion will dampen out until the satellite reaches its final orientation.

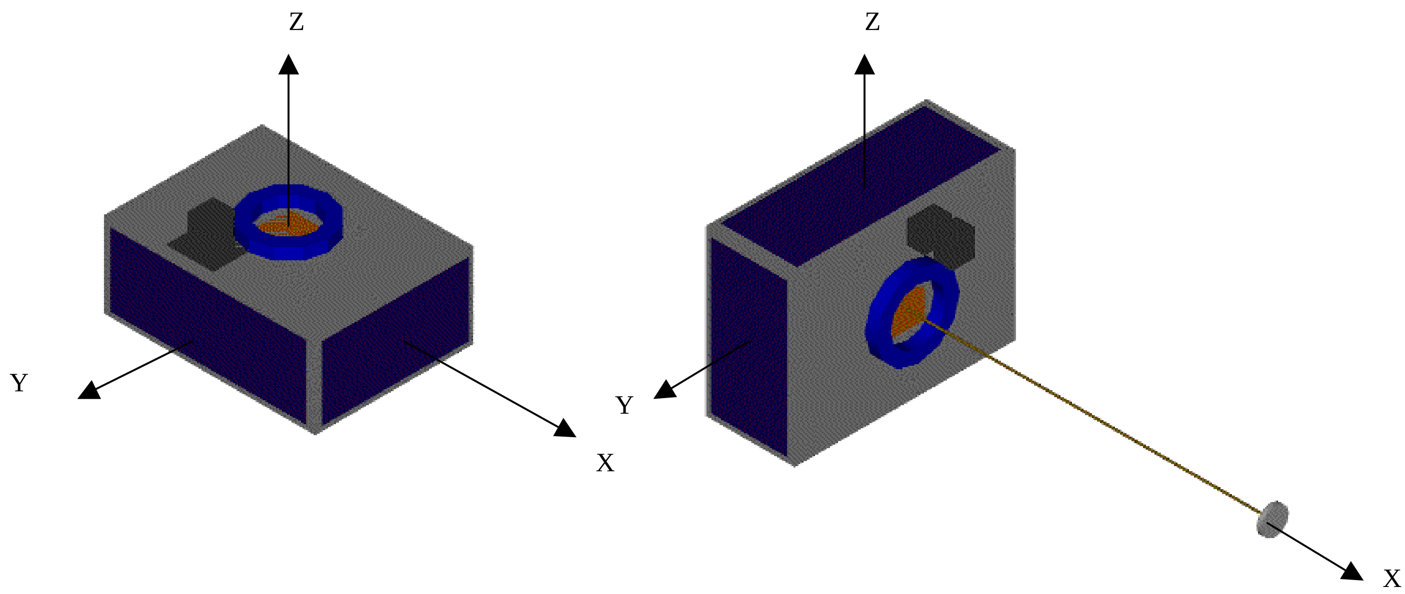


Figure 5.1: LOCO-1's stable orientations (X = nadir, Y = direction of flight, Z = orbit normal)

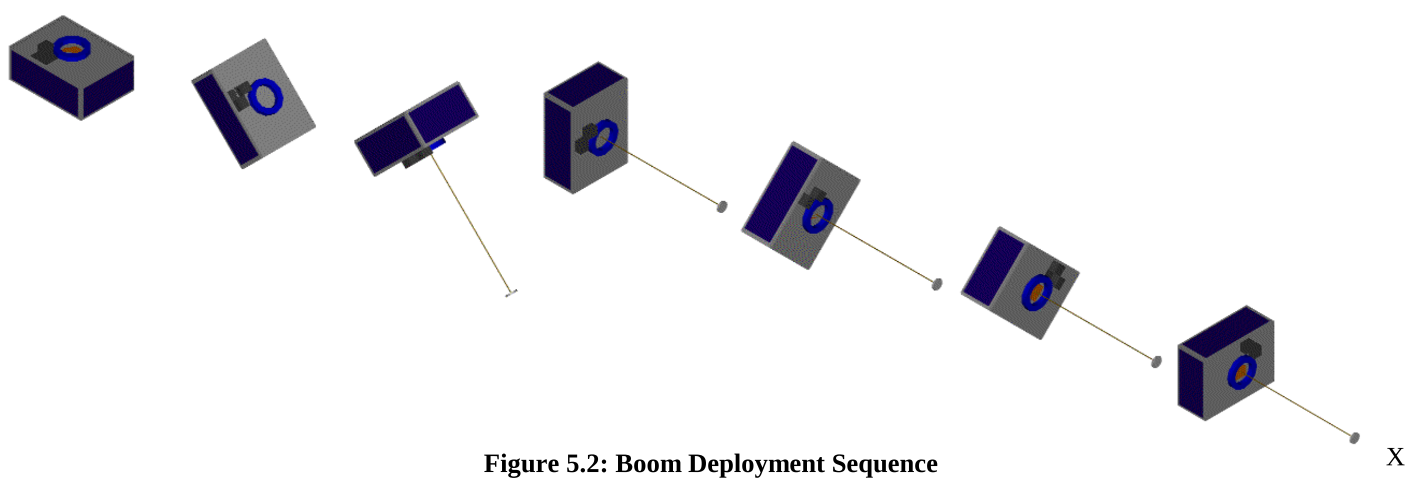


Figure 5.2: Boom Deployment Sequence

5.3 Gravity-Gradient Control

Gravity-gradient torque can be a disturbance torque or a control torque, depending on the application. The worst-case gravity-gradient torque for a satellite in orbit is:

$$T_g = \frac{3\mu}{2r^3}\left|I_3 - I_1\right|\sin 2\theta \qquad \text{[Eqn. 5.1]}$$

where r is the distance from the center of the Earth to the satellite, μ is a gravitational constant, I is the moment of inertia about the respective axis, and θ is the maximum deviation of the Z axis from the local vertical. This equation is derived from Newton's Law of Universal Gravitation, where the attraction between two bodies varies with $1/r^2$. Basically, gravitational attraction decreases as the distance r increases. Thus, a satellite in space will have a greater attraction on its lower side than its upper side. The difference in attraction results in the axis of minimum inertia (the long axis) aligning with the radius of the Earth. This torque can be effective if the satellite is long and thin, which explains the need for a long boom. Therefore stability is achieved by designing a boom and tip mass feature for the satellite that takes advantage of the gravity-gradient torque.

Many considerations need to be taken into account when designing a boom length and tip mass. The boom length and tip mass for LOCO-1 were defined by the minimum angle, θ, at which the worst-case aerodynamic torque was cancelled out by the worst-case gravity-gradient torque. For LOCO-1, the resultant boom length is 3.949 m with a tip mass of 3.510 kg. The equilibrium angle, θ, is 0.0001^{o}.

5.4 Magnetic Control

In order to control the satellite during detumbling and the deployment sequence, and to dampen out oscillations due to external torques, electromagnetic rods will be used. LOCO-1 will consist of three TORQRODS®. Ithaco Space Systems, who have an excellent record in the design and manufacturing of attitude determination and control devices, manufacture these rods. Each rod is aligned with one of the principle axes. The total dipole produced by each rod is 1 A-m^2. The total magnetic torque the rods exert on the spacecraft is 5.1122 x 10^{-5} N-m (at a 400-km altitude). This is more than enough torque to counteract any of the worst-case disturbance torques, as well as guarantee reorientation of the satellite. To calculate the magnetic torque the rods exert on the satellite the following equation was used:

$$T_m = \frac{2M}{r^3} D \qquad \text{[Eqn. 5.2]}$$

where D is the total dipole of the torque rod, M is the magnetic moment of the Earth (M = $7.96\text{x}10^{-15}$ tesla-m^3), and r is the distance from the center of the Earth to the satellite. Like the gravity gradient torque, the magnetic torque loses effectiveness as the orbit altitude increases (as r increases, T_m decreases).

5.5 Three- Axis Magnetometer

Ithaco Space System's three-axis magnetometer will act as the primary attitude determination sensor during the detumbling and deployment phase. The sensor will provide pitch, roll, and yaw measurements within $\pm 1^{o}$ of accuracy. The magnetometer works by taking vector measurements of the Earth's magnetic field. Once the measurements are taken, the data is compared to a fixed model of the Earth's magnetic field so that the orientation of the satellite can be determined. Since the magnetometer is sensitive to magnetic fields, the sensor will have to turn off when the magnetic torque rods are used. Once boom deployment and a stable orientation has been reached the magnetometer will switch to backup mode where it will serve as a backup for pitch and roll data for the earth sensor. Yaw measurements will continue to be measured primarily by the magnetometer.

5.6 Earth Horizon Sensor

The primary attitude determination device is the Mini Dual Earth Sensor (MiDES) which is manufactured by Servo Corporation of America. Due to its low power consumption and small size, the MiDES is the perfect sensor for LOCO-1. Once LOCO-1 has deployed its boom and reached its final stable orientation, the MiDES takes over all primary pitch and roll measurements from the three-axis magnetometer. The magnetometer will then act as a backup for pitch and roll information, and give primary yaw data. Performance of the MiDES allows attitude knowledge within $\pm 0.06^{o}$. The sensor works by detecting the temperature difference in the CO_2 band about the horizon of the Earth. The temperatures are read as voltages and then used in an algorithm to calculate the position of the horizon. Two pyroelectric arrays are set 90^{o} apart from each other about the pitch and roll axes. Each sensor covers a range of $\pm 5.5^{o}$ (11^{o} total) about the horizon. The accuracy of the sensor can be calibrated for any altitude.

5.7 Disturbance Torques

There are only a few disturbance torques that have an effect on the stability of LOCO-1. Analysis for this report is based on a 400-km altitude. This altitude is a critical since once below this altitude the Earth's atmosphere becomes apparent and aerodynamic torque will eventually overcome the satellite (and it will burn up). One design parameter for LOCO-1 is to ability to perform at very low altitudes (400-km), since its mission is within LEO. Therefore the effect of aerodynamic torque must be small enough to allow good performance. The following equations were used to calculate worst case disturbance torques on LOCO-1:

Gravity-gradient: $$T_g = \frac{3\mu}{2r^3}\left|I_3 - I_1\right|\sin 2\theta \qquad \text{[Eqn. 5.1]}$$

Aerodynamic: $$T_a = 0.5\rho C_D A V^2 \left(c_{pa} - c_g\right) \qquad \text{[Eqn. 5.3]}$$

Magnetic: $$T_m = \frac{2M}{r^3} D \qquad \text{[Eqn. 5.2]}$$

Solar Radiation: $$T_{sp} = \frac{F_s}{c} A_s (1+q)\left(c_{ps} - c_g\right)\cos i \qquad \text{[Eqn. 5.4]}$$

The values of the disturbance torques (at a 400-km altitude) are:

$T_g = 1.00 \times 10^{-04}$ N-m

$T_a = 3.53 \times 10^{-10}$ N-m

$T_m = 5.11 \times 10^{-05}$ N-m

$T_{sp} = 4.97 \times 10^{-12}$ N-m

6. COMMUNICATIONS

6.1 Introduction

This section will describe the trade-offs associate during the analysis and design of LOCO-1's communication's package. It will give detailed descriptions of what communication requirements were defined and how the final system's components were chosen. Before it is explained, lets review LOCO-1's mission requirements:

(1) Uplink and Downlink signals for communication between the ground station(s) and the spacecraft would be store and forward via the primary download helix antenna, while an onmidirectional monopole antenna will provide back up service if needed.

(2) Uplink communications will be attained with one onmidirectional antenna which will provide the communication link between LOCO-1 and the ground segment. This link must be maintained at all time, all inclinations, and in any orientation LOCO-1 may be in.

(3) The Uplink segment of the communications will require a lower gain antenna (receiving system) due ground station ablity to transmit a higher gain signal.

(4) The Downlink segment requires the utilization of a higher gain antenna (transmitting) mainly due to the lack of onboard power.

(5) Communications transmission rates must be at least 56 kbps or greater (matching competitor capability).

Table 6.1 shows the communications system's components, while **Table 6.2** and **Table 6.3** show the communications system (antennas) characteristics. **Figure 6.1** shows the communication system's overall design process.

Component	Company	Qty.	Part #	Description	Total Part Cost	Total Weight (g)	Time to Deliver
UHF Transmitter	Ayedin Telemetry	1	T-110U	10 Watt Transmitter	$5,600	450	6
UHF Receiver	Ayedin Telemetry	1	RC-103	5.6 Watt Reciever	$5,400	510.3	6
Antenna Swich	DowKey S&S Tech. Inc.	1	401		$90	71	2
Antennas	In-House	3	N/A	(2) Onmi & (1) Helix	$1,000	400	1
Encoder	Ayedin Telemetry	1	MMSC-800	Data Format	$28,000	600	6
Wiring	In-House		1	Wiring System	$1,000	100	neglible

Table 6.1: Communication System Components Description

HPBW	Wavelength	Circumference	Length	Diameter	Gain	Turn Angle	Spacing	# of turns	Weight
degrees	m	m	m	m		degrees	m		kg
90	0.5	0.204275292	1	0.0650228	11.3974	13	0.2249511	4.4454115	0.04

Table 6.2: Helix Antenna Characteristics

HPBW	Wavelength	Length	Gain	Weight
degrees	m	m	dB	kg
180	0.5	0.5	1	0.015

Table 6.3: Onmidirectional Antenna Characteristics

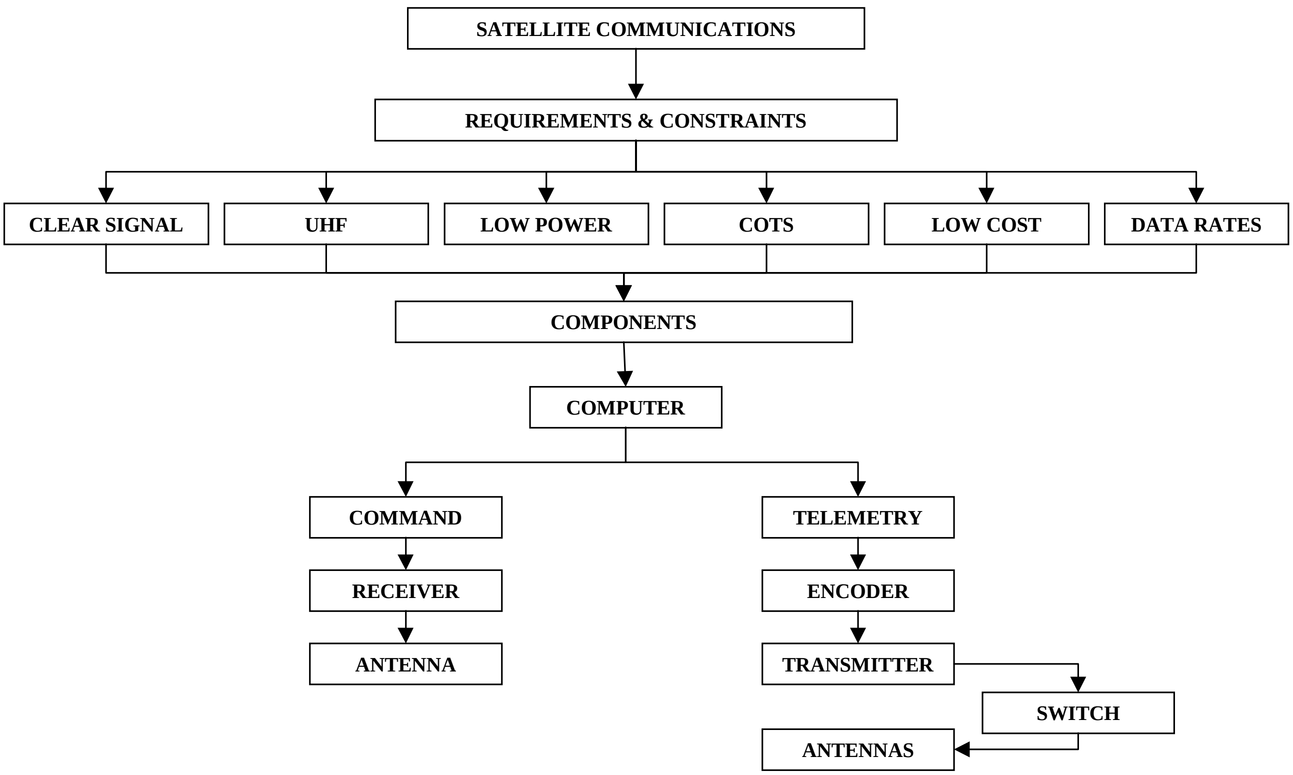


Figure 6.1: Top Level Communications System Design Process

6.2 Frequency Management

The initial task for fulfilling the designing of LOCO-1's communications package was a long and confusing process full of surprise hurdles and roadblocks. Not having a clear path with which to follow made the process of decision making very tedious. Much time was consumed attempting to make sure all aspects involved were covered and not missed. A point well taken at every presentation to industry in which questions to explain the selection of our components or other relevant details, which are required to have a complete analysis of a communication, were asked. Operating at the 'correct' frequencies was a complicated set of issues we were not familiar with, adding to the difficult task of understanding and implementing which frequencies are allocated according to user and purpose of use.

Familiarization came through studying the Federal Communications Commission (FCC) standard format of frequency spectrum allocation. It was learned that communications frequency allocations were split up in forms of certain frequency blocks. Each one of these blocks are capable of allowing multiple frequency slot allocations specifically designated to user's purpose and requirements. There were blocks of frequencies specially designated for amateur radio, fixed (land-based) radio, mobile radio, amateur satellite communications, space communication, and space research (**see Table 6.4**). All of which have a designated frequency to operate within for both transmitting and receiving.

Band Designation	Frequency Range	Wavelength	Usage
Audible	20 Hz-20kHz	>100 Km	Acoustics
Extremely/Very Low Frequency (ELF/VLF) Radio	3 kHz-30 kHz	100 Km-10 Km	Navigation Weather Submarine Communications
Low Frequency (LF) Radio	30 kHz-300 kHz	10 Km-1 Km	Navigation Maritime Communiations
Medium Frequency (MF) Radio	300 kHz-3 MHz	1 Km-100 m	Navigation AM Radio
High Frequency (HF) Radio	3 MHz-30 MHz	100 m-10 m	Citizens Band (CB) Radio
Very High Frequency (VHF) Radio	30 MHz-300 MHz	10 m-1m	Amateur (HAM) Radio VHF TV/FM Radio
Ultra High Frequency (UHF) Radio	300 MHz-3 GHz	1 m-10 cm	Microwave Satellite UHF TV
Super High Frequency (SHF) Radio	3 GHz-30 GHz	10 cm- 1 cm	Microwave Satellite
Extremely High Frequency (EHF) Radio	30 GHz-300 GHz	1 cm- 0.1 mm	Microwave Satellite
Infrared Light	10^3-10^5 GHz	300μ–3μ	Infrared
Visible Light	10^13-10^15 GHz	1μ–0.3μ	Fiber Optics
X-Rays	10^15-10^18 GHz	10^3μ–10^7μ	N/A
Gamma & Cosmic Rays	> 10^18 GHz	<01^7μ	N/A

Table 6.4: Frequency Spectrum

After determining that frequency management was an extremely complicated set of both political and technical issues, the design process continued. Looking at different frequency designations, it was noticed that most commercial satellites, regardless of mission, operate at the higher L, S, C, and Ku bands. These frequency band designations are much sought after due to their increased performance capabilities. Most of today's high tech communications satellites contain and operate numerous transponders, leading to millions of dollars in revenue for the operating company. These satellites require higher frequencies to be capable of compressing and transmitting clear signals. With multiple Tele-communication and television/satellite companies emerging from ever increasing consumer demand, each of which willing and capable to operate at near full capacity of their respective transmitting capabilities, is leading to a severe congestion of certain bandwidths.

The congestion of certain bandwidths (higher frequencies) and increased difficulty of allocating a frequency license from the Federal Communication Commission and the International Telecommunications Union (for overseas communication needs/requirements) drove the decision to operate at the low end of satellite communications frequency spectrum. For LOCO-1's communication system, it was chosen to operate at an Ultra High Frequency (UHF) of 400 MHz. Telemetry and command signal transmissions were to be accomplished using an overall minimal frequency bandwidth. Uploading and downloading capabilities were to be done without interfering with one another's signal quality. A total UHF transmitting frequency bandwidth of 5 MHz was chosen to be capable of fulfilling the satellite's communications requirements. The 395 MHz - 400 MHz frequency range was base-lined for both the preliminary and final design.

6.3 Data Rates Required

The total amounts of data (in any form) to be downloaded by LOCO-1 was base--lined at 56 kbps, keeping pace with targeted competitors current capabilities. To achieve the base-line requirements, it was a matter of choosing a simple modulation scheme such as binary phase shift keying (BPSK). After running simulations in Satellite Tool Kit (STK version 4.0.6) it did not seem likely that BPSK would be able to achieve downloading the entire stored memory within reasonable time. Ayedin Telemetry Inc. was contacted, from whom we attained further technical capability information on the transmitter. The information provided, indicated the transmitter could reach much higher data transmission rates than our base-lined requirements. The actual transmission rate is dependent of modulation scheme.

6.4 Transmitter/Receiver/Encoder

With the communications parameters documented, better understood, and committed to, the next step was to determine how the requirements were to be met. Options included the actual in house design and testing of a capable transmitter and receiver, which would meet the UHF frequency needs, or to look for readily available technology. Without proper education or understanding of electrical engineering and radio technology, along with the relatively small time scale to work with, it was opted to build the communications system component by component. Consulting with professionals from industry, heavily involved within the communications sector, on which components would be capable of meeting our very stringent needs, the process of analyzing the transmitting capabilities began. The stringent needs arose from the low power generation LOCO-1 produces due to the small area of its body mounted solar arrays.

One other factor, which contributed to the decision of choosing one telemetry company over another, was the shear lack of commercially available products that operate at the targeted frequency. Looking for examples of other small-scale satellites, such as San Jose State University's micro satellite, the SPARTAN, it was learned that most small-scale satellites builders developed, tested, and implemented their own communications packages.

The search for contacts willing to listen and send useful responses took some time, but paid off when individuals from the telemetry industry showed interest in LOCO-1 and responded in a timely manner to questions and needs. While this networking came during the second quarter, information from vendors came in a steady stream. Within weeks, enough information was produced that kept us busy trying to determine which transmitter, receiver, and encoder would most economically fulfill all of our requirements.

Commercial of the shelf (COTS) units were chosen for transmitting, receiving, and encoding. Ayedin Telemetry Inc. provided the UHF components for LOCO-1. The transmitter (model T-102) consumes 56 Watts of power (input) to transmit 10 Watts out (output). This transmitter is capable of handling analog and digital signals, transmitting them at rates up to 700 kbps (depending on modulation scheme). To conserve power, the transmitter is only to be activated while the satellite is over the designated ground station. One two-second warm up interval, prior to downloading, is required when the transmitter comes on line.

The receiver (model RCC-102) requires 6 Watts of continuous power. It needs to be operational at all times to receive commands at any time.

The encoder (model MMSC-800) operates when the computer downloads data. It receives the input data from the onboard computer and outputs the data in the transmitter's modulation format. This set up allows efficient use of both computer and transmitter networking.

All units are housed in separate enclosures, by the manufacture, to provide shielding. Each unit is housed in rugged casings to withstand severe environmental flight conditions, allowing easier and quicker integration. (Required wiring is standard coaxial cable and connectors, specs are provided by manufacture).

6.5 Choosing the Antennas

Choosing the best-suited antenna to complete the communication system was a learn-by-doing process. Antenna design is best left for those who posses a strong understanding of electrical engineering and know how to implement the theoretical into practical designs. We decided to seek the advice of the electrical engineering department and professionals as well, to aid our antenna design.

Selection of the antennas had to take into account satellite performance requirements. These requirements included the download transmission rates, orbit pass times, coverage area, and link budget. The download transmission rate is given by the capabilities of the transmitter and the modulation scheme chosen. Time per orbit was then determined through STK analysis, giving us an understanding of coverage area and downloaded data.

Taking into account the aforementioned parameters, different antenna design types were analyzed. We considered dipole, horn, helix, and parabolic (**see Figure 6.4**). After review we chose a helix antenna design as our best-suited down link option given its high gain, low attenuation, simplicity, easier deployment, and directional beam pattern. In addition to the helix antenna, two monopole (Omni-directional) antennas will be used on the satellite (**see Figure 6.5**). One of the two monopole antennas is used for receiving, while the other will serve as the down link antenna while the satellite is de-tumbling from the launch vehicle or any other irregular mode. The computer will activate a switch to go back and forth between the helix and monopole antennas.

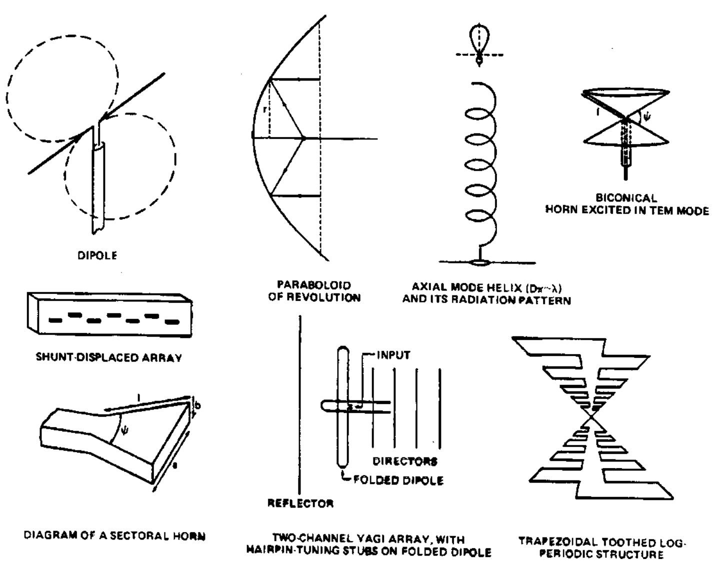


Figure 6.4: Different Antenna Types

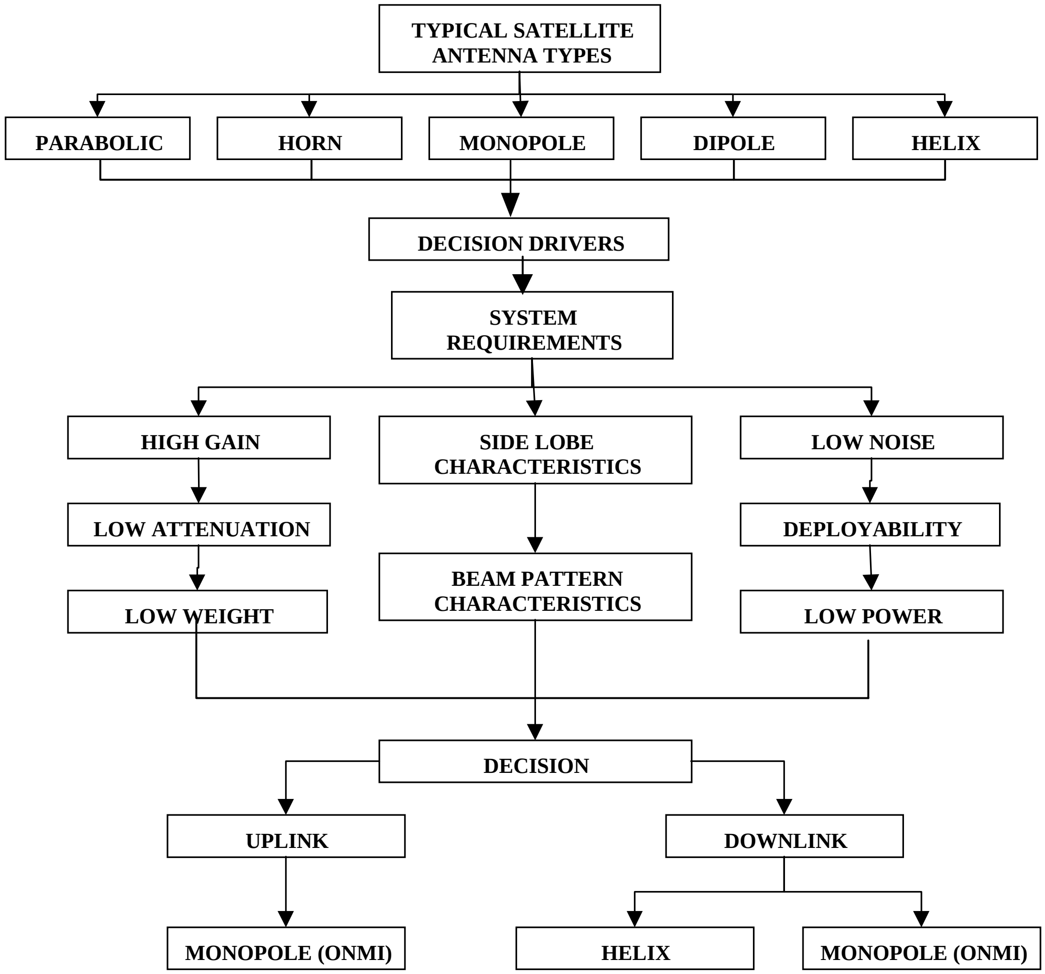


Figure 6.5: Antenna Decision Process

6.6 Helix Antenna Design

The next step was to determine the half-power beam-width, gain, and physical dimensions of the helix antenna (See **Appendix C**). This was done using empirical equations and graphical relationships provide in a report by the National Radio Astronomy Observatory and information gained from Space Mission Analysis and Design The values for the physical helix antenna dimensions, and performance were obtained from the following equations. Table 6.5 shows the final outcome of the parameters of the helix antenna.

$$Gain\ (dB) = 10.3+10\log(C^2L/\lambda^3) \qquad \text{[Eqn 6.1]}$$

$$HPBW\ (degrees) = 52/(C^2L/\lambda^3) \qquad \text{[Eqn 6.2]}$$

HPBW	Wavelength	Circumference	Length	Diameter	Gain	Turn Angle	Spacing	# of turns	Weight
degrees	m	m	m	m		degrees	m		kg
90	0.5	0.204275292	1	0.0650228	11.3974	13	0.2249511	4.4454115	0.04

Table 6.5: Final Helix Antenna Parameters

6.7 Onmidirectional Antenna Design

We also had to determine the half-power beam-width, physical dimensions, and gain of the Nomi-directional antenna. The following was determined for the Nomi-directional antennas:
Note: One of the Omni-directional antennas will be 0.45 m and operate at a different frequency.

HPBW	Wavelength	Length	Gain	Weight
degrees	m	m	dB	kg
180	0.5	0.5	1	0.015

Table 6.6: Final Omnidirectional Antenna Parameters

6.8 Switch

The computer-controlled switch to go from the helix antenna to the monopole is a DowKey 401 Series Latching Switch. The switch is a COTS component from DowKey Microwave. This switch is both space rated and flight proven.

6.9 Cables

The cable required for connecting the antennas to the transmitter/receiver is standard coaxial cable, but with regular coax we would have out gassing problems. Space rated flexible cable will be used on our satellite. The cable must be thin and flexible in order to be fitted in the cramped satellite compartments. MICRO-COAX provides miniature, flexible coaxial cable in a variation of diameters, all with standard industry connectors.

6.10 Link Budget Analysis

The link budget is an overall analysis of the overall communications system. An actual link budget takes into account all the systems (for both Uplink and Downlink) parameters such as the operating frequencies, transmitter power, and other such variables listed in **Table 6.7**. A typical link budget analysis proves to be a powerful tool the communication system’ s designer can use to determine whether or not the given system performance parameters will allow for clear signal transfer.

For the Uplink portion, we had to assume certain ground station elements. Among other things which we assumed (due to the lack of detail required for ground station required to complete our satellite's overall design) were the antenna type, diameter of the antenna, pointing offset, data rate, and others.

The Downlink portion of the link budget was more exact due to the fact that the analysis had been calculated using equations taking into account our systems constraints and parameters.

The following Table shows the link budget analysis along with gain margin, according to the specifications of the communication package (see **Appendix D**).

ITEM	SYMBOL	UNITS	SOURCE	UPLINK	DOWNLINK
Frequency	f	GHz	Input parameter	0.38	0.40
Transmitter Power	P	Watts	Input parameter	1.00	10.00
Transmitter Power	P	dBW	10logP	0.00	10.00
Transmitter Line Loss	Ll	dB	Input parameter	-0.50	-0.50
Transmit Antenna Beam Width	t	deg	Input parameter	55.26	90.00
Peak Transmit Antenna Gain	Gpt	dB	Eqn 6.3	69.40	11.50
Transmit Antenna Diameter	Dt	m	Eqn 6.4	1.00	n/a
Transmit Antenna Pointing Offset	et	deg	Input parameter	n/a	n/a
Transmit Antenna Pointing Loss	Lpt	dB	Eqn 6.5	-0.12	-0.75
Transmit Antenna Gain	Gt	dB	Gpt+Lpt	69.28	10.75
Equiv. Isotropic Radiated Power	EIRP	dBW	P+Ll+Gt	68.78	20.25
Propagation Path Length	S	km	Input parameter	6078.00	6078.00
Space Loss	Ls	dB	Eqn 6.6	-159.72	-160.17
Propagation & Polarization Loss	La	dB	Fig. 6.7	0.00	0.00
Receive Antenna Diameter	Dr	m	Input parameter	0.06	1.00
Peak Receive Antenna Gain	Grp	dB	Eqn 6.8	-21.94	9.85
Receive Antenna Beam Width	r	deg	Eqn 6.9	180.00	52.50
Receive Antenna Pointing Error	er	deg	Input parameter	45.00	5.25
Receive Antenna Pointing Loss	Lpr	dB	Eqn 6.10	-0.75	-0.12
Receive Antenna Gain	Gr	dB	Grp+Lpr	-22.69	9.73
System Noise Temperature	Ts	K	Table	1295.00	375.00
Data Rate	R	bps	Input parameter	10000.00	100000.00
Eb/No (1)	Eb/No	dB	Eqn 6.11	43.84	22.68
Carrier-to-Noise Density Ratio	C/No	dB-Hz	Eqn 6.12	83.84	72.68
Bit Error Rate	BER	-	Input parameter	10^-7	10^-4
Required Eb/No (2)	Req Eb/No	dB-Hz	Figure	15.00	9.00
Implementation Loss (3)	-	dB	Estimate	-6.00	-3.00
Margin	-	dB	(1)-(2)+(3)	22.84	10.68

Table 6.7: Link Budget Parameters

6.11 Satellite Tool Kit

Satellite Tool Kit was a tool used as allow further study and analysis of the communication system's chosen components. By imputing all of the operational requirements of the transmitter, receiver, encoder (data rate), antenna parameters (gain, beam pattern), altitude, orbit, inclination, and overall mass it provided a wealth of information. The information presented by STK included the amount of coverage time to expect from the passes LOCO-1 would make over a designated ground station location. It also allowed for a visual model to be represented, allowing for a visual inspection of the contrasting beam width patters attained by varying the types of antennas.

By analyzing the per pass and total time allowed to access the ground station, it showcased the performance of the commercial of the shelf components. With this information it could be determined it the communication's components were capable of satisfying LOCO-1's pre set requirements. Overall STK proved to be a very valuable tool to facilitate the design process for the antennas, transmitter, and receiver through its programmed simulation analysis.

Note: **Appendix E** provides the per-pass and total access time for LOCO-1. The heading of the appendix indicates the ground station location (West ground station at VAFB/Cal Poly and East ground station at – Kennedy Space Center), varying inclination (Min – Max range), and extreme altitudes of LEO (400 km – 700 km).

7. COMPUTER

The onboard computer responsible for house keeping of our spacecraft as well as storing all the data of the payload (for later transmission) is a single board computer (SBC). We wanted a SBC that was power effective, memory capable and low cost. We looked at many variations of SBC's and although many newer Pentium, 486, and RISC processors provided greater improvements over older, slower 386/387 processors, they required considerably more power.

Keeping in mind that LOCO-1 would not be able to produce large amounts of power, the SBC had to be capable of processing all of the necessary data routing, but do so on the barest of power consumption. Therefore the decision was essentially predetermined, in order to meet our stringent power requirements a SBC running on an older 386 processor was chosen. The decision was also aided due to the 386/387 combo's space proven reliability. Its memory capability was designed to be up to 64 MB, which is more than enough to store both payload and house keeping data.

Vendors for such SBC's were more that abundant. We were able to not only compare differing capabilities, but also look for bargain prices. The computer board we intended to use for our satellite would be completely stock off the shelf except for the epoxy used to hold all the components in place.

Radiation hardening was another issue to deal with. Radiation exposure in the harsh space environment was a factor considered in selecting a computer system set up which would allow for reliable operations. Radiation hardened (RAD HARD) computers were determined financially ineffective for LOCO-1's requirements. With orbit, altitude, and designed lifetime determined (LEO, 400 km, and > 1 year), it's believed with proper error detection and correction software LOCO-1's mission will be not only successful but prove to be cost effective over a radiation hardened computer.

8. POWER

8.1 Introduction

LOCO-1's power system is designed to balance redundancy and cost effectiveness. The power subsystem is required to power LOCO-1 for at least one year in low Earth orbit (LEO). In an attempt at keeping costs low, impacts on power requirements were looked at from a systems standpoint. Tradeoffs were analyzed to insure that the power subsystem would have the least number of possible failure points. Solar cells are the most common source of energy production for spacecraft in Earth orbit and are the primary power source for LOCO-1.

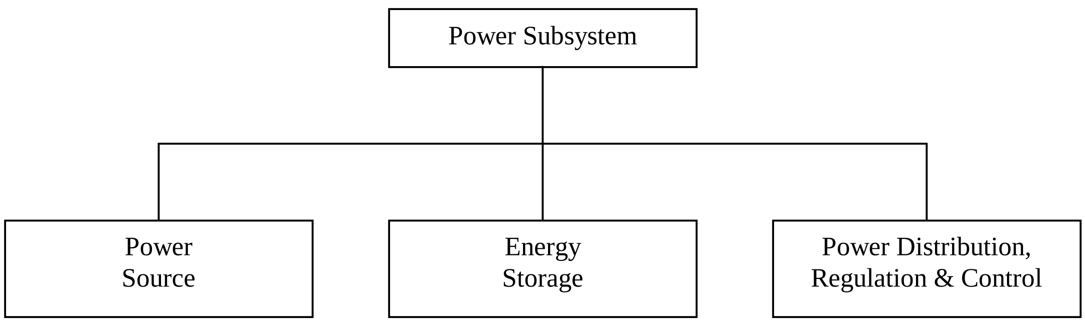


Figure 8.1: Functional Representation of LOCO-1's Power Subsystem

8.2 Power Architecture

The first tradeoff is whether LOCO-1 will have deployable solar arrays. This impacts structure by adding complexity and costs from the hinges and deploying mechanisms. However, the larger exposed area may allow lower cost, less efficient solar cells to provide sufficient power. A synopsis of tradeoffs can be found below in **Table 8.1**.

	Deployable Solar Arrays	**Body-Mounted Solar Arrays**
Advantages	• More Exposed Area • May allow less expensive cells	• Simple • No possibility of deployment failure
Disadvantages	• Possible Failure With Deployment • Adds complexity to structure • Higher costs for hinges and deployment mechanisms	• Require higher efficiency solar cells • Retains more heat than deployables (lower cell efficiency)

Table 8.1: Deployable Solar Array Tradeoffs

The possibility of deployment failure of the solar arrays was the deciding factor for body-mounted solar arrays. Body-mounted arrays remove that possibility of a single-point failure of the bus.

The voltage of the bus depends on the different subsystems. For maximum efficiency, the bus voltage should match the voltage of the majority of the components and also match the voltage of the largest power-consuming component. To reduce conversion losses from different voltages, the solar arrays and batteries should also operate at the same voltage. The configuration for LOCO-1 is a 28-volt bus. The power system is a direct energy transfer system, using a shunt regulator in parallel to the array. The shunt regulator dissipates heat from excess current. The direct energy transfer system is advantageous over the alternative peak-power tracker system because it has few parts, lower mass, and has a higher efficiency at the end of the satellite's life. The peak-power tracker system is also in series with the array and uses 4-7% of the total power. Since the arrays and batteries are both at 28 volts, power regulation is unregulated but uses chargers for controlling current. There are also two mechanical switches that complete the circuit of the bus when the satellite is released from the launcher. Since a secondary payload has to be shut off during launch, the mechanical switch is necessary to turn on the satellite and initiate the power subsystem. This architecture delivers 28 volts of power to the power board, which then distributes the power to all the subsystems. See **Figure 8.2** to see the power regulation schematic.

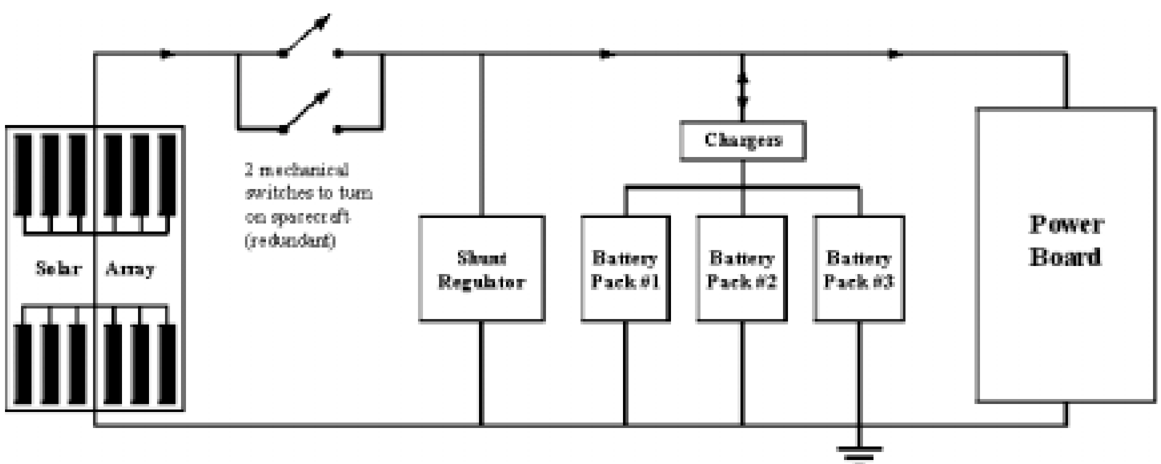


Figure 8.2: Power Regulation Design

8.3 Power Source

8.3.1 Solar Cell Type

Since LOCO-1 is to use solar photovoltaics as its primary source of energy, two different solar cells, Silicon (Si) and Gallium Arsenide (GaAs) were compared as a possibility for the LOCO-1. Silicon cells is a more mature technology and are less expensive, but have low efficiencies of about 15%. Silicon cells also degrade more per year than GaAs cells from radiation and have a low cell voltage, requiring more cells in series to achieve 28 volts. Though Gallium Arsenide cells are more expensive, they are more efficient, degrade less, and are more resistant to thermal changes. Since LOCO-1 will not employ deployable solar arrays, Gallium Arsenide cells are required to provide sufficient power to the bus.

Gallium Arsenide cells come in three different types: Single Junction, Dual Junction, and Triple Junction. Multi-Junction cells are made of layers of different material, which react to different light wavelengths. This means that multi-junction cells have higher efficiencies. A dual junction GaAs solar cell type would be required for the amount of power required by LOCO-1. (See **Figure 8.2** for solar cell comparisons)

Solar Cell Type	**Efficiency**	**Radiation Degradation per year**	**Cell Voltage**	**Min. # cells required in series for a 28-volt bus**
Silicon	15%	3%	0.45	63
GaAs Single Junction	19%	1%	0.90	32
GaAs Dual Junction	21.5%	1%	2.06	14
GaAs Triple Junction	24%	3%	2.26	13

Table 8.2: Solar Cell Comparisons (Data received from *Spectrolab, Inc.*)

Spectrolab, Inc provides the GaAs Dual Junction solar cells that LOCO-1 employs. They are 3.5 cm by 7 cm and weight 100 mg/cm^2 at 175 x 10^{-6} m thickness. They have a solar absorptance of 0.91 and emittance of 0.85.

8.3.2 Solar Panel Construction

Each solar panel of the solar array consists of 15 cells in series for a total of 30.9 volts. Each panel is approximately 18 cm x 22 cm. This gives the panels some redundancy to account for the effects of radiation degradation, temperature, or a single-cell failure. A higher voltage is necessary when charging the secondary batteries. The solar panels are mounted on five of the six sides of the satellite, with the sixth side being reserved for the launcher ring, earth sensor, and payload viewing.

Satellite Side	**Side Dimension**	**# of solar panels**	**Peak Power at EOL**
Large Side #1	600 mm x 460 mm	6	55 W
Large Side #2	600 mm x 460 mm	0 (Payload side)	0 W
Medium Side #1	600 mm x 250 mm	3	27 W
Medium Side #2	600 mm x 250 mm	3	27 W
Small Side #1	460 mm x 250 mm	2	18 W
Small Side #2	460 mm x 250 mm	2	18 W

Table 8.3: Solar Panel Distribution

8.3.3 Available Power

To calculate the power a solar panel produces, the equation used is:

$$P = AG_s \eta I_d \cos\theta \qquad \text{[Eqn. 8.1]}$$

Inherent degradation are losses due to assembly, the temperature of the array, and shadowing of the cells in orbit. Since there are no protrusions on the sides of the spacecraft, the primary concern would be assembly and temperature. GaAs is more resistant to losses from temperature than silicon and so a 10% overall loss is assumed.

To ensure that the satellite receives enough power, the amount of power produced is graphed as the satellite traverses its orbit. Since the primary launch site is Vandenberg AFB, the most typical orbit, a polar, circular orbit at 400-km altitude is assumed. Orbits ranging from noon-to-midnight to 6pm-to-6am were examined. (See **Figure 8.3**) The amount of power received is similar for these different orbits with the exception of 4:30 pm to 4:30 am.

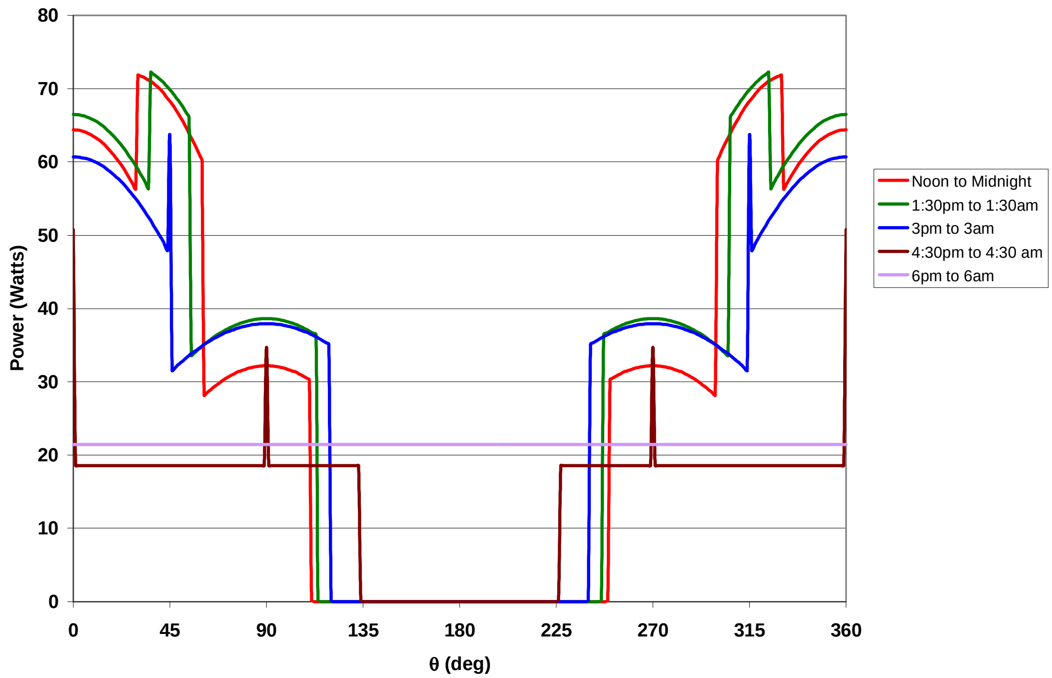


Figure 8.3: Power Produced vs. Orbit Angle

8.4 Power Storage

8.4.1 Power Storage Type

The secondary batteries are required to provide sufficient power to the satellite during eclipses. The batteries are also subjected to 16 charge/discharge cycles each day in a low-Earth orbit. For a one-year mission, this means that the batteries must withstand approximately 5700 cycles. Nickel Cadmium cells provide the optimal depth-of-discharge (DOD) for its given lifetime. NiCd cells are the most technologically mature cell type, providing power to spacecraft since the beginning of space exploration. Newer cells, such as Lithium Ion or Nickel Hydrogen, have a lower available depth-of-discharge.

8.4.2 NiCd Battery Pack Configuration

Commercial-grade NiCd cells are also available, which increase redundancy and reduce cost over MIL-SPEC cells. Commercial cells from SAFT batteries cost $140 each, as opposed to MIL-SPEC cells from Eagle-Picher, which cost $4200 each. Less-expensive commercial cells allow the battery packs to be configured for the 28-volts that the bus operates at. There are three battery packs that consist of 24 cells each. At 20% depth-of-discharge, these packs must provide an average eclipse load of 35 W. To calculate the battery capacity required, the equation used is:

$$C_r = \frac{P_e T_e}{(DOD)Nn} \qquad \text{[Eqn. 8.2]}$$

Assuming a 90% efficiency and a maximum eclipse time of 0.6 hours for a 400-km altitude, there is a required battery capacity of 38.9 W-hr or 1.35 A-hr. SAFT VR1.6 CE cells are 1.6 A-hr, 1.2-volts each, which allow an average load of 41.5 W during eclipses. Having three battery packs allow the mission to continue even if one pack fails, either by a decrease in power usage or a decrease in the satellite's lifetime.

8.5 Power Board

The power board contains two space-rated DC/DC converters from *Interpoint* to lower the voltage for certain subsystems. (See **Figure 8.4** for the power board schematic) A 5-volt converter provides power to the three magnetic torquers, fifteen thermal sensors, and the on-board computer at 70% efficiency. The ±12-volt converter provides power to the magnetometer at 80% efficiency. There are fourteen electronic switches on the board, regulating power to the payload, magnetic torquers, magnetometer, heaters, and transmitter. The remaining subsystems: receiver, thermal sensors, and on-board computer are vital and require power continuously. The switches are controlled by commands from the on-board computer.

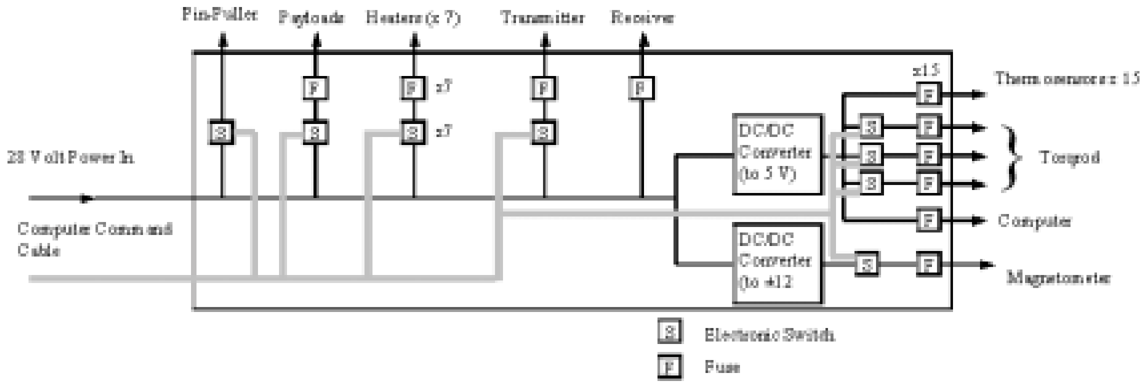


Figure 8.4: Power Board Schematic

9. THERMAL

9.1 Introduction

The space environment contains many dangerous hazards for satellites. One of these hazards is direct solar radiation, which can cause parts of the satellite to reach temperatures of over 100^{o}C. While these parts absorb highly concentrated heat loads, other non-illuminated areas of the spacecraft maintain extremely low temperatures. With this large range of temperatures, the electrical instruments need to be regulated at the correct operating temperatures. In addition, the structure and component materials need to contribute to dissipating the heat away from the spacecraft.

9.2 Thermal Analysis

LOCO-1's thermal analysis contains two parts: sunny-side and dark-side. The sunny-side occurs when any part of the satellite comes in contact with solar radiation. The dark-side is the exact opposite: none of the satellite's surfaces come into contact with solar radiation. Once the satellite moves into the shadow of the earth, the eclipse creates an immediate blanket of cold temperatures. Preliminary calculations show that LOCO-1’s temperature reaches extreme freezing temperatures during the eclipse time and remains cold throughout the orbit. To solve this problem, thermal blankets are outfitted inside of the satellite. **Table 9.1** shows the effectiveness of different types of thermal blankets.

Material	**Solar Absorptivity a**	**Infrared Emissivity e**	**a/e**	**Equilibrium Temperature (K)**
Gold	0.30	0.69	0.43	747
Teflon	0.80	0.16	4.91	216
Silverized Teflon	0.66	0.08	8.25	232
Aluminum Tape	0.40	0.10	4.00	264

Table 9.1: Comparison of Different Thermal Blankets

Looking at **Table 9.1** the best type of blanket needed for thermal insulation is thin gold foil. Gold foil has a high emissivity rate allowing radiation to bounce off the sides of the satellite interior, keeping heat inside. By keeping the heat within the interior of the satellite the temperature of LOCO-1 can be regulated during the eclipse. The exterior surface of the satellite needs to absorb as much heat as possible before entering the period of the eclipse. Therefore, a material with a high absorbtivity is needed for the outer surface. Also, since the solar cells are to be glued to the outside, the material must allow for mounting of the cells. Thermal blankets can be used, but paint base is the most cost-effective choice. **Table 9.2** is a listing of different thermal paints and their characteristics. Chemglaze Black Paint is the best choice since it has the highest absorbtivity at a convenient price.

Material	**Solar Absorbtivity A**	**Infrared Emissivity e**	**a/e**	**Equilibrium Temperature (K)**
Chemglaze Black	0.96	0.84	1.14	404
Anodize Black	0.88	0.88	1.00	365
Delrin Black Plastic	0.96	0.87	1.10	397
Tedlar Black Plastic	0.96	0.90	1.07	414

Table 9.2: Thermal Paints and Their Characteristics

For the thermal calculations, the heat concentrations are assumed constant over the entire spacecraft. The aluminum structure provides excellent conductance throughout the satellite and dissipates heat concentrations. LOCO-1's structure will provide heat dissipation to keep temperatures at equilibrium. The structure material, aluminum, has a thermal conductivity of 164 W/mK. In addition, the gallium arsenide solar cells radiate heat just as fast as they absorb it such that the solar panels are kept at an acceptable temperature. Equations 9.1 and 9.2 below are used for the thermal analysis of the maximum and minimum temperatures.

$$T_{min} = \left(\frac{\left(0.5q_1\varepsilon(1-\cos\rho) + \frac{Q_w}{\pi D^2} \right)}{\sigma\varepsilon} \right)^{0.25} \qquad \text{[Eqn. 9.1]}$$

$$T_{max} = \left(\frac{\left(0.25G_s\alpha + 0.5q_1\varepsilon(1-\cos\rho) + 0.5G_s a\alpha K_a(1-\cos\rho) + \frac{Q_w}{\pi D^2} \right)}{\sigma\varepsilon} \right)^{0.25} \qquad \text{[Eqn. 9.2]}$$

To maintain a range of temperatures, thermistors (heaters) are employed on most components to help maintain the correct operating temperature (0^o – 40^o C) during eclipses. The heaters are simple resistors that give off heat through certain voltages. Eight heaters are placed on temperature sensitive components. These components are the three battery packs, earth sensor, receiver, transmitter, computer power board, and the three-axis magnetometer. The payload is responsible for maintaining it's own thermal environment. Thermal sensors are placed along the interior surface as well as on the temperature sensitive components in order to record thermal data. Temperature readings will be taken once every minute while in orbit. All thermal instruments are purchased through the muRata Corporation.

10. STRUCTURES

10.1 Material

For the structure, the design must be easy to build to allow minimum assembly time. A skin stringer structure is the best configuration since it has a simple design. Materials that are used in the satellite market are shown in **Table 10.1**.

	Aluminum	**Titanium**	**Magnesium**	**Steel**	**Beryllium**	**Fiber Glass**
Advantages	Good Strength Weldable Good Rigidity Available Low Cost	Excellent Strength Excellent Rigidity High-Temperature	High Stiffness	High Strength High Temperature	Low Density High Strength	Lightweight Good Strength Good Stiffness
Disadvantages	N/A	Expensive Low Availability	Limited Availability Difficult to Process	Out-gases	Toxic Expensive	Hard to Manufacture

Table 10.1: Structure Materials

When deciding on a material the following properties are necessary: high strength, rigidity, lightweight and does not out-gas. When exposed to severe vacuum conditions, the material can produce a violent release of entrapped gas, solvents, oxidizers and other molecular particles. This can result in a loss of material forcing the properties of the material to become brittle and porous. After comparing different materials, Aluminum 6061 L-beams were chosen for the skeletal structure due to their low weight, rigidity, and strength. In addition, L-beams provide a means for mounting system components. Aluminum also allows excellent thermal conductivity for the satellite. The skin of the satellite must also provide excellent loading as well as low weight. The best and most common material is a honeycomb product. Honeycomb allows extra support of the structure but allows for a lighter overall weight. Therefore, the skin of the satellite is made of aluminum honeycomb for added rigidity, while still providing excellent thermal protection. Plascore Inc. supplies both the honeycomb and L-beams. An overall schematic of the satellite structure can be seen in **Appendix A**.

The solar arrays will attach directly to the aluminum honeycomb with a white epoxy. This epoxy is proven not to out-gas. All material will be bought in bulk. This allows the structure to be built far in advance, cutting down production time, and lowering cost.

10.2 Analysis

Equations 10.1 and 10.2 below are used to calculate the stress on the satellite truss. A preliminary stress analysis is done on the satellite truss before applying the finite element method.

$$\sigma = \frac{P_{eq}}{A_{tot}} \qquad \text{[Eqn. 10.1]}$$

$$\sigma_{cr} = \left(\frac{k\pi^2 E}{(12 - 12\upsilon^2)}\right)\left(\frac{t}{b}\right)^2 = 0.93kE\left(\frac{t}{b}\right)^2 \qquad \text{[Eqn. 10.2]}$$

10.3 Finite Element Analysis

A finite element analysis was conducted using COSMOS. The structure is drawn in COSMOS with all distributed weight loads. The requirement for a Delta II launch is to survive a 10-g loading in all directions. Figure 10.1 shows the resulting stress analysis (the results are in mega-Pascals, MPa). The maximum stress for aluminum, for failure to occur, is 1.67 MPa. From COSMOS, the maximum value reached for the 10g loading is 0.13 MPa. The main concentration of stress is located about the ring attachment. This attachment supports all of the weight of the spacecraft and vibrational loads. Therefore, LOCO-1 will survive launch loads. Further analysis will be conducted when the prototype is built and tested. In Figure 10.2 the displacement of the satellite is shown. The maximum displacement is 0.001 inches. This concludes that LOCO-1's ring attachment will not deform or break during launch.

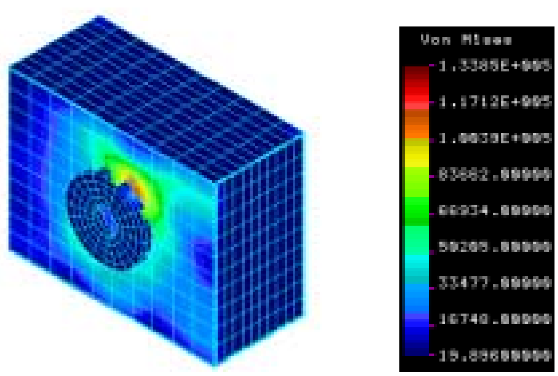


Figure 10.1: Stress Analysis

The maximum displacement location is far away from the center and in the positive moment direction. For all analysis, the forces and moments are assumed to be only in the positive direction, which is why the deflection is up and to the right.

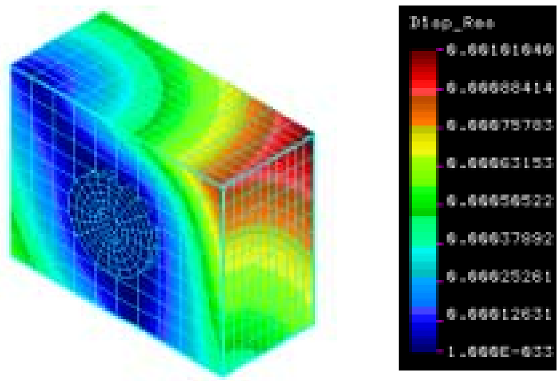


Figure 10.2: Displacement Analysis

10.4 Secondary Structure

LOCO-1 uses some of the components as load bearing supports. These items include the batteries, mounting brackets, computer and power board. The finite element analysis ignores that the batteries and mounting brackets are load-bearing components. This will increase the structural integrity of the spacecraft rather than harm it. Table 10.2 shows the mass budget for all the system components on the satellite. To account for rivets and bolts the estimated overall weight will be relatively higher.

Subsystem	**Mass (kg)**
Payload	20
Power	6.4
Communications	2
Command & Data Handling	N/A
Structures	6.0
Thermal	0.2
ACD	1.9
Deployables	5.2
Wiring	0.1
Total	**41.8**

Table 10.2: Mass Budget

10.5 Payload Area

The payload area will consist of two spaces whose volume is 218 x 428 x 152 mm each (see **Appendix A** for a more detailed view). This space is allotted to the payload for integration of its instruments and computer. The payload box will be made of 7072 aluminum plating. This five-sided box has one-side open to make integration easier. The box is made of solid 6.35-inch thick aluminum plating. To reinforce the satellite structure the box will be bolted into position. LOCO-1 allows a maximum mass of 10 kg for each payload area.

11. DEPLOYMENT

11.1 Launcher Release

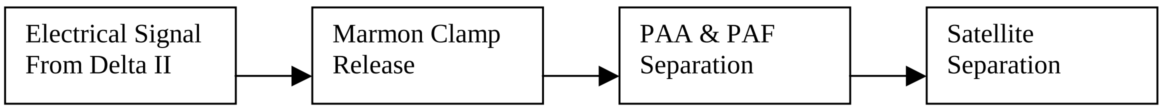


Figure 11.1: Satellite Release Sequence from Delta II.

The release sequence from the Delta II is shown in **Figure 11.1**. The LOCO-1 satellite is mounted on the Delta II launcher via a payload attachment adapter (PAA) which is secured to the payload attachment fairing (PAF) with a marmon clamp. This clamp then releases the satellite from the launcher by bolt-cutters, which are provided by the launcher. The satellite is then released when the bolt-cutters free the PAF from the launcher. The two monopole (omni-directional) antennas that are contained within two cavities on the PAA are rolled up for storage during liftoff, then upon the satellite's release, they freely unwind out of their cavities (**Figure 11.2**). Two switches located on the surface of the PAA turn on the satellite, which are pressed down by the PAF prior to satellite deployment.

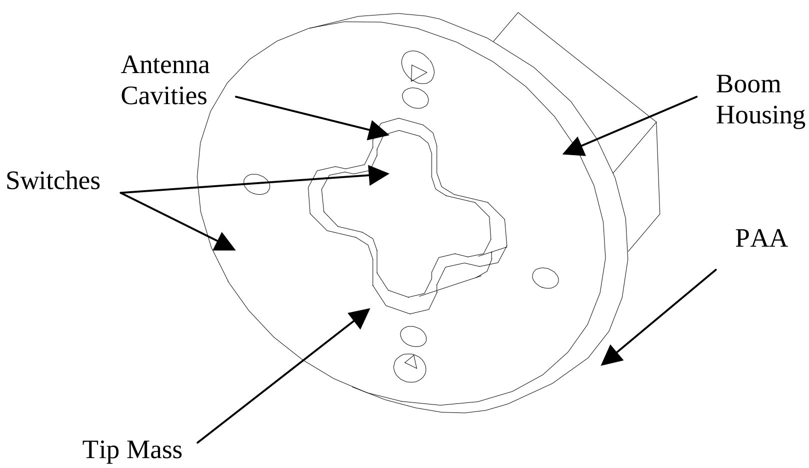


Figure 11.2: Omni-directional Antenna and Tip Mass Storage

11.2 Boom Deployment

The boom is stowed in a housing that is attached underneath the PAA. The tip mass is stored within the center opening of the PAA. It compresses the helix antenna within the opening. The boom is held in the stowed position with a non-explosive pin puller. An electrical signal is sent to the pin puller and starts the deployment sequence. When the pin puller releases the boom reel, the helix antenna springs out pushing on the tip mass which deploys the boom. The deployed configuration of the boom and helix antenna is shown in **Figure 11.3**.

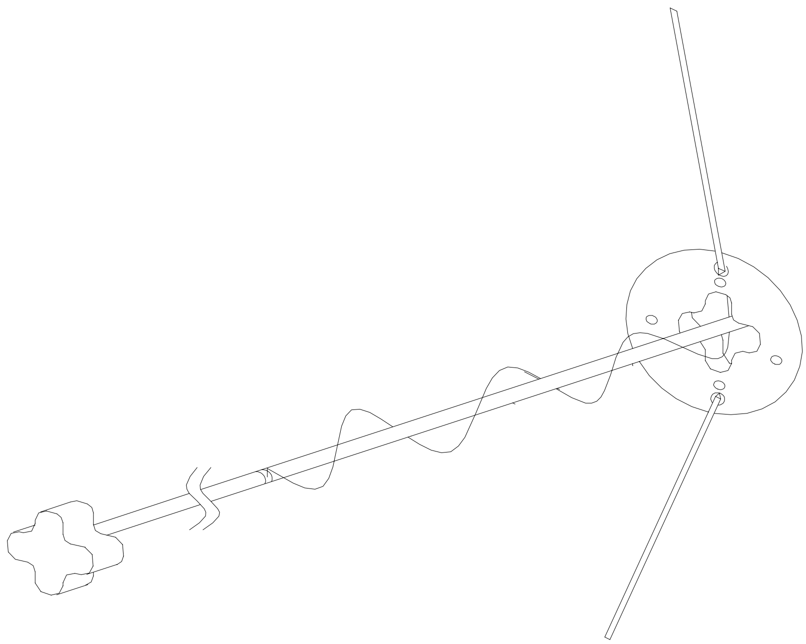


Figure 11.3: Boom and Antennas in Deployed Configuration

11.3 Boom Selection

The selection process followed is shown in **Figure 11.4**. The main considerations are to make the boom simple, reliable, and cost effective. This resulted in choosing a tubular type of boom that uses a single reel.

The boom length is 3.949 m and made of berylluim copper since it is a common material used for booms. The 3.51 kg tip mass is made of tungsten, which again is a common material used, and the shape determined for easy clearance from the PAA. The boom and tip mass is provided by *TRW Astro Aerospace*.

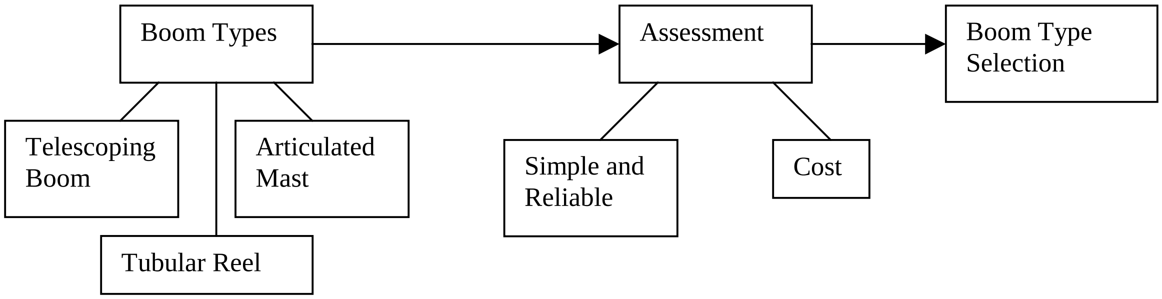


Figure 11.4: Boom Selection Process

The type of deployment device was chosen with great consideration to protecting the components and payload inside LOCO-1. Pyroshock is a primary consideration since using pyrotechnic devices can lead to damaging the spacecraft and/or components from direct interaction or by exposing the spacecraft and components to a shock environment. This lead to the selecting a non-explosive pin puller deployment device. The LOCO-1 uses a Model 8003 non-explosive pin puller provided by *G&H Technology, Inc.*; the pin puller specifications are shown in Table 11.1.

Shaft Spring Force	55 lb armed; 12 lb
Shaft Stroke	0.5 in. max.
Shaft Stroke Time	50 ms after electrical initiation (approx.)
Housing	Aluminum Alloy (Anodized)
Weight	186 g

Table 11.1: Model 8003 G&H Tech. Inc. non-explosive pin puller specifications.

12. OPERATIONS

12.1 Software

The operating system of LOCO-1 will be programmed using either C++ or another higher level language since it is easier to code and since the amount of system memory available is no longer a restricting factor. The operating system is encoded onto a chip on the computer board and is loaded into dynamic memory (RAM) on boot-up. This allows the satellite to overwrite its operating system and update any necessary changes from the ground. Having a version of the operating system on the board also acts as a back-up in case of an interruption in power or if the operating system in memory is corrupted.

12.2 LOCO-1 Housekeeping Information

The on-board computer will continuously record housekeeping data about the satellite once per minute to download to the ground with each pass. The thermal sensors, data from the earth sensor, magnetometer and all three capacities of the batteries are recorded. For a 24-hour period, this data packet is estimated to be about 70,000 bytes of information. This information can be used to analyze attitude data, current battery usage, and also help troubleshoot any abnormal behaviors that LOCO-1 may be exhibiting.

12.3 Modes

LOCO-1 is designed to have minimal autonomy, but it still requires several modes of operations so that it can operate while in orbit. The modes required are the Beacon, Safe, De-Tumbling, Transmitting and Standard Operations mode. Both Standard Operations and Transmitting have sub-categories for day and eclipse periods with the primary difference of the heaters being on during the eclipse. Though different modes switch on and off different components there are some that require being on constantly. These vital systems include the power subsystem, the UHF receiver, thermal sensors, heaters as necessary, and the on-board computer. (See **Table 12.1** for Spacecraft Modes Power Budget)

12.3.1 Beacon Mode

Beacon mode is used initially after separation from the launch vehicle and also when the spacecraft needs to indicate a problem to the ground. The satellite switches from the downlink helix antenna to the omni-directional antenna in case pointing accuracy or knowledge is compromised. Beacon mode uses minimal power, keeping only the thermal system and communications system. The satellite then broadcasts a one-second beep every minute to advertise its position and situation. Beacon mode is initiated by a trigger or can be commanded from the ground. These triggers are:

- Satellite does not receive a signal from the ground for 3 days
- Satellite battery levels fall under 40%

- Earth sensor and magnetometer readings differ more than 20 degrees

Beacon mode stores which event caused its initialization and transmits that data along with housekeeping data to the ground when contact is established. Beacon mode can only be halted via a command from the ground.

12.3.2 Safe Mode

Safe mode is a sub-category of Beacon mode. Safe mode is the satellite's last effort to recover itself. Similarly to Beacon mode, the satellite switches from the helix antenna to the omni-directional antenna. Safe mode uses the absolute minimum power. It allows the temperature for each component drop to within 5 degrees Celsius of their lowest operational temperature. Safe mode can either be commanded from the ground or is triggered if the battery levels fall below 25% capacity. Safe mode does not allow the satellite to broadcast but will revert to Beacon mode if the safe mode was initiated because of low batteries and if the battery levels rise above 30%. The only other way for the satellite to exit safe mode is to receive a command from the ground to do so.

12.3.3 De-Tumbling Mode

The De-Tumbling sequence is initiated after contact with the satellite is established, after release from the launch vehicle. (See **Figure 12.1** for a typical mission timeline) De-Tumbling can also be initiated from the ground if the satellite requires it. The de-tumbling sequence is a separate mode from Standard Operations (where housekeeping attitude determination and control is conducted) since all non-essential systems are shut down and that the spacecraft is dedicated to de-tumbling the spacecraft. This was necessary for the initial release from the launcher since battery levels may be lower than average and that the spacecraft is unstable when the boom is not deployed. Once the spacecraft is in the correct orientation, only minimal housekeeping throughout its orbit is necessary.

12.3.4 Transmitting

The UHF Transmitter that LOCO-1 employs requires 56 watts of power when it is transmitting. Though this is a large drain of power on the satellite, it is in relatively short intervals. The satellite does not do any attitude housekeeping to reduce the risk of magnetic interference but continues to supply the payload with 10 watts of power.

12.3.5 Standard Operations

LOCO-1 will be operating in Standard Operations mode for most of its lifetime. This mode involves providing the payload with 10 watts of continuous power and keeping the satellite orientated correctly.

System	Safe Mode	Beacon Mode	De-Tumble Mode	Day Trans.	Eclipse Trans.	Day Op.	Eclipse Op.
Payload	0	0	0	10	10	10	10
Communications							
UHF Receiver	4	4	4	4	4	4	4
UHF Transmitter	0	0 / 56	0	56	56	0	0
On-Board Computer	1.5	1.5	1.5	1.5	1.5	1.5	1.5
ADC							
Mini-Dual Earth Sensor	0	0	0	0.8	0.8	0.8	0.8
3-Axis Magnetometer	0	0	1.25	1.25	1.25	1.25	1.25
Torqrods (3)	0	0	0.54	0	0	0.54	0.54
Thermal							
Sensor	1.71	1.71	1.71	1.71	1.71	1.71	1.71
Heaters	2.5	2.5	2.5	0	2.5	0	2.5
Avg. Power Available EOL	22.2	22.2	22.2	22.2	22.2	22.2	22.2
Power Used	9.71	9.71	11.5	75.26	77.76	19.8	22.3
Leftover Margin	12.49	12.49	10.7	-53.06	-55.56	2.4	-0.1

Table 12.1: Spacecraft Modes Power Budget

12.4 Typical Mission Timeline

Satellite Tool Kit helped construct a typical mission (See **Table 12.2**, next page) that follows a satellite from launch until it starts its standard operations. This "strawman" mission is a midnight launch on May 21, 1999 from Vandenberg AFB. The satellite is put into a 400 km altitude, polar, circular orbit. One ground station, located at the California Polytechnic State University (roughly 35 deg N, 120 deg W) uploads the commands to the satellite during available satellite passes.

Date	Time (PST)	Event	Mode
May 21, 1999	0:00:00	Launch from Vandenberg AFB on Delta II	Satellite Off
May 21, 1999	1:24:00	**Release from Launch Vehicle** **Omni-directional antennas deployed** **Satellite switched on mechanically** - Batteries turned on - Computer turned on - Operating System loaded into memory	Beacon Mode
May 21, 1999	2:04:04	Contact with Spacecraft Established	De-Tumbling Mode
May 21, 1999	13:59:48	**Deployment of Boom and Helix Antenna**	De-Tumbling Mode
May 21, 1999	15:33:13	**Confirmation of Boom Deployment** - Attitude Dynamics Analyzed	De-Tumbling Mode
May 22, 1999	1:16:11	**Switch from Omni-directional to Helix Antenna**	De-Tumbling Mode
May 22, 1999	2:51:12	**First downlink through Helix Antenna** - Confirmation of Helix Deployment	Transmitting (Helix)
May 22, 1999	2:54:12	**Start of Standard Operations**	Standard Operations

Table 12.2: Typical Mission Timeline

13. INTEGRATION

13.1 Components

LOCO-1 consists of a modular design. Each system component has its own section of the spacecraft. If a system is corrupted or malfunctions during testing, it can then be easily replaced. All components will be integrated into the satellite in an assembly line fashion. Considerations for electrical connections and data lines are required so that the smallest amount of cabling is used. This is done to keep both the weight and complexity down. Figure 13.1 is an example of the different connections that relay from one component to another. The red line represents power connections from the power board. The black lines are the data transfers from the component interfaces. The arrows show the path that the data or power is flowing.

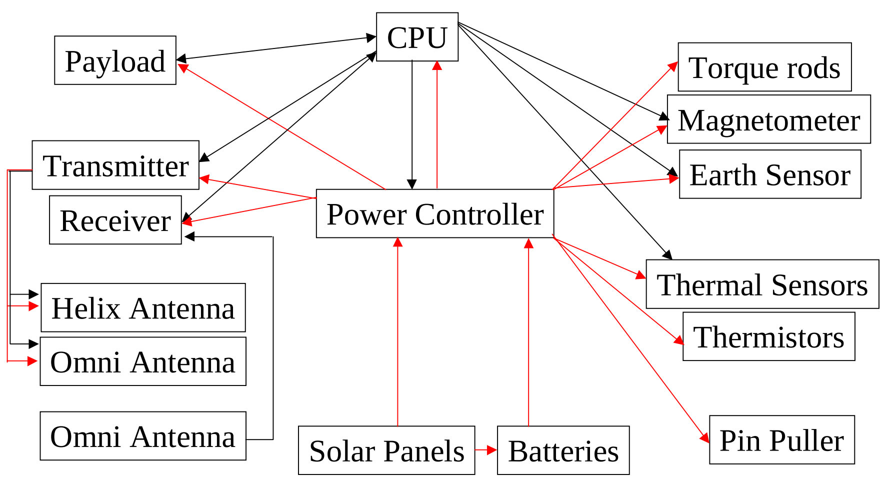


Figure 13.1: Integration Architecture

Figure 13.2 shows the different areas assigned to the different subsystems. Red is the battery packs. Yellow is a combination of the power board and computer where bottom side contains the computer and the top side is the power board. This arrangement allows a short connection between the two. Green is both the transmitter and receiver. By placing them near the front wall, this allows a short connection to the antennas. The payload areas are the empty space to the left and right of the system compartment.

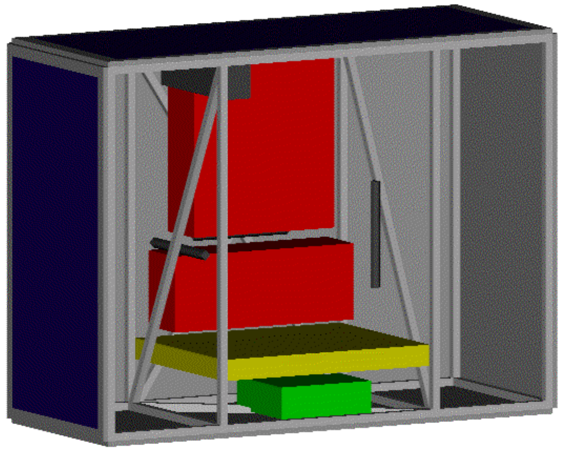


Figure 13.2: Modular Design

The assembly process starts with the string structure. Components will be integrated into the bus once they are received from the manufacturer. Next, the honeycomb skin walls will be set in place with rivets. Then, the solar arrays will be epoxied to the outer walls, leaving the one side open, the side that holds the ring. This provides access to the satellite for last minute testing and necessary part changes.

13.2 Payload

After all component systems are operational the payload box is then integrated into the bus. The customer is responsible for integrating the payload into the payload box. Celestial Concepts will provide a payload integration guide that contains all thermal, structural, and vibrational loading that the satellite will encounter plus and limits the customer must know.

14. TESTING

14.1 Prototype

Three engineering models will verify that the design is mission ready. Each model will undergo extreme vibration, environment, and shock loading. After each test, a functional test of the satellite will be performed to verify that the satellite will work properly. After complete integration, every satellite is spin balanced in order to make sure the satellite is balanced about its principle axes. If the axes are misaligned then ballast weights are placed in specific areas to correct any error.

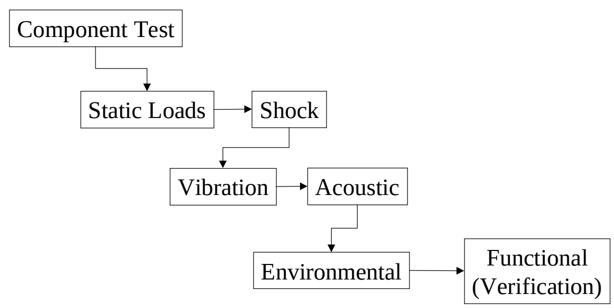


Figure 14.1: Testing Flow Chart

14.2 Satellite

During the assembly of each bus, the satellite's components are functionally tested. This requires that the components are receiving power and sending the correct data. In addition, the boom and helix deployment sequences are performed to assure proper deployment. This involves multiple testing of the switches, omni-directional antennas, and pin-puller. Most testing equipment is leased or rented depending on the frequency of use.

14.3 Payload

The design of LOCO-1 allows easy payload integration by supplying a payload integration box is supplied to the customer. The customer assembles the payload into the box and ships it back to Celestial Concepts where it is integrated in to the bus.

After the final configuration has been determined, the entire mass of the satellite should be 35-45 kg, depending on the payload's mass. LOCO-1 can accommodate a variety of payloads therefore the satellite's weight can fluctuate. An integration pamphlet will be provided so that the customer knows of the specifications regarding payload placement, weight, size, testing, power and computer interfaces. This is the key to better customer support and to speed up assembly process.

15. BUSINESS OBJECTIVES

15.1 Introduction

Celestial Concepts is a small company that hopes to take a profitable share of the micro-satellite market. From the market analysis in Section 2 there is a definite opening for this market. To successfully compete in this market, however, Celestial Concepts must make at least four to five bus sales a year. This amount of sales is equivalent to 10% of the micro-satellite market, as defined earlier. Celestial Concepts must first consider what expenses are necessary to effectively run the company. From there, a cost analysis can be performed to determine a sale price of the bus. The following analysis shows potential business paths that Celestial Concepts might endure.

15.2 Sources of Funding

Celestial Concepts needs a source of cash in order to successfully start off. The usual source for money is through a loan or grant. Since the government always has an interest in scientific research, they would be the primary target for a loan or grant. NASA, the Department of Defense, and the U.S. Air Force frequently distribute funds to companies and universities with a good proposal for a scientific project. The federal government can also distribute small business and educational grants.

The government is not the only group interested in satellite technology though. Private investors, such as other businesses and corporations, can grant money to those willing to start a company (only if seems profitable). Ownership of the company is then through another company. This can be risky since the same company can usurp complete control of their investment and start anew.

15.3 Yearly Budget

A yearly budget helps define the limits of what a company can spend. From the budget, a company can figure out their projected cash profit. The budget includes salaries, rent of office facilities and test equipment, and maintenance. Table 15.1 shows what Celestial Concepts is willing to spend in order to smoothly run all operations. Taxes and interest payments on loans are not included in the budget. All salaries include both income and benefits (retirement, health, and medical bonuses). Celestial Concepts will consist of five engineers, and two technicians. The respective salaries between the engineers and technicians break down to $65K and $37.5K apiece. The office space, assembly room, and testing equipment will be leased from a private company. Vandenburg Air Force base has numerous facilities and equipment that is leased by many companies. Celestial Concepts hopes to get a share of those facilities. Apart from the testing equipment are office supplies (computers, software, etc.) which are also incorporated into the budget. These will be bought, not leased, by the company. With all this equipment in the company's possession, a part of budget will have to go toward maintenance. In the first year Celestial

Concepts will have to spend nearly $710K for the proposed budget. The budget in subsequent years will be less since not as much will be spent on new equipment, maintenance and testing.

Salaries - income - benefits	$400,000
Facilities - office - assembly room	$60,000
Testing	$100,000 for prototype $5,000 for subsequent bus
Equipment - office supplies (computers, software, etc.) - maintenance	$150,000 first year $20,000 subsequent years
Total	$710,000 first year $500,000 subsequent years

Table 15.1: Estimated Yearly Budget

15.4 Subsystem Budget

The yearly budget from above does not include the price of the bus components. Table 15.2 shows the estimated cost for each subsystem. These are the prices of the individual parts within the bus. The overall cost of the system components is about $500K.

15.5 Production Scheduling

In order to determine the company's cash flow throughout the year, a production schedule needs to be implemented. Figure 15.1 shows the production schedule for LOCO-1. The schedule accounts for the shipping and testing time for both the subsystem components and payload. Also included in the schedule is a product review, where Celestial Concepts helps customers integrate the payload into the payload integration box. Looking at the schedule, it is assumed that all units are produced and delivered within a nine-month period.

Power	$92,318
ADC	$290,000
Communications	$40,090
Structures	$2,690
Deployables	$65,000
Thermal	$2,205
Command & Data Handling	$10,000
Total	$502,300

Table 15.2: Estimated Subsystem Budget

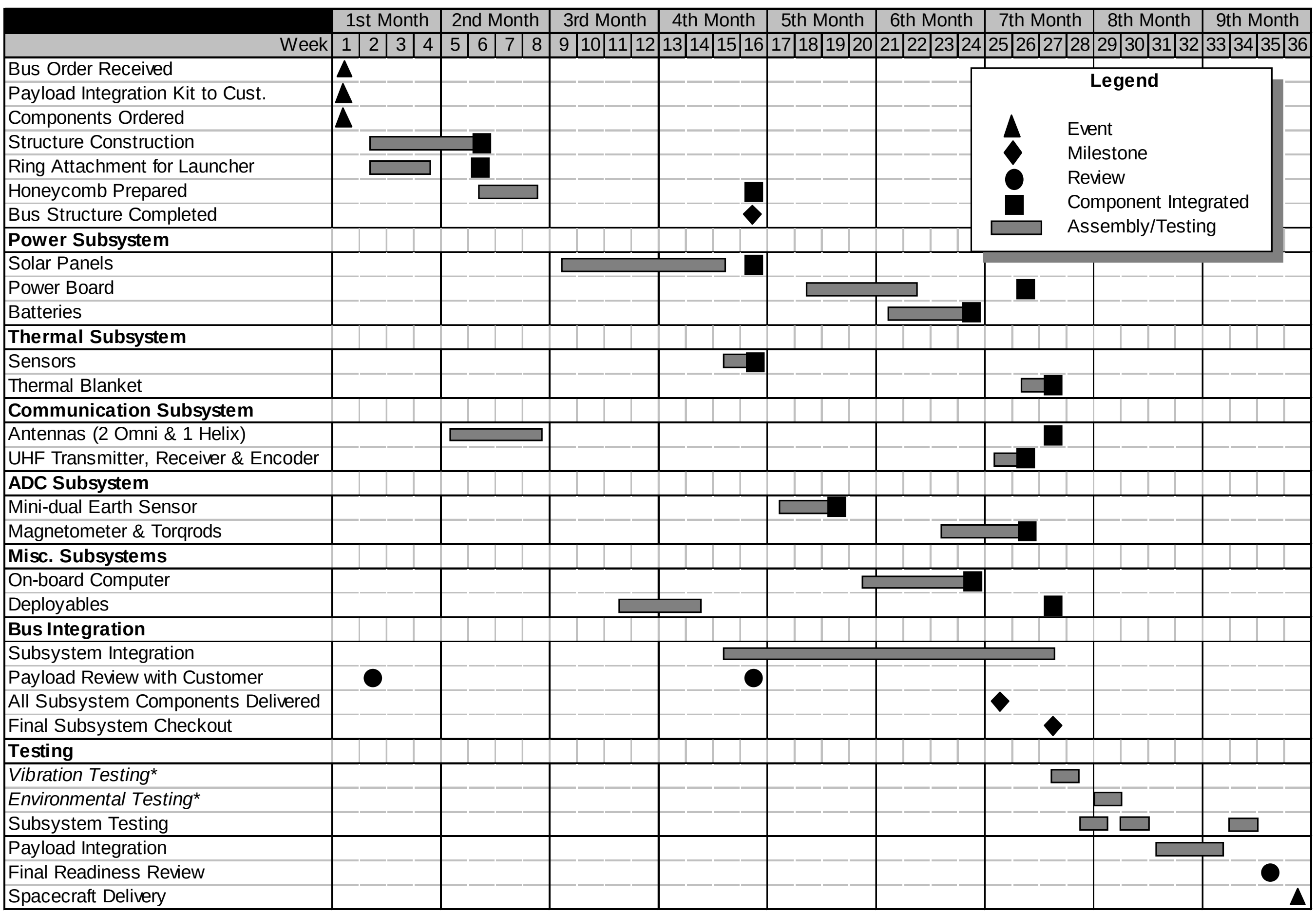


Figure 15.1: Production Schedule for LOCO-1

15.6 Cash Flow

A cash flow graph shows how much the company is worth. **Figure 15.2** shows the company value within a five-year period. The colored lines represent the different methods of payment by the customer. Realistically, no customer will pay upon ordering and payment upon delivery will hurt the company. Therefore the customer is expected to pay half upon ordering and the rest upon delivery of the satellite. During the first cycle of production two buses are ordered, but only one is delivered. The first satellite is a prototype, which will make no money since its sale price is $500K, the same price as the satellite's components. If the prototype is successful then the production cycle continues. The success of the prototype should gain acceptance for LOCO-1. Continuing into the second year, four buses are ordered, but only two are delivered. However, according to the yearly budget, less is being spent on testing and equipment maintenance. In order for Celestial Concepts to break even by the third year, a fully built and delivered LOCO-1 should cost about $775K. After the second year four satellites should be ordered and delivered within the production schedule. If this trend continues then the value of the company breaks even after three years and continues to make money.

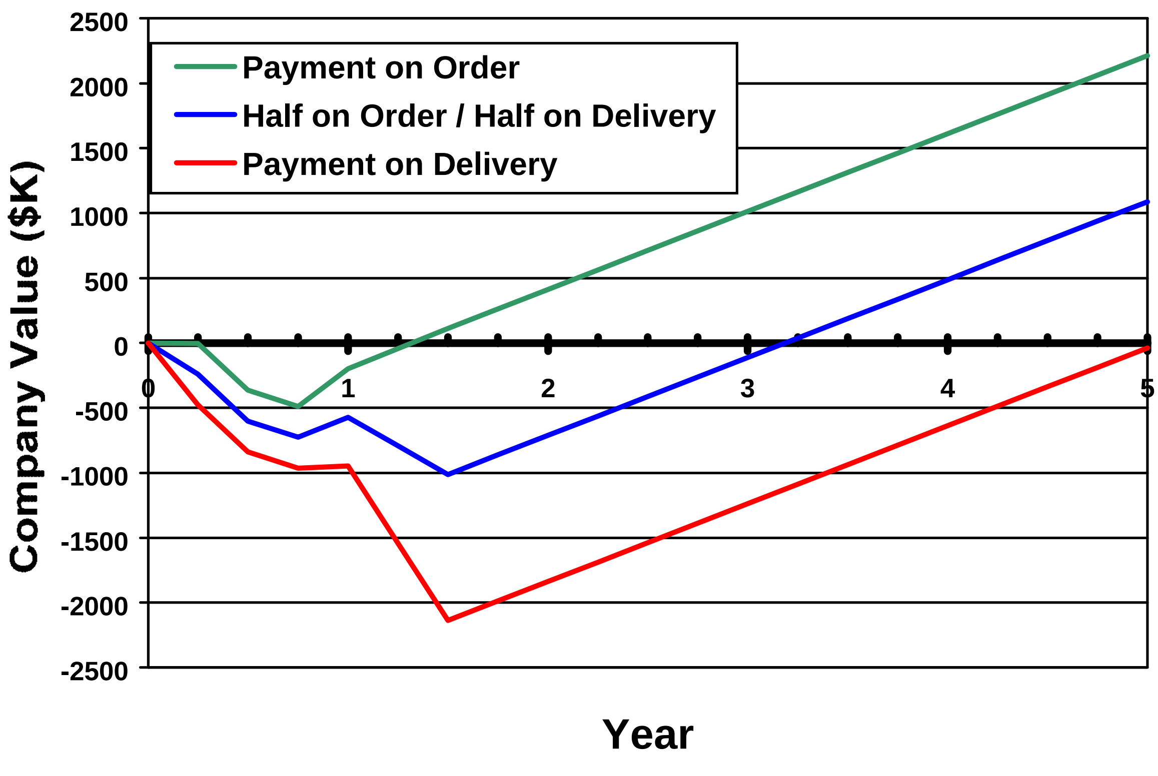


Figure 15.2: Company Value within a Five-Year Period

If, however, the prototype fails, then production is set back a year. **Figure 15.3** shows the effect of a prototype failure. During the first production cycle, the prototype is ordered, but then fails. After this the cycle resumes like the previous. Two satellites are ordered in the next cycle, however only one is delivered: the prototype. From here the cycle continues as before except now the company breaks even the third quarter of the fourth year.

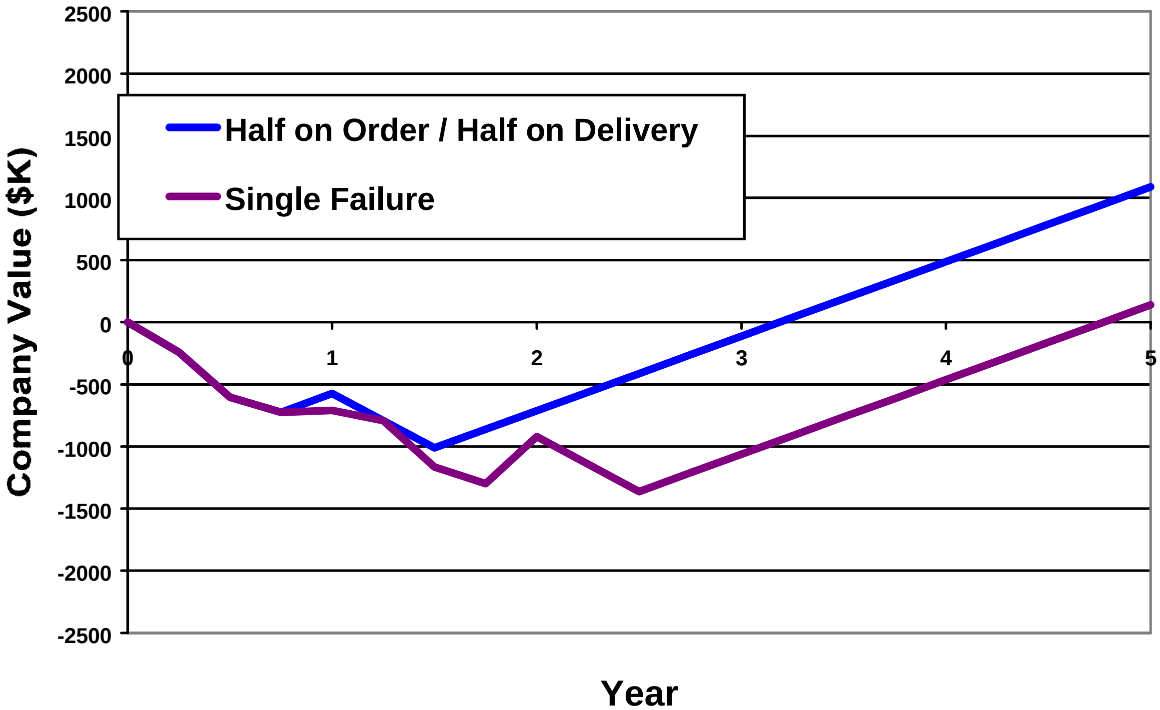
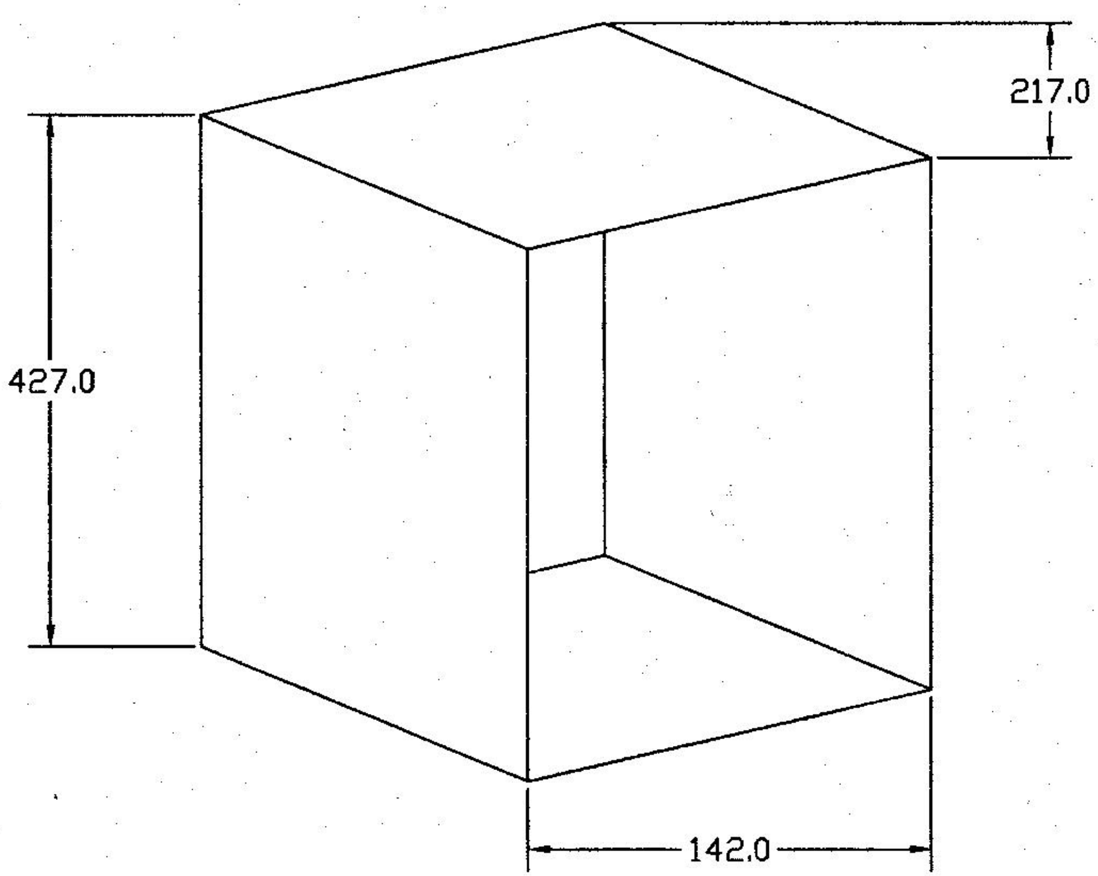


Figure 15.3: Effect of a Prototype Failure

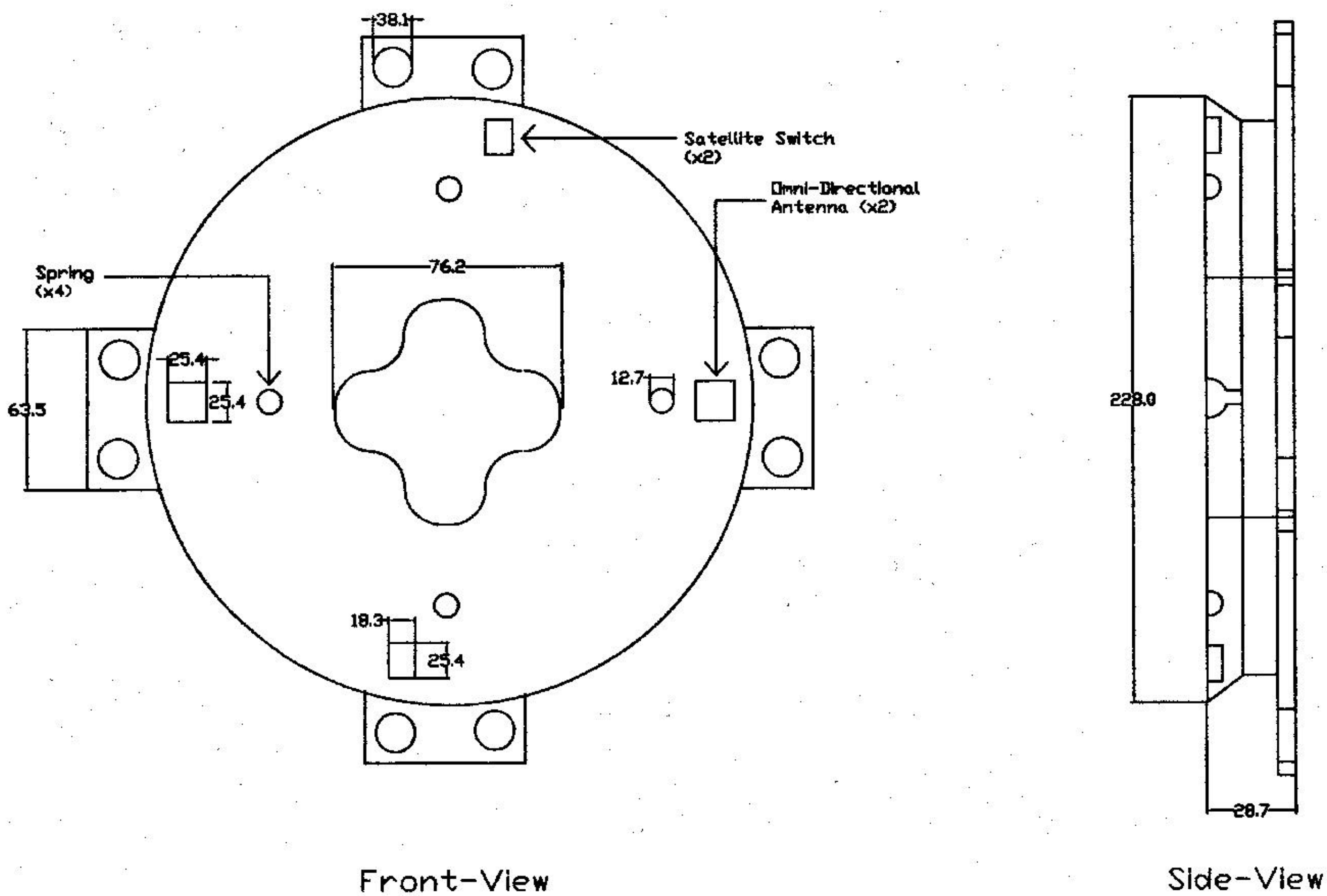
15.7 Conclusion

For the company to start successfully an initial investment of $1M is necessary. After four years Celestial Concepts will be worth $500K, so 50% of the initial investment can be returned in four years. This of course assumes that Celestial Concepts is producing four LOCO-1's a year at a price of $775K. From the market analysis, to gain 10% of the market means four satellites a year. However this projection could change. The company needs to be ready for any rapid market changes and possible failures. Celestial Concepts LOCO-1 satellite has the potential to successfully compete in the world's satellite market.

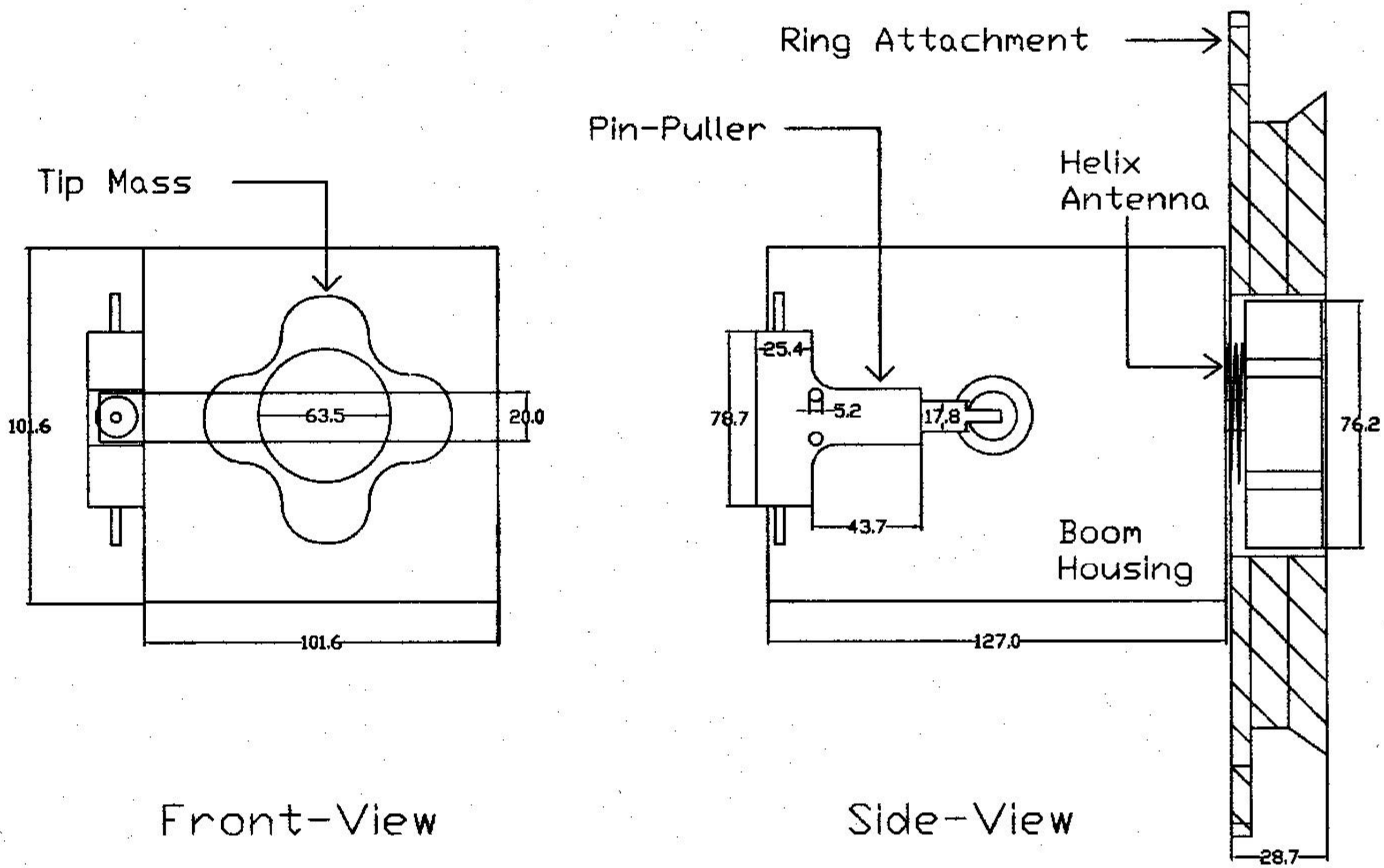
Appendix A: Technical Drawings



Celestial Concepts	Payload Box	Dimensions in millimeters	Drawn by: T. Luzod	California Polytechnic State University
		Date: 4/29/99	Reviewed by: CC	Space Craft Design Team

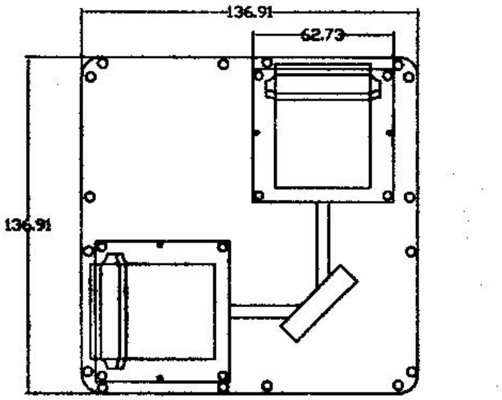


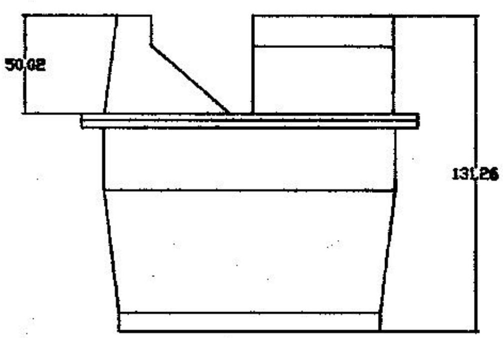
Celestial Concepts	Ring Attachment	Dimensions in millimeters	Drawn by: T. Luzod	California Polytechnic State University
		Date: 5/16/99	Reviewed by: CC	Space Craft Design Team



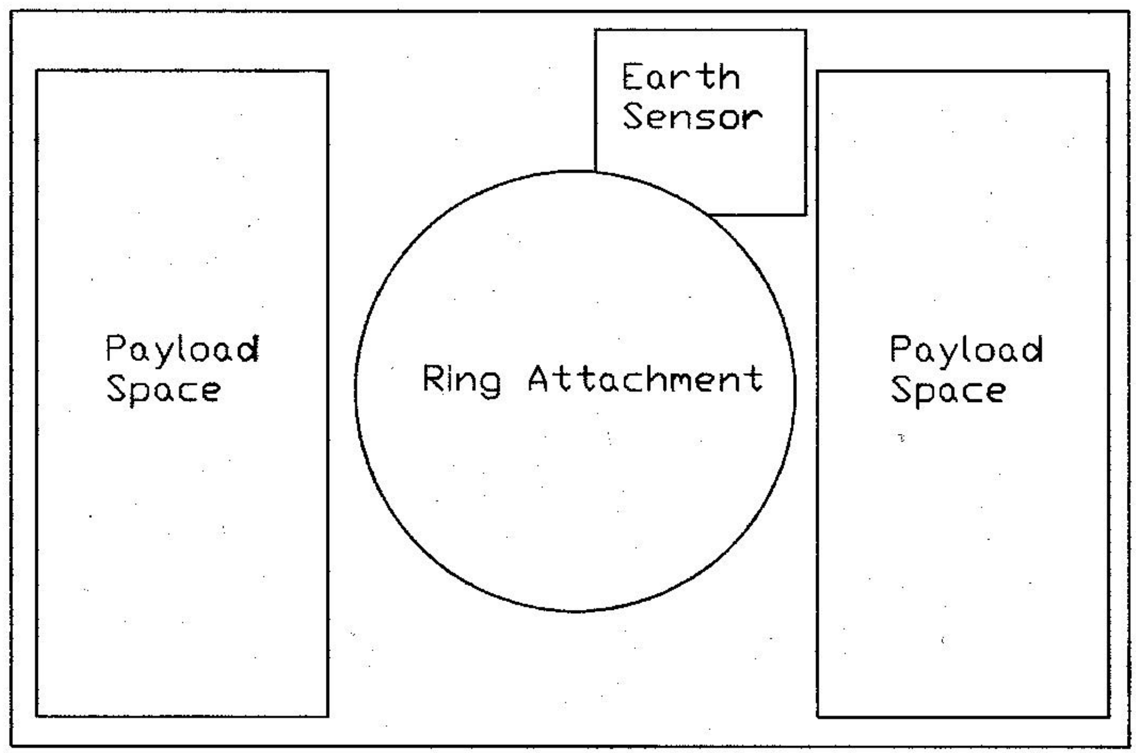
Celestial Concepts	Boom & Housing Stowage	Dimensions in millimeters	Drawn by: T. Luzod	California Polytechnic State University
		Date: 5/16/99	Reviewed by: CC	Space Craft Design Team

Top-View

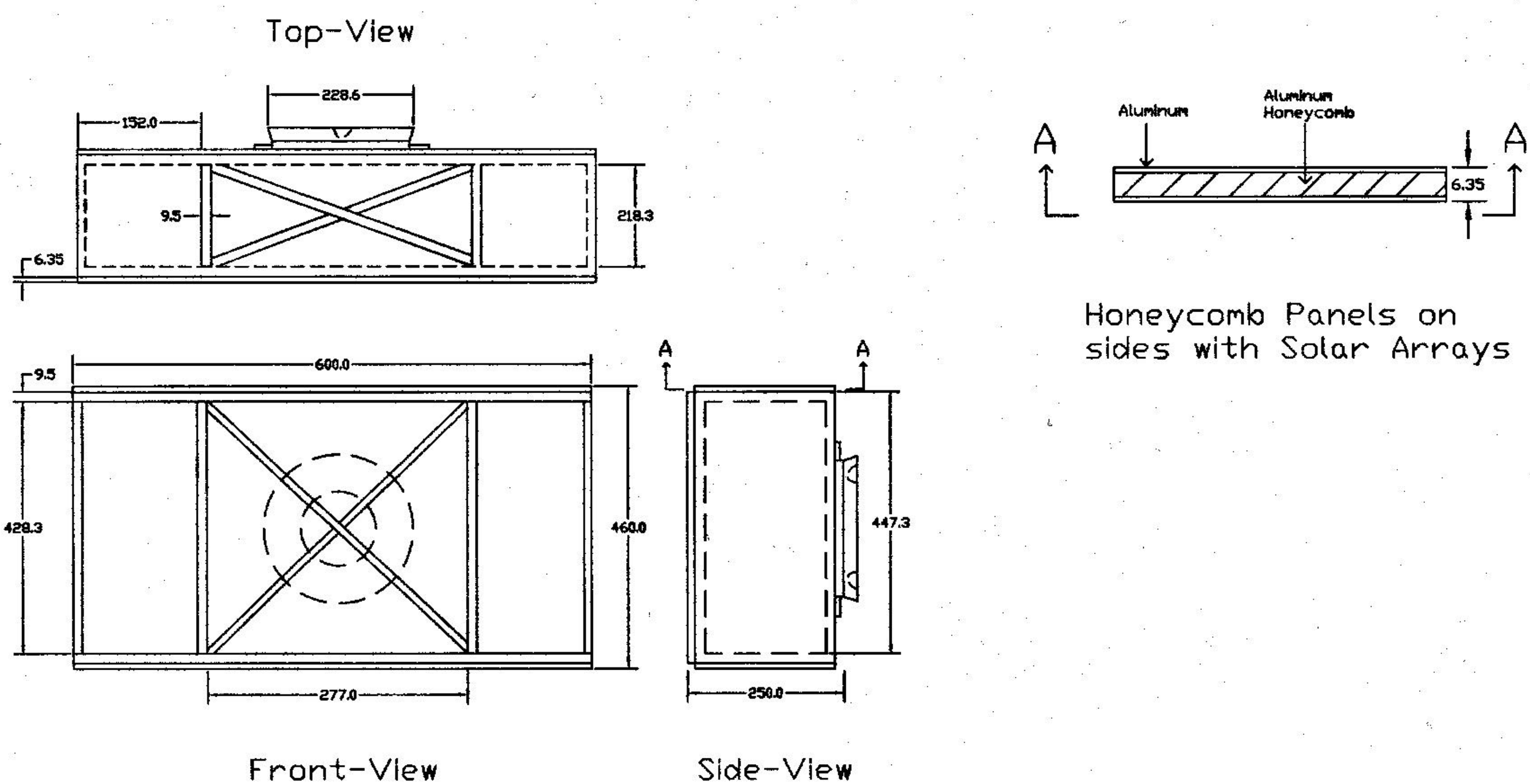




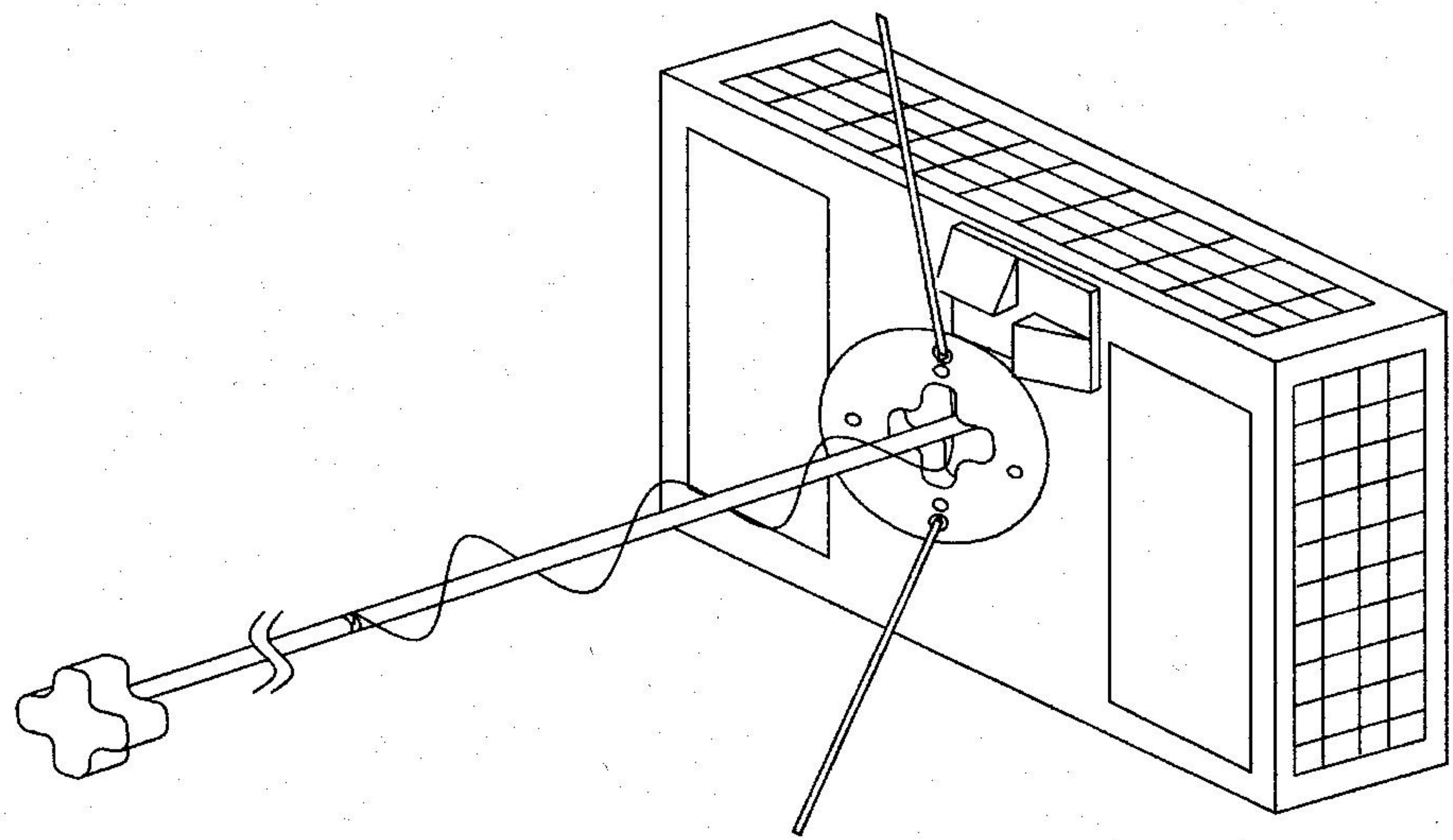
Side-View



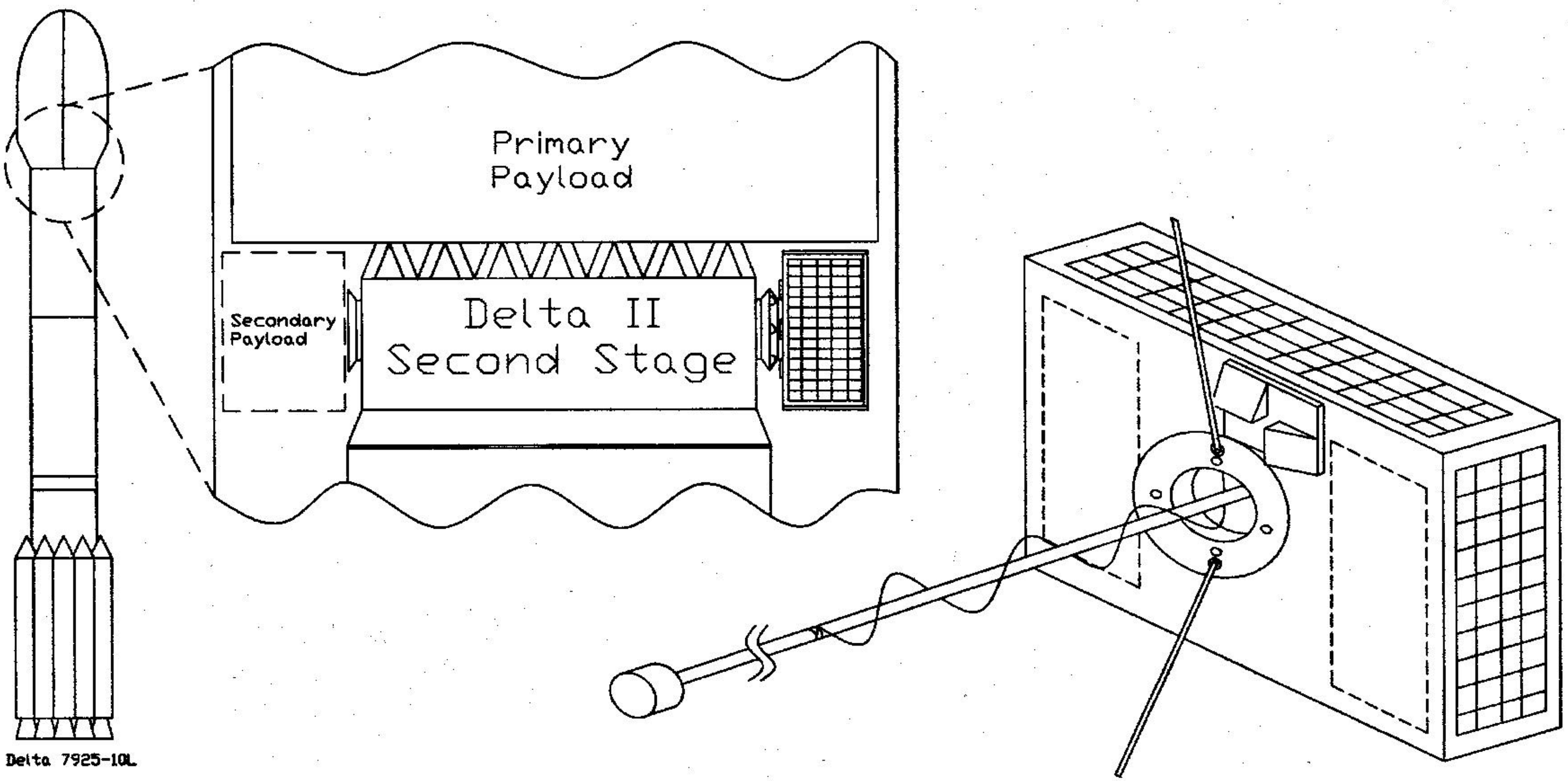
Celestial Concepts	Earth Sensor	Dimensions in millimeters	Drawn by: T. Luzod	California Polytechnic State University
		Date: 5/16/99	Reviewed by: CC	Space Craft Design Team



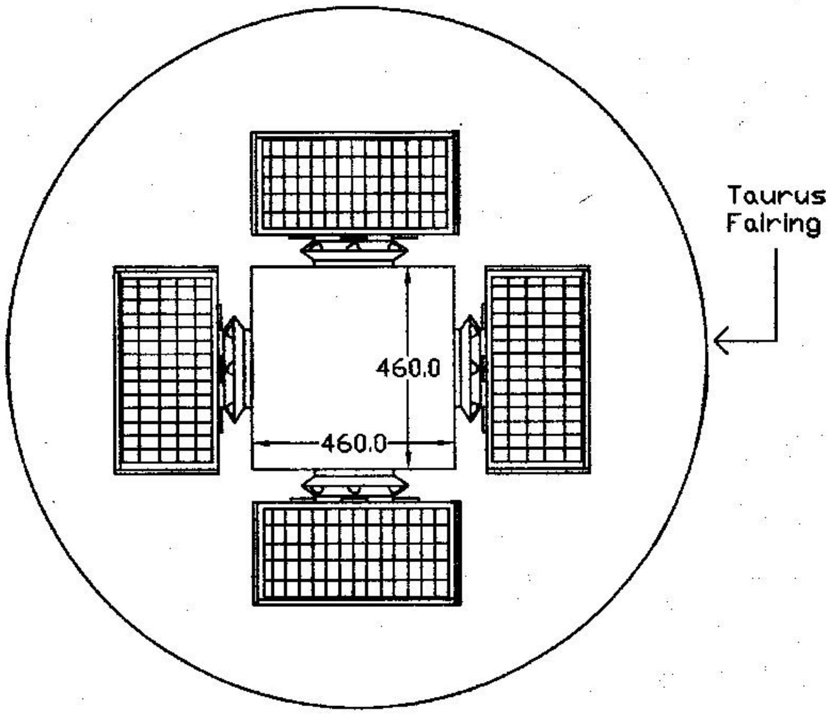
Celestial Concepts	Structure	Dimensions in millimeters	Drawn by: T. Luzod	California Polytechnic State University
		Date: 5/16/99	Reviewed by: CC	Space Craft Design Team



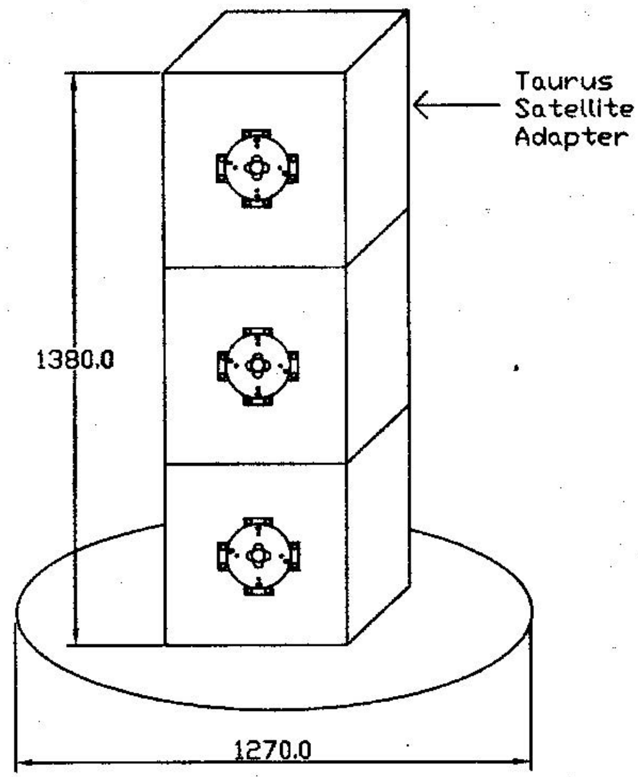
Celestial Concepts	Isometric	Dimensions in millimeters	Drawn by: T. Luzod	California Polytechnic State University
		Date: 5/16/99	Reviewed by: CC	Space Craft Design Team



Celestial Concepts	LOCO SAT	Date: 4/29/99	Drawn by: T. Luzod	California Polytechnic State University
			Reviewed by: CC	Space Craft Design Team



Top-View



Side-View

Celestial Concepts	Taurus Configuration	Dimensions in millimeters	Drawn by: T. Luzod	California Polytechnic State University
		Date: 5/16/99	Reviewed by: CC	Space Craft Design Team

Appendix B: Moments of Inertia and Center of Mass Calculations

Aluminum Siding (Back)			Distance to CM	
I_1 =	0.006625	kg-m^2	x_c =	0.165273
I_2 =	0.002367	kg-m^2	y_c =	0.010370
I_3 =	0.004259	kg-m^2	z_c =	0.002013

Aluminum Siding (Front)			Distance to CM	
I_1 =	0.006624	kg-m^2	x_c =	0.078377
I_2 =	0.002366	kg-m^2	y_c =	0.010370
I_3 =	0.004258	kg-m^2	z_c =	0.002013

Aluminum Siding (Side x2)			Distance to CM (L-R)		
I_1 =	0.000936	kg-m^2	x_c =	0.043448	0.043448
I_2 =	0.001200	kg-m^2	y_c =	0.313545	0.292805
I_3 =	0.000264	kg-m^2	z_c =	0.002013	0.002013

Aluminum Siding (Top x2)			Distance to CM (B-T)		
I_1 =	0.002260	kg-m^2	x_c =	0.043448	0.043448
I_2 =	0.000354	kg-m^2	y_c =	0.010370	0.010370
I_3 =	0.002613	kg-m^2	z_c =	0.228838	0.224812

Truss Members (dia-front L to R)			Distance to CM	
I_{11} =	0.002783	kg-m^2	x_c =	0.157348
I_{12} =	0	kg-m^2	y_c =	0.010370
I_{13} =	0	kg-m^2	z_c =	0.002013
I_{21} =	0	kg-m^2		
I_{22} =	0.001963	kg-m^2		
I_{23} =	-0.001268	kg-m^2		
I_{31} =	0	kg-m^2		
I_{32} =	-0.001268	kg-m^2		
I_{33} =	0.000822	kg-m^2		

Truss Members (dia-front R to L)			Distance to CM	
I_{11} =	0.002783	kg-m^2	x_c =	0.157348
I_{12} =	0	kg-m^2	y_c =	0.010370
I_{13} =	0	kg-m^2	z_c =	0.002013
I_{21} =	0	kg-m^2		
I_{22} =	0.001963	kg-m^2		
I_{23} =	0.001267993	kg-m^2		
I_{31} =	0	kg-m^2		
I_{32} =	0.001267993	kg-m^2		
I_{33} =	0.000822	kg-m^2		

Earth Sensor			Distance to CM	
I_1 =	0.004374	kg-m^2	x_c =	0.065942
I_2 =	0.004197	kg-m^2	y_c =	0.059675
I_3 =	0.004197	kg-m^2	z_c =	0.143682

Magnetometer			Distance to CM	
I_1 =	0.000058	kg-m^2	x_c =	0.011798
I_2 =	0.000488	kg-m^2	y_c =	0.130870
I_3 =	0.000479	kg-m^2	z_c =	0.191137

Torqrods (1)			Distance to CM	
I_1 =	0.000002	kg-m^2	x_c =	0.054648
I_2 =	0.000135	kg-m^2	y_c =	0.142520
I_3 =	0.000135	kg-m^2	z_c =	0.022213

Torqrods (2)			Distance to CM	
I_1 =	0.000135	kg-m^2	x_c =	0.146248
I_2 =	0.000002	kg-m^2	y_c =	0.010370
I_3 =	0.000135	kg-m^2	z_c =	0.022213

Solar Panels (Back)			Distance to CM	
I_1 =	0.009004	kg-m^2	x_c =	0.168535
I_2 =	0.003241	kg-m^2	y_c =	0.010370
I_3 =	0.005762	kg-m^2	z_c =	0.002013

Solar Panels (Side x 2)			Distance to CM (L-R)		
I_1 =	0.002161	kg-m^2	x_c =	0.043448	0.043448
I_2 =	0.002701	kg-m^2	y_c =	0.316808	0.296067
I_3 =	0.000540	kg-m^2	z_c =	0.002013	0.002013

Solar Panels (Top x 2)			Distance to CM (B-T)		
I_1 =	0.005762	kg-m^2	x_c =	0.043448	0.043448
I_2 =	0.000810	kg-m^2	y_c =	0.010370	0.010370
I_3 =	0.006573	kg-m^2	z_c =	0.232101	0.228074

Truss Members (dia-top L to R)			Distance to CM	
I_{11} =	5.681442E-04	kg-m^2	x_c =	0.043448
I_{12} =	4.466953E-04	kg-m^2	y_c =	0.010370
I_{13} =	0	kg-m^2	z_c =	0.216887
I_{21} =	4.466953E-04	kg-m^2		
I_{22} =	3.533689E-04	kg-m^2		
I_{23} =	0	kg-m^2		
I_{31} =	0	kg-m^2		
I_{32} =	0	kg-m^2		
I_{33} =	9.201788E-04	kg-m^2		

Truss Members (dia-top R to L)			Distance to CM	
I_{11} =	5.681442E-04	kg-m^2	x_c =	0.043448
I_{12} =	-4.46695E-04	kg-m^2	y_c =	0.010370015
I_{13} =	0	kg-m^2	z_c =	0.220913
I_{21} =	-4.46695E-04	kg-m^2		
I_{22} =	3.533689E-04	kg-m^2		
I_{23} =	0	kg-m^2		
I_{31} =	0	kg-m^2		
I_{32} =	0	kg-m^2		
I_{33} =	9.201788E-04	kg-m^2		

Bus (full) MOI (cont.)			Bus (deployed) MOI		
I_{11} =	1.854249	kg-m^2	I_{11} =	1.857067	kg-m^2
I_{12} =	-0.000080	kg-m^2	I_{12} =	-0.000080	kg-m^2
I_{13} =	-0.000021	kg-m^2	I_{13} =	-0.000021	kg-m^2
I_{21} =	-0.000080	kg-m^2	I_{21} =	-0.000080	kg-m^2
I_{22} =	0.869457	kg-m^2	I_{22} =	53.336330	kg-m^2
I_{23} =	-0.000005	kg-m^2	I_{23} =	-0.000005	kg-m^2
I_{31} =	-0.000021	kg-m^2	I_{31} =	-0.000021	kg-m^2
I_{32} =	-0.000005	kg-m^2	I_{32} =	-0.000005	kg-m^2
I_{33} =	1.493704	kg-m^2	I_{33} =	53.960741	kg-m^2

m_{boom}	0.072576	kg	T_g =	1.000381E-04	N-m	
l_{boom}	3.949	m	T_a =	3.532372E-10	N-m	
m_{tip}	3.510	kg	T_m =	5.112243E-05	N-m	
			T_{sp} =	4.97255E-12	N-m	
h =	400000		y_{pa} =	0.010370		0.010370
θ =	0.0001		z_{pa} =	0.002013		0.002013
ρ =	1.05E-11					
S =	0.148631		c_p =	0.000005		

Torqrods (3)			Distance to CM	
I_1 =	0.000135	kg-m^2	x_c =	0.054648
I_2 =	0.000135	kg-m^2	y_c =	0.121780
I_3 =	0.000002	kg-m^2	z_c =	0.022213

Ring Attachment			Distance to CM	
I_1 =	0.079198	kg-m^2	x_c =	0.098062
I_2 =	0.010224	kg-m^2	y_c =	0.010370
I_3 =	0.010224	kg-m^2	z_c =	0.002013

Truss Members (vertical x 8)			Distance to CM (L-RM, B-F)		
I_1 =	0.001648	kg-m^2	x_c =	0.157348	0.070452
I_2 =	0.001648	kg-m^2	y_c =	0.305620	0.305620
I_3 =	1.620453E-06	kg-m^2	z_c =	0.002013	0.002013
			x_c =	0.157348	0.070452
Distance to CM (R, B-F)			y_c =	0.153620	0.153620
x_c =	0.157348	0.070452	z_c =	0.002013	0.002013
y_c =	0.284880	0.284880	x_c =	0.157348	0.070452
z_c =	0.002013139	0.002013	y_c =	0.132880	0.132880
			z_c =	0.002013	0.002013

Truss Members (hor-front x 4)			Distance to CM (B, B-F; T, B-F)		
I_1 =	0.004529	kg-m^2	x_c =	0.157348	0.070452
I_2 =	2.270071E-06	kg-m^2	y_c =	0.010370	0.010370
I_3 =	0.004529	kg-m^2	z_c =	0.220913	0.216887
			x_c =	0.157348	0.070452
			y_c =	0.010370	0.010370
			z_c =	0.220913	0.216887

Truss Members (hor-side x 8)			Distance to CM		
I_1 =	8.259276E-07	kg-m^2	x_c =	0.043448	0.043448
I_2 =	0.000218	kg-m^2	y_c =	0.305620	0.305620
I_3 =	0.000218	kg-m^2	z_c =	0.220913	0.216887
			x_c =	0.043448	0.043448
Distance to CM			y_c =	0.153620	0.153620
x_c =	0.043448	0.043448	z_c =	0.220913	0.216887
y_c =	0.284880	0.284880	x_c =	0.043448	0.043448
z_c =	0.220913	0.216887	y_c =	0.132880	0.132880
			z_c =	0.220913	0.216887

Truss Members (dia-side L to R)			Distance to CM	
I_{11} =	0.001850	kg-m^2	x_c =	0.043448
I_{12} =	0	kg-m^2	y_c =	0.153620
I_{13} =	0.000942	kg-m^2	z_c =	0.002013
I_{21} =	0	kg-m^2		
I_{22} =	0.002330	kg-m^2		
I_{23} =	0	kg-m^2		
I_{31} =	0.000942	kg-m^2		
I_{32} =	0	kg-m^2		
I_{33} =	0.000482	kg-m^2		

Truss Members (dia-side R to L)			Distance to CM	
I_{11} =	0.001850	kg-m^2	x_c =	0.043448
I_{12} =	0	kg-m^2	y_c =	0.132880
I_{13} =	-0.000942	kg-m^2	z_c =	0.002013
I_{21} =	0	kg-m^2		
I_{22} =	0.002330	kg-m^2		
I_{23} =	0	kg-m^2		
I_{31} =	-0.000942	kg-m^2		
I_{32} =	0	kg-m^2		
I_{33} =	0.000482	kg-m^2		

Transmitter			Distance to CM	
I_1 =	0.000340	kg-m^2	x_c =	0.042248
I_2 =	0.000195	kg-m^2	y_c =	0.010370
I_3 =	0.000448	kg-m^2	z_c =	0.199148

Receiver			Distance to CM	
I_1 =	0.000681	kg-m^2	x_c =	0.027602
I_2 =	0.000309	kg-m^2	y_c =	0.010370
I_3 =	0.000866	kg-m^2	z_c =	0.197113

ρ_{as}	8.329600E-08	kg/mm^3		
$m_{AS(back)}$	0.141954	kg		
$m_{AS(side)}$	0.056143	kg		
$m_{AS(top)}$	0.075309	kg		
$m_{AS(ring\ cut)}$	2.412113E-03	kg		
ρ_{tm}	2.787052E-06	kg/mm^3		
$m_{tm(vert)}$	0.107731	kg		
$m_{tm(hor\text{-}front)}$	0.150919	kg		
$m_{tm(hor\text{-}side)}$	0.054909	kg		
$m_{tm(dia\text{-}top)}$	0.088710	kg		
$m_{tm(dia\text{-}side)}$	0.120917	kg		
$m_{tm(dia\text{-}front)}$	0.128298	kg		
$m_{sp(back)}$	0.220500	kg		
$m_{sp(side)}$	0.147000	kg		
$m_{sp(top)}$	0.220500	kg		
m_{ra}	2.098418	kg	Σmr	m
$CM_{1(bus)}$	0.168623	m	6.7409	39.976267
$CM_{2(bus)}$	0.316895	m	12.66828	39.976267
$CM_{3(bus)}$	0.232188	m	9.282015	39.976267
m_{es}	1.400000	kg		
m_{mag}	0.227000	kg		
m_{torq}	0.100000	kg	Σmr	m
$CM_{1(deploy)}$	0.520168	m	20.79436	39.976267
$CM_{2(deploy)}$	0.316895	m	12.66828	39.976267
$CM_{3(deploy)}$	0.232188	m	9.282015	39.976267
m_{tot}	36.393691	kg		
x_c =	0.351545	m		
y_c =	0.000000	m	Distance from CM_{bus} to CM_{deploy}	
z_c =	0.000000	m		
m_{rec}	0.510300	kg		
m_{trans}	0.450000	kg		
m_{batt}	1.800000	kg		
$m_{payload(l)}$	10.000000	kg		
$m_{payload(r)}$	10.000000	kg		
$m_{boombox(pre)}$	2.000000	kg		
$m_{boombox(post}$	1.927424	kg		

Batteries (short x 2)			Distance to CM		
I_1 =	0.007308	kg-m^2	x_c =	0.120098	0.120098
I_2 =	0.002583	kg-m^2	y_c =	0.010370	0.01037
I_3 =	0.005992	kg-m^2	z_c =	0.041137	0.155137

Battery (long)			Distance to CM	
I_1 =	0.009346	kg-m^2	x_c =	0.120098
I_2 =	0.001876	kg-m^2	y_c =	0.010370
I_3 =	0.009346	kg-m^2	z_c =	0.081763

Boom Box (post-deployment)			Distance to CM	
I_1 =	0.003316	kg-m^2	x_c =	0.333493
I_2 =	0.004249	kg-m^2	y_c =	0.010370
I_3 =	0.004249	kg-m^2	z_c =	0.002013

Boom Box (pre-deployment)			Distance to CM	
I_1 =	0.003441	kg-m^2	x_c =	0.018052
I_2 =	0.004409	kg-m^2	y_c =	0.010370
I_3 =	0.004409	kg-m^2	z_c =	0.002013

Boom			Distance to CM	
I_1 =	0.000003	kg-m^2	x_c =	1.704332
I_2 =	0.094318	kg-m^2	y_c =	0.010370
I_3 =	0.094318	kg-m^2	z_c =	0.002013

Tip Mass			Distance to CM	
I_1 =	0.002548	kg-m^2	x_c =	3.685182
I_2 =	0.001321	kg-m^2	y_c =	0.010370
I_3 =	0.001321	kg-m^2	z_c =	0.002013

Payload (left)			Distance to CM	
I_1 =	0.169789	kg-m^2	x_c =	0.043448
I_2 =	0.192580	kg-m^2	y_c =	0.229620
I_3 =	0.056634	kg-m^2	z_c =	0.011513

Payload (right)			Distance to CM	
I_1 =	0.169789	kg-m^2	x_c =	0.043448
I_2 =	0.192580	kg-m^2	y_c =	0.208880
I_3 =	0.056634	kg-m^2	z_c =	0.011513

Appendix C: Determining Helix Antenna Dimensions

APPENDIX C

DETERMINING HELIX ANTENNA DIMENSIONS

Using the following two interdependant equations, the antenna dimensions were determined, along with the performance parameters:

Gain (dB) = $10.3+10\log(C^2L/\lambda^3)$ [Eqn 6.1]

HPBW (degrees) = $52/(C^2L/\lambda^3)$ [Eqn 6.2]

HPBW	Wavelength	Circum.	Length	Diameter	Gain	Turn Angle	Spacing	# of turns
degrees	m	m	m	m	Db	degrees	m	
10	0.1	0.367696	0.2	0.117041	10.491096	13	0.04499	4.445411
20	0.15	0.275772	0.3	0.087781	10.609466	13	0.067485	4.445411
30	0.2	0.24513	0.4	0.078027	10.726384	13	0.08998	4.445411
40	0.25	0.22981	0.5	0.073151	10.84185	13	0.112476	4.445411
50	0.3	0.220617	0.6	0.070225	10.955864	13	0.134971	4.445411
60	0.35	0.214489	0.7	0.068274	11.068426	13	0.157466	4.445411
70	0.4	0.210112	0.8	0.066881	11.179536	13	0.179961	4.445411
80	0.45	0.206829	0.9	0.065836	11.289194	13	0.202456	4.445411
90	**0.5**	**0.204275**	**1**	**0.065023**	**11.3974**	**13**	**0.224951**	**4.445411**
100	0.55	0.202233	1.1	0.064373	11.504154	13	0.247446	4.445411
110	0.6	0.200561	1.2	0.063841	11.609456	13	0.269941	4.445411
120	0.65	0.199168	1.3	0.063397	11.713306	13	0.292436	4.445411

Appendix D: Link Budget Equations

APPENDIX D

LINK BUDGET EQUATIONS

P = transmitter power in Watts:

$P[dBW]=10 \log (P[W])$

Gpt = peak transmit antenna gain [Eqn 6.3]

$Gpt = 10.25+1.22*(D)-0.0726*(D)^2$

D = helix antenna diameter (m)

L_θ = transmit antenna pointing loss (dB) [Eqn 6.4]

$L_\theta = -12(e/\theta)^2$

e = error fraction of beam-width

Ll = line loss (dB)

Gt = transmit antenna gain (dB)

$Gt = Gpt + L_\theta$

EIRP = Equivalent Isotropic Radiated Power (dBW)

$EIRP = P+Ll+Gt$

Ls = space loss (dB)

$Ls = 147.55-20\log S-20\log f$ [Eqn 6.5]

S = path length (m)

f = frequency (Hz)

La = propagation loss (dB)

Grp = peak receive antenna gain (dB)

$Grp = -159.59+20 \log D + 20 \log f +10 \log \eta$ [Eqn 6.6]

D = receiver antenna diameter (m)

f = frequency (Hz)

η = antenna efficiency

APPENDIX D (CONT.)

LINK BUDGET EQUATIONS

θr = receive antenna beam width

$\theta r = (21/f_{GHz}D)$ [Eqn 6.7]

f = frequency (GHz)

D = receiver antenna diameter (m)

L_{pr} = Receive antenna pointing loss (dB)

$L_{pr} = L_{\theta} = -12(e/\theta)^2$ [Eqn 6.8]

e = error fraction of beam-width

Gr = receive antenna gain (dB)

Gr = Grp + Lpr

Ts = system noise temperature (K)

Eb/No = energy per bit noise density ratio

Eb/No = P + Ll + Gt + Ls + La + Gr + 228.6+10 log Ts – 10 log R [Eqn 6.9]

Eb/No, Ll , Gt, Ls, La, and Gr are in dB, P is in dBW, Ts in K, R in BPS

C/No = Carrier to noise density ratio (dB)

C/No = (Eb/No)+10 log RC/No [Eqn 6.10]

Appendix E: STK Simulations

APPENDIX E

STK SIMULATIONS
HELIX ANTENNA DOWNLOAD PASSES

WEST-400 KM

GROUND STATION: CAL POLY-LOCO-1 AT 56 DEGREES INCLINATION- ALTITUDE: 400 KM

Satellite-LOCO1-Sensor-LOCOsensor-To-Facility-CalPoly-Sensor-CPsensor: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	1 May 1999 03:39:28.52	1 May 1999 03:41:29.55	121.034
2	1 May 1999 18:39:58.95	1 May 1999 18:40:57.15	58.199
3	3 May 1999 18:29:03.19	3 May 1999 18:30:49.05	105.861
4	4 May 1999 02:35:46.20	4 May 1999 02:37:47.02	120.825
5	4 May 1999 17:36:11.18	4 May 1999 17:37:19.75	68.568
6	6 May 1999 17:25:22.66	6 May 1999 17:27:04.96	102.305
7	7 May 1999 01:32:04.02	7 May 1999 01:34:04.32	120.295
8	7 May 1999 16:32:24.37	7 May 1999 16:33:41.40	77.028

Global Statistics

Min Duration	2	1 May 1999 18:39:58.95	1 May 1999 18:40:57.15	58.199
Max Duration	1	1 May 1999 03:39:28.52	1 May 1999 03:41:29.55	121.034
Mean Duration			96.764	
Total Duration			774.116	

GROUND STATION: CAL POLY-LOCO-1 AT 66 DEGREES INCLINATION- ALTITUDE: 400 KM

Satellite-Satellite1-Sensor-helix-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 17:07:39.58	21 May 1999 17:09:38.91	119.332
2	22 May 1999 02:50:21.07	22 May 1999 02:51:41.25	80.186
3	23 May 1999 01:58:50.69	23 May 1999 01:59:38.84	48.152
4	24 May 1999 16:07:20.63	24 May 1999 16:09:12.36	111.735
5	25 May 1999 01:49:41.28	25 May 1999 01:51:36.69	115.410
6	26 May 1999 15:59:06.64	26 May 1999 16:00:14.06	67.422
7	27 May 1999 15:07:21.55	27 May 1999 15:08:27.38	65.828

Global Statistics

Min Duration	3	23 May 1999 01:58:50.69	23 May 1999 01:59:38.84	48.152
Max Duration	1	21 May 1999 17:07:39.58	21 May 1999 17:09:38.91	119.332
Mean Duration			86.867	
Total Duration			608.066	

GROUND STATION: CAL POLY-LOCO-1 AT 76 DEGREES INCLINATION- ALTITUDE: 400 KM

Satellite-Satellite1-Sensor-helix-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 03:42:13.64	21 May 1999 03:42:58.37	44.728
2	22 May 1999 16:17:29.78	22 May 1999 16:19:16.69	106.911
3	24 May 1999 02:43:50.33	24 May 1999 02:45:44.61	114.280
4	24 May 1999 16:10:13.62	24 May 1999 16:12:02.15	108.529
5	26 May 1999 02:36:43.02	26 May 1999 02:38:20.55	97.528
6	27 May 1999 15:12:42.53	27 May 1999 15:13:55.22	72.687

Global Statistics

Min Duration	1	21 May 1999 03:42:13.64	21 May 1999 03:42:58.37	44.728
Max Duration	3	24 May 1999 02:43:50.33	24 May 1999 02:45:44.61	114.280
Mean Duration			90.777	
Total Duration			544.663	

GROUND STATION: CAL POLY-LOCO-1 AT 86 DEGREES INCLINATION- ALTITUDE: 400 KM

Satellite-Satellite1-Sensor-helix-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 03:41:57.80	21 May 1999 03:43:28.73	90.922
2	22 May 1999 16:18:12.79	22 May 1999 16:19:44.02	91.232
3	23 May 1999 03:35:13.80	23 May 1999 03:37:05.07	111.272
4	24 May 1999 16:12:05.40	24 May 1999 16:12:44.28	38.879
5	25 May 1999 03:28:36.64	25 May 1999 03:30:34.04	117.398
6	27 May 1999 03:22:05.12	27 May 1999 03:23:56.83	111.711

Global Statistics

Min Duration	4	24 May 1999 16:12:05.40	24 May 1999 16:12:44.28	38.879
Max Duration	5	25 May 1999 03:28:36.64	25 May 1999 03:30:34.04	117.398

APPENDIX E
STK SIMULATIONS (CONT.)

Mean Duration	93.569
Total Duration	561.415

GROUND STATION: CAL POLY-LOCO-1 AT 96 DEGREES INCLINATION- ALTITUDE: 400 KM

Satellite-Satellite1-Sensor-helix-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 15:35:21.45	21 May 1999 15:36:41.75	80.295
2	22 May 1999 04:24:43.88	22 May 1999 04:26:38.34	114.460
3	24 May 1999 04:18:05.86	24 May 1999 04:20:00.98	115.123
4	26 May 1999 04:11:32.99	26 May 1999 04:13:18.02	105.034

Global Statistics

Min Duration	1	21 May 1999 15:35:21.45	21 May 1999 15:36:41.75	80.295
Max Duration	3	24 May 1999 04:18:05.86	24 May 1999 04:20:00.98	115.123
Mean Duration			103.728	
Total Duration			414.912	

GROUND STATION: CAL POLY-LOCO-1 AT 104 DEGREES INCLINATION- ALTITUDE: 400 KM

Satellite-Satellite1-Sensor-helix-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 15:34:52.42	21 May 1999 15:36:40.65	108.224
2	23 May 1999 05:06:37.07	23 May 1999 05:08:05.58	88.502
3	23 May 1999 15:27:43.02	23 May 1999 15:29:26.75	103.734
4	25 May 1999 04:59:13.06	25 May 1999 05:01:07.15	114.091
5	26 May 1999 16:02:41.35	26 May 1999 16:03:55.53	74.176
6	27 May 1999 04:52:33.78	27 May 1999 04:53:20.87	47.092

Global Statistics

Min Duration	6	27 May 1999 04:52:33.78	27 May 1999 04:53:20.87	47.092
Max Duration	4	25 May 1999 04:59:13.06	25 May 1999 05:01:07.15	114.091
Mean Duration			89.303	
Total Duration			535.818	

APPENDIX E

STK SIMULATIONS (CONT.)

WEST-700 KM

GROUND STATION: CAL POLY-LOCO-1 AT 56 DEGREES INCLINATION- ALTITUDE: 700 KM

15 Apr 1999 06:37:07

Satellite-Satellite1-Transmitter-Transmitter1-To-Facility-Facility1-Receiver-Receiver1: Access Summary Report

Transmitter1-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 00:32:48.75	1 Jul 1998 00:46:11.43	802.674
	2	1 Jul 1998 13:13:22.31	1 Jul 1998 13:23:48.93	626.615
	3	1 Jul 1998 14:53:08.32	1 Jul 1998 15:07:50.45	882.136
	4	1 Jul 1998 16:37:03.20	1 Jul 1998 16:50:13.09	789.897
	5	1 Jul 1998 18:23:20.02	1 Jul 1998 18:33:03.87	583.854
	6	1 Jul 1998 20:08:15.15	1 Jul 1998 20:18:31.47	616.311
	7	1 Jul 1998 21:50:50.89	1 Jul 1998 22:04:33.83	822.940
	8	1 Jul 1998 23:33:20.39	1 Jul 1998 23:47:50.73	870.338

Global Statistics

Min Duration	5	1 Jul 1998 18:23:20.02	1 Jul 1998 18:33:03.87	583.854
Max Duration	3	1 Jul 1998 14:53:08.32	1 Jul 1998 15:07:50.45	882.136
Mean Duration				749.346
Total Duration				5994.764

GROUND STATION: CAL POLY-LOCO-1 AT 66 DEGREES INCLINATION- ALTITUDE: 700 KM

15 Apr 1999 06:40:29

Satellite-Satellite2-Transmitter-Transmitter2-To-Facility-Facility1-Receiver-Receiver1: Access Summary Report

Transmitter2-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 00:32:08.76	1 Jul 1998 00:46:26.10	857.342
	2	1 Jul 1998 02:17:15.01	1 Jul 1998 02:22:14.10	299.091
	3	1 Jul 1998 13:13:31.55	1 Jul 1998 13:26:02.27	750.723
	4	1 Jul 1998 14:53:47.42	1 Jul 1998 15:08:08.12	860.698
	5	1 Jul 1998 16:39:10.73	1 Jul 1998 16:48:18.47	547.741
	6	1 Jul 1998 21:53:57.21	1 Jul 1998 22:04:18.27	621.064
	7	1 Jul 1998 23:34:23.64	1 Jul 1998 23:48:54.94	871.299

Global Statistics

Min Duration	2	1 Jul 1998 02:17:15.01	1 Jul 1998 02:22:14.10	299.091
Max Duration	7	1 Jul 1998 23:34:23.64	1 Jul 1998 23:48:54.94	871.299
Mean Duration				686.851
Total Duration				4807.957

GROUND STATION: CAL POLY-LOCO-1 AT 76 DEGREES INCLINATION- ALTITUDE: 700 KM

15 Apr 1999 06:42:48

Satellite-Satellite3-Transmitter-Transmitter3-To-Facility-Facility1-Receiver-Receiver1: Access Summary Report

Transmitter3-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 00:32:07.15	1 Jul 1998 00:46:31.56	864.408
	2	1 Jul 1998 02:14:01.94	1 Jul 1998 02:23:55.38	593.442
	3	1 Jul 1998 13:13:43.82	1 Jul 1998 13:27:23.60	819.782
	4	1 Jul 1998 14:54:26.29	1 Jul 1998 15:07:33.16	786.875
	5	1 Jul 1998 23:35:58.27	1 Jul 1998 23:49:25.51	807.236

Global Statistics

Min Duration	2	1 Jul 1998 02:14:01.94	1 Jul 1998 02:23:55.38	593.442
Max Duration	1	1 Jul 1998 00:32:07.15	1 Jul 1998 00:46:31.56	864.408
Mean Duration				774.349
Total Duration				3871.743

APPENDIX E
STK SIMULATIONS (CONT.)

WEST-700 KM

GROUND STATION: CAL POLY-LOCO-1 AT 86 DEGREES INCLINATION- ALTITUDE: 700 KM

15 Apr 1999 06:44:18
Satellite-Satellite4-Transmitter-Transmitter4-To-Facility-Facility1-Receiver-Receiver1: Access Summary Report

LOCO1-To-Cal Poly

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 00:32:41.47	1 Jul 1998 00:46:28.65	827.184
	2	1 Jul 1998 02:12:19.45	1 Jul 1998 02:24:36.38	736.926
	3	1 Jul 1998 11:39:23.23	1 Jul 1998 11:46:07.26	404.027
	4	1 Jul 1998 13:13:50.53	1 Jul 1998 13:27:58.61	848.082
	5	1 Jul 1998 14:55:05.17	1 Jul 1998 15:05:58.76	653.583
	6	1 Jul 1998 23:38:10.21	1 Jul 1998 23:49:07.83	657.621

Global Statistics

Min Duration	3	1 Jul 1998 11:39:23.23	1 Jul 1998 11:46:07.26	404.027
Max Duration	4	1 Jul 1998 13:13:50.53	1 Jul 1998 13:27:58.61	848.082
Mean Duration				687.904
Total Duration				4127.423

15 Apr 1999 06:44:43

GROUND STATION: CAL POLY-LOCO-1 AT 96 DEGREES INCLINATION- ALTITUDE: 700 KM

15 Apr 1999 06:46:10
Satellite-Satellite5-Transmitter-Transmitter5-To-Facility-Facility1-Receiver-Receiver1: Access Summary Report

LOCO1-To-Cal Poly

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 00:33:52.49	1 Jul 1998 00:46:16.05	743.563
	2	1 Jul 1998 02:11:24.78	1 Jul 1998 02:24:55.44	810.652
	3	1 Jul 1998 03:53:42.76	1 Jul 1998 03:55:58.30	135.539
	4	1 Jul 1998 11:38:06.32	1 Jul 1998 11:48:44.72	638.399
	5	1 Jul 1998 13:13:48.14	1 Jul 1998 13:27:50.23	842.088
	6	1 Jul 1998 14:55:58.94	1 Jul 1998 15:03:07.16	428.223
	7	1 Jul 1998 23:41:47.73	1 Jul 1998 23:47:12.69	324.957

Global Statistics

Min Duration	3	1 Jul 1998 03:53:42.76	1 Jul 1998 03:55:58.30	135.539
Max Duration	5	1 Jul 1998 13:13:48.14	1 Jul 1998 13:27:50.23	842.088
Mean Duration				560.489
Total Duration				3923.421

GROUND STATION: CAL POLY-LOCO-1 AT 104 DEGREES INCLINATION- ALTITUDE: 700 KM

15 Apr 1999 06:47:38
Satellite-Satellite6-Transmitter-Transmitter6-To-Facility-Facility1-Receiver-Receiver1: Access Summary Report

Transmitter6-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 00:35:18.90	1 Jul 1998 00:45:55.87	636.974
	2	1 Jul 1998 02:11:07.99	1 Jul 1998 02:25:00.56	832.567
	3	1 Jul 1998 03:49:44.39	1 Jul 1998 03:58:50.14	545.745
	4	1 Jul 1998 11:37:28.81	1 Jul 1998 11:49:47.62	738.807
	5	1 Jul 1998 13:13:39.45	1 Jul 1998 13:27:13.78	814.332

Global Statistics

Min Duration	3	1 Jul 1998 03:49:44.39	1 Jul 1998 03:58:50.14	545.745
Max Duration	2	1 Jul 1998 02:11:07.99	1 Jul 1998 02:25:00.56	832.567
Mean Duration				713.685
Total Duration				3568.425

APPENDIX E
STK SIMULATIONS (CONT.)

EAST-400 KM

GROUND STATION: KENNEDY SPACE CENTER-LOCO-1 AT 30 DEGREES INCLINATION-ALTITUDE: 400 KM

15 Apr 1999 05:25:13
Satellite-Satellite1-Transmitter-Transmitter1-To-Facility-East-Receiver-Receiver1: Access Summary Report

Transmitter1-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 12:23:28.07	1 Jul 1998 12:32:32.05	543.976
	2	1 Jul 1998 14:00:09.08	1 Jul 1998 14:10:52.00	642.927
	3	1 Jul 1998 15:37:57.65	1 Jul 1998 15:48:44.14	646.494
	4	1 Jul 1998 17:15:56.84	1 Jul 1998 17:26:43.72	646.880
	5	1 Jul 1998 18:53:49.06	1 Jul 1998 19:04:30.06	641.005
	6	1 Jul 1998 20:32:13.51	1 Jul 1998 20:41:04.20	530.687
Global Statistics				
Min Duration	6	1 Jul 1998 20:32:13.51	1 Jul 1998 20:41:04.20	530.687
Max Duration	4	1 Jul 1998 17:15:56.84	1 Jul 1998 17:26:43.72	646.880
Mean Duration				608.662
Total Duration				3651.969

GROUND STATION: KENNEDY SPACE CENTER-LOCO-1 AT 40 DEGREES INCLINATION-ALTITUDE: 400 KM

15 Apr 1999 05:27:50
Satellite-Satellite2-Transmitter-Transmitter2-To-Facility-East-Receiver-Receiver1: Access Summary Report

Transmitter2-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 10:50:17.40	1 Jul 1998 10:54:10.64	233.235
	2	1 Jul 1998 12:23:42.65	1 Jul 1998 12:34:07.25	624.600
	3	1 Jul 1998 14:00:59.33	1 Jul 1998 14:11:19.12	619.780
	4	1 Jul 1998 15:39:40.77	1 Jul 1998 15:48:28.75	527.972
	5	1 Jul 1998 17:17:54.53	1 Jul 1998 17:26:45.05	530.519
	6	1 Jul 1998 18:55:02.80	1 Jul 1998 19:05:25.29	622.496
	7	1 Jul 1998 20:32:16.44	1 Jul 1998 20:42:37.32	620.873
	8	1 Jul 1998 22:12:35.79	1 Jul 1998 22:15:38.25	182.461
Global Statistics				
Min Duration	8	1 Jul 1998 22:12:35.79	1 Jul 1998 22:15:38.25	182.461
Max Duration	2	1 Jul 1998 12:23:42.65	1 Jul 1998 12:34:07.25	624.600
Mean Duration				495.242
Total Duration				3961.935

GROUND STATION: KENNEDY SPACE CENTER-LOCO-1 AT 50 DEGREES INCLINATION-ALTITUDE: 400 KM

15 Apr 1999 05:21:11
Satellite-Satellite2-Transmitter-Transmitter2-To-Facility-East-Receiver-Receiver1: Access Summary Report

Transmitter2-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 10:49:59.98	1 Jul 1998 10:57:07.70	427.724
	2	1 Jul 1998 12:24:17.49	1 Jul 1998 12:34:56.45	638.954
	3	1 Jul 1998 14:02:43.63	1 Jul 1998 14:10:26.36	462.735
	4	1 Jul 1998 18:57:44.95	1 Jul 1998 19:05:27.39	462.440
	5	1 Jul 1998 20:33:14.79	1 Jul 1998 20:43:53.73	638.935
	6	1 Jul 1998 22:11:03.30	1 Jul 1998 22:18:11.56	428.265
Global Statistics				
Min Duration	1	1 Jul 1998 10:49:59.98	1 Jul 1998 10:57:07.70	427.724
Max Duration	2	1 Jul 1998 12:24:17.49	1 Jul 1998 12:34:56.45	638.954
Mean Duration				509.842
Total Duration				3059.052

APPENDIX E

STK SIMULATIONS (CONT.)

EAST-400 KM

GROUND STATION: KENNEDY SPACE CENTER-LOCO-1 AT 57 DEGREES INCLINATION-ALTITUDE: 400 KM

15 Apr 1999 05:32:21
Satellite-Satellite4-Transmitter-Transmitter4-To-Facility-East-Receiver-Receiver1: Access Summary Report

Transmitter4-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 10:50:07.94	1 Jul 1998 10:58:33.90	505.966
	2	1 Jul 1998 12:24:48.57	1 Jul 1998 12:35:04.09	615.513
	3	1 Jul 1998 14:05:46.34	1 Jul 1998 14:07:35.50	109.161
	4	1 Jul 1998 20:34:22.61	1 Jul 1998 20:44:33.60	610.991
	5	1 Jul 1998 22:10:45.38	1 Jul 1998 22:19:25.19	519.815

Global Statistics

Min Duration	3	1 Jul 1998 14:05:46.34	1 Jul 1998 14:07:35.50	109.161
Max Duration	2	1 Jul 1998 12:24:48.57	1 Jul 1998 12:35:04.09	615.513
Mean Duration				472.289
Total Duration				2361.445

APPENDIX E

STK SIMULATIONS (CONT.)

EAST-700 KM

GROUND STATION: KENNEDY SPACE CENTER-LOCO-1 AT 30 DEGREES INCLINATION-ALTITUDE: 700 KM

15 Apr 1999 06:03:21
Satellite-Satellite1-Transmitter-Transmitter1-To-Facility-East-Receiver-Receiver1: Access Summary Report

Transmitter1-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 11:31:11.66	1 Jul 1998 11:42:46.57	694.911
	2	1 Jul 1998 13:13:59.37	1 Jul 1998 13:28:33.70	874.335
	3	1 Jul 1998 14:58:26.52	1 Jul 1998 15:13:27.54	901.026
	4	1 Jul 1998 16:43:26.77	1 Jul 1998 16:58:25.43	898.667
	5	1 Jul 1998 18:28:20.87	1 Jul 1998 18:43:20.16	899.285
	6	1 Jul 1998 20:13:23.23	1 Jul 1998 20:27:20.52	837.293
	7	1 Jul 1998 22:00:09.62	1 Jul 1998 22:08:59.68	530.061
Global Statistics				
Min Duration	7	1 Jul 1998 22:00:09.62	1 Jul 1998 22:08:59.68	530.061
Max Duration	3	1 Jul 1998 14:58:26.52	1 Jul 1998 15:13:27.54	901.026
Mean Duration				805.083

GROUND STATION: KENNEDY SPACE CENTER-LOCO-1 AT 40 DEGREES INCLINATION-ALTITUDE: 700 KM

15 Apr 1999 06:10:39
Satellite-Satellite2-Transmitter-Transmitter2-To-Facility-East-Receiver-Receiver1: Access Summary Report

Transmitter2-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 11:31:34.12	1 Jul 1998 11:44:47.87	793.746
	2	1 Jul 1998 13:14:29.79	1 Jul 1998 13:29:24.15	894.363
	3	1 Jul 1998 14:59:34.77	1 Jul 1998 15:13:29.78	835.010
	4	1 Jul 1998 16:45:07.14	1 Jul 1998 16:58:26.15	799.007
	5	1 Jul 1998 18:29:40.94	1 Jul 1998 18:44:02.52	861.580
	6	1 Jul 1998 20:13:45.17	1 Jul 1998 20:28:33.25	888.079
	7	1 Jul 1998 21:59:06.16	1 Jul 1998 22:10:19.07	672.915
Global Statistics				
Min Duration	7	1 Jul 1998 21:59:06.16	1 Jul 1998 22:10:19.07	672.915
Max Duration	2	1 Jul 1998 13:14:29.79	1 Jul 1998 13:29:24.15	894.363
Mean Duration				820.671
Total Duration				5744.700

GROUND STATION: KENNEDY SPACE CENTER-LOCO-1 AT 50 DEGREES INCLINATION-ALTITUDE: 700 KM

15 Apr 1999 06:06:16
Satellite-Satellite3-Transmitter-Transmitter3-To-Facility-East-Receiver-Receiver1: Access Summary Report

Transmitter3-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 09:55:41.00	1 Jul 1998 09:57:52.04	131.044
	2	1 Jul 1998 11:32:03.93	1 Jul 1998 11:46:15.00	851.072
	3	1 Jul 1998 13:15:19.81	1 Jul 1998 13:29:25.61	845.798
	4	1 Jul 1998 15:01:51.53	1 Jul 1998 15:12:01.85	610.323
	5	1 Jul 1998 16:48:50.96	1 Jul 1998 16:56:31.65	460.683
	6	1 Jul 1998 18:32:21.04	1 Jul 1998 18:43:59.05	698.016
	7	1 Jul 1998 20:14:56.98	1 Jul 1998 20:29:36.30	879.327
	8	1 Jul 1998 21:58:43.51	1 Jul 1998 22:11:37.43	773.918
Global Statistics				
Min Duration	1	1 Jul 1998 09:55:41.00	1 Jul 1998 09:57:52.04	131.044
Max Duration	7	1 Jul 1998 20:14:56.98	1 Jul 1998 20:29:36.30	879.327
Mean Duration				656.273
Total Duration				5250.181

APPENDIX E

STK SIMULATIONS (CONT.)

EAST-700 KM

GROUND STATION: KENNEDY SPACE CENTER-LOCO-1 AT 57 DEGREES INCLINATION-ALTITUDE: 700 KM

15 Apr 1999 06:12:03

Satellite-Satellite4-Transmitter-Transmitter4-To-Facility-East-Receiver-Receiver1: Access Summary Report

Transmitter4-To-Receiver1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	1 Jul 1998 09:54:51.00	1 Jul 1998 10:01:04.71	373.710
	2	1 Jul 1998 11:32:25.53	1 Jul 1998 11:46:55.83	870.294
	3	1 Jul 1998 13:16:05.12	1 Jul 1998 13:28:52.63	767.509
	4	1 Jul 1998 15:06:27.10	1 Jul 1998 15:07:36.70	69.602
	5	1 Jul 1998 18:35:38.73	1 Jul 1998 18:42:46.07	427.342
	6	1 Jul 1998 20:16:13.37	1 Jul 1998 20:30:09.64	836.268
	7	1 Jul 1998 21:58:46.11	1 Jul 1998 22:12:28.22	822.114
Global Statistics				
Min Duration	4	1 Jul 1998 15:06:27.10	1 Jul 1998 15:07:36.70	69.602
Max Duration	2	1 Jul 1998 11:32:25.53	1 Jul 1998 11:46:55.83	870.294
Mean Duration				595.263
Total Duration				4166.838

APPENDIX E
STK SIMULATIONS
ONMI ANTENNA DOWNLOAD PASSES

WEST-400 KM
CAL POLY-56 DEGREES INCLINATION-400 KM ALTIDUDE

18 May 1999 20:15:24
Satellite-Satellite2-Sensor-onmi1-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 00:26:48.60	21 May 1999 00:34:58.08	489.483
2	21 May 1999 02:02:04.38	21 May 1999 02:12:42.63	638.251
3	21 May 1999 03:39:21.06	21 May 1999 03:46:19.80	418.739
4	21 May 1999 17:02:46.73	21 May 1999 17:12:58.74	612.005
5	21 May 1999 18:39:03.16	21 May 1999 18:48:46.40	583.240
6	21 May 1999 20:18:48.49	21 May 1999 20:23:25.91	277.415
7	21 May 1999 23:34:01.73	21 May 1999 23:40:25.06	383.330
8	22 May 1999 01:08:57.68	22 May 1999 01:19:16.80	619.119
9	22 May 1999 02:45:14.02	22 May 1999 02:54:32.36	558.342
10	22 May 1999 16:10:23.01	22 May 1999 16:19:19.20	536.193
11	22 May 1999 17:45:19.74	22 May 1999 17:55:45.73	625.989
12	22 May 1999 19:23:56.76	22 May 1999 19:30:45.91	409.159
13	22 May 1999 22:41:24.71	22 May 1999 22:45:27.22	242.506
14	23 May 1999 00:15:58.48	23 May 1999 00:25:28.54	570.058
15	23 May 1999 01:51:39.63	23 May 1999 02:02:00.92	621.291
16	23 May 1999 15:18:50.61	23 May 1999 15:24:59.48	368.862
17	23 May 1999 16:51:59.00	23 May 1999 17:02:37.06	638.055
18	23 May 1999 18:29:28.64	23 May 1999 18:37:57.62	508.977
19	23 May 1999 20:11:07.87	23 May 1999 20:11:30.01	22.137
20	23 May 1999 23:23:05.33	23 May 1999 23:31:18.42	493.089
21	24 May 1999 00:58:21.96	24 May 1999 01:09:00.30	638.340
22	24 May 1999 02:35:42.07	24 May 1999 02:42:32.63	410.562
23	24 May 1999 15:59:03.14	24 May 1999 16:09:16.96	613.827
24	24 May 1999 17:35:22.44	24 May 1999 17:45:03.44	580.991
25	24 May 1999 19:15:10.56	24 May 1999 19:19:42.00	271.447
26	24 May 1999 22:30:18.19	24 May 1999 22:36:46.29	388.102
27	25 May 1999 00:05:14.91	25 May 1999 00:15:35.37	620.461
28	25 May 1999 01:41:33.00	25 May 1999 01:50:47.64	554.634
29	25 May 1999 15:06:37.98	25 May 1999 15:15:38.40	540.420
30	25 May 1999 16:41:38.16	25 May 1999 16:52:03.03	624.872
31	25 May 1999 18:20:17.79	25 May 1999 18:27:02.47	404.686
32	25 May 1999 21:37:40.56	25 May 1999 21:41:49.62	249.062
33	25 May 1999 23:12:15.45	25 May 1999 23:21:47.98	572.529
34	26 May 1999 00:47:57.74	26 May 1999 00:58:17.53	619.787
35	26 May 1999 14:15:02.62	26 May 1999 14:21:21.22	378.597
36	26 May 1999 15:48:16.50	26 May 1999 15:58:54.71	638.214
37	26 May 1999 17:25:48.81	26 May 1999 17:34:14.42	505.609
38	26 May 1999 22:19:22.07	26 May 1999 22:27:38.72	496.651
39	26 May 1999 23:54:39.56	27 May 1999 00:05:17.93	638.375
40	27 May 1999 01:32:03.22	27 May 1999 01:38:45.29	402.076
41	27 May 1999 14:55:19.58	27 May 1999 15:05:35.16	615.578
42	27 May 1999 16:31:41.76	27 May 1999 16:41:20.45	578.697
43	27 May 1999 18:11:32.67	27 May 1999 18:15:58.06	265.383
44	27 May 1999 21:26:34.66	27 May 1999 21:33:07.48	392.820
45	27 May 1999 23:01:32.16	27 May 1999 23:11:53.91	621.755

Global Statistics

Min Duration	19	23 May 1999 20:11:07.87	23 May 1999 20:11:30.01	22.137
Max Duration	39	26 May 1999 23:54:39.56	27 May 1999 00:05:17.93	638.375
Mean Duration			494.216	
Total Duration			22239.717	

APPENDIX E
STK SIMULATIONS (CONT.)

CAL POLY-66 DEGREES INCLINATION-400 KM ALTIDUDE

18 May 1999 20:21:47
Satellite-Satellite2-Sensor-onmi1-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 02:02:48.34	21 May 1999 02:12:36.80	588.459
2	21 May 1999 03:37:57.79	21 May 1999 03:47:14.28	556.486
3	21 May 1999 15:31:48.59	21 May 1999 15:36:47.48	298.893
4	21 May 1999 17:03:25.82	21 May 1999 17:13:57.98	632.153
5	21 May 1999 18:41:03.15	21 May 1999 18:47:31.77	388.620
6	22 May 1999 01:11:37.94	22 May 1999 01:19:38.03	480.085
7	22 May 1999 02:45:46.32	22 May 1999 02:56:10.81	624.489
8	22 May 1999 16:11:59.61	22 May 1999 16:22:00.32	600.706
9	22 May 1999 17:47:35.12	22 May 1999 17:56:38.39	543.269
10	23 May 1999 00:21:16.26	23 May 1999 00:25:25.55	249.286
11	23 May 1999 01:53:59.64	23 May 1999 02:04:24.48	624.840
12	23 May 1999 03:30:05.96	23 May 1999 03:37:29.36	443.398
13	23 May 1999 15:21:23.12	23 May 1999 15:29:30.29	487.166
14	23 May 1999 16:54:59.39	23 May 1999 17:05:16.16	616.770
15	23 May 1999 18:34:58.28	23 May 1999 18:37:14.47	136.190
16	24 May 1999 01:02:32.72	24 May 1999 01:11:58.34	565.616
17	24 May 1999 02:37:24.41	24 May 1999 02:47:08.16	583.746
18	24 May 1999 14:32:56.45	24 May 1999 14:35:07.07	130.621
19	24 May 1999 16:03:04.18	24 May 1999 16:13:34.05	629.866
20	24 May 1999 17:40:04.57	24 May 1999 17:47:27.97	443.400
21	25 May 1999 00:11:30.78	25 May 1999 00:18:44.43	433.651
22	25 May 1999 01:45:21.02	25 May 1999 01:55:51.70	630.684
23	25 May 1999 03:23:25.86	25 May 1999 03:26:24.10	178.236
24	25 May 1999 15:11:50.16	25 May 1999 15:21:29.18	579.016
25	25 May 1999 16:46:53.77	25 May 1999 16:56:24.23	570.466
26	25 May 1999 23:22:02.55	25 May 1999 23:23:30.96	88.414
27	26 May 1999 00:53:39.98	26 May 1999 01:03:54.36	614.382
28	26 May 1999 02:29:21.39	26 May 1999 02:37:37.78	496.385
29	26 May 1999 14:21:33.10	26 May 1999 14:28:44.40	431.304
30	26 May 1999 15:54:29.87	26 May 1999 16:04:56.21	626.345
31	26 May 1999 17:33:17.89	26 May 1999 17:37:46.99	269.100
32	27 May 1999 00:02:18.92	27 May 1999 00:11:16.26	537.340
33	27 May 1999 01:36:53.77	27 May 1999 01:46:57.95	604.180
34	27 May 1999 15:02:45.70	27 May 1999 15:13:08.40	622.699
35	27 May 1999 16:39:12.01	27 May 1999 16:47:20.15	488.138
36	27 May 1999 23:11:27.92	27 May 1999 23:17:44.49	376.561

Global Statistics

Min Duration	26	25 May 1999 23:22:02.55	25 May 1999 23:23:30.96	88.414
Max Duration	4	21 May 1999 17:03:25.82	21 May 1999 17:13:57.98	632.153
Mean Duration			476.971	
Total Duration			17170.961	

APPENDIX E
STK SIMULATIONS (CONT.)

CAL POLY-76 DEGREES INCLINATION-400 KM ALTIDUDE

18 May 1999 20:26:56
Satellite-Satellite2-Sensor-onmi1-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 02:04:36.18	21 May 1999 02:11:54.31	438.130
2	21 May 1999 03:37:26.03	21 May 1999 03:47:40.49	614.459
3	21 May 1999 15:31:21.44	21 May 1999 15:39:22.97	481.525
4	21 May 1999 17:04:03.91	21 May 1999 17:14:04.51	600.606
5	22 May 1999 02:46:51.00	22 May 1999 02:57:05.16	614.163
6	22 May 1999 04:22:00.21	22 May 1999 04:29:05.92	425.708
7	22 May 1999 16:13:14.73	22 May 1999 16:23:37.34	622.612
8	22 May 1999 17:49:45.29	22 May 1999 17:55:58.77	373.475
9	23 May 1999 01:56:51.05	23 May 1999 02:05:28.73	517.683
10	23 May 1999 03:30:21.15	23 May 1999 03:40:08.36	587.211
11	23 May 1999 15:23:25.26	23 May 1999 15:32:34.74	549.480
12	23 May 1999 16:57:12.34	23 May 1999 17:06:36.21	563.863
13	24 May 1999 01:08:10.98	24 May 1999 01:12:02.20	231.224
14	24 May 1999 02:39:32.04	24 May 1999 02:49:57.16	625.118
15	24 May 1999 04:15:49.05	24 May 1999 04:20:26.14	277.092
16	24 May 1999 14:35:11.67	24 May 1999 14:40:20.03	308.365
17	24 May 1999 16:05:58.58	24 May 1999 16:16:22.96	624.382
18	24 May 1999 17:44:16.80	24 May 1999 17:47:25.52	188.722
19	25 May 1999 01:49:16.47	25 May 1999 01:58:47.48	571.008
20	25 May 1999 03:23:24.03	25 May 1999 03:32:24.47	540.441
21	25 May 1999 15:15:43.42	25 May 1999 15:25:36.21	592.796
22	25 May 1999 16:50:33.29	25 May 1999 16:59:00.23	506.943
23	26 May 1999 00:59:48.66	26 May 1999 01:06:21.81	393.149
24	26 May 1999 02:32:18.55	26 May 1999 02:42:39.45	620.899
25	26 May 1999 14:26:39.41	26 May 1999 14:34:02.04	442.629
26	26 May 1999 15:58:52.15	26 May 1999 16:09:03.12	610.970
27	27 May 1999 01:41:49.21	27 May 1999 01:51:54.11	604.897
28	27 May 1999 03:16:37.94	27 May 1999 03:24:25.51	467.570
29	27 May 1999 15:08:13.02	27 May 1999 15:18:30.26	617.246
30	27 May 1999 16:44:10.82	27 May 1999 16:51:12.84	422.021

Global Statistics

Min Duration	18	24 May 1999 17:44:16.80	24 May 1999 17:47:25.52	188.722
Max Duration	14	24 May 1999 02:39:32.04	24 May 1999 02:49:57.16	625.118
Mean Duration			501.146	
Total Duration			15034.388	

CAL POLY-86 DEGREES INCLINATION-400 KM ALTIDUDE
18 May 1999 20:28:05
Satellite-Satellite2-Sensor-onmi1-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 03:37:32.66	21 May 1999 03:47:47.90	615.238
2	21 May 1999 05:12:13.62	21 May 1999 05:17:57.56	343.937
3	21 May 1999 15:31:13.34	21 May 1999 15:40:42.31	568.968
4	21 May 1999 17:04:37.22	21 May 1999 17:13:15.28	518.057
5	22 May 1999 02:48:26.49	22 May 1999 02:57:04.30	517.811
6	22 May 1999 04:20:59.31	22 May 1999 04:30:28.43	569.125
7	22 May 1999 14:43:44.43	22 May 1999 14:49:27.81	343.378
8	22 May 1999 16:13:53.75	22 May 1999 16:24:09.03	615.277
9	23 May 1999 03:30:57.52	23 May 1999 03:41:15.47	617.953
10	23 May 1999 05:06:06.13	23 May 1999 05:10:48.54	282.401
11	23 May 1999 15:24:30.53	23 May 1999 15:34:13.50	582.975
12	23 May 1999 16:58:18.80	23 May 1999 17:06:30.95	492.160
13	24 May 1999 02:41:43.74	24 May 1999 02:50:43.70	539.961
14	24 May 1999 04:14:31.56	24 May 1999 04:23:44.02	552.459
15	24 May 1999 14:36:42.08	24 May 1999 14:43:14.25	392.173
16	24 May 1999 16:07:22.65	24 May 1999 16:17:33.09	610.444
17	25 May 1999 01:54:18.10	25 May 1999 01:57:49.95	211.854
18	25 May 1999 03:24:23.16	25 May 1999 03:34:41.72	618.565
19	25 May 1999 05:00:10.64	25 May 1999 05:03:26.89	196.254
20	25 May 1999 15:17:49.24	25 May 1999 15:27:43.72	594.489
21	25 May 1999 16:52:02.90	25 May 1999 16:59:44.71	461.807
22	26 May 1999 02:35:02.36	26 May 1999 02:44:21.15	558.797
23	26 May 1999 04:08:05.02	26 May 1999 04:16:57.86	532.848
24	26 May 1999 14:29:44.28	26 May 1999 14:36:56.75	432.470
25	26 May 1999 16:00:52.92	26 May 1999 16:10:56.33	603.410
26	27 May 1999 01:47:06.95	27 May 1999 01:52:01.97	295.026
27	27 May 1999 03:17:49.58	27 May 1999 03:28:06.66	617.086
28	27 May 1999 15:11:09.38	27 May 1999 15:21:13.04	603.656

APPENDIX E
STK SIMULATIONS (CONT.)

29	27 May 1999 16:45:50.05	27 May 1999 16:52:56.05	426.007

Global Statistics

Min Duration	19	25 May 1999 05:00:10.64	25 May 1999 05:03:26.89	196.254
Max Duration	18	25 May 1999 03:24:23.16	25 May 1999 03:34:41.72	618.565
Mean Duration			493.606	
Total Duration			14314.584	

CAL POLY-96 DEGREES INCLINATION-400 KM ALTIDUDE

18 May 1999 20:29:17
Satellite-Satellite2-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 03:38:15.38	21 May 1999 03:47:37.07	561.688
2	21 May 1999 05:10:20.57	21 May 1999 05:19:02.90	522.325
3	21 May 1999 15:31:01.69	21 May 1999 15:41:06.93	605.244
4	21 May 1999 17:05:13.22	21 May 1999 17:11:20.28	367.062
5	22 May 1999 02:51:13.53	22 May 1999 02:55:23.56	250.028
6	22 May 1999 04:20:32.72	22 May 1999 04:30:43.90	611.174
7	22 May 1999 05:55:03.12	22 May 1999 05:58:40.41	217.287
8	22 May 1999 14:43:05.20	22 May 1999 14:51:04.72	479.520
9	22 May 1999 16:13:50.26	22 May 1999 16:23:35.80	585.535
10	23 May 1999 03:31:44.04	23 May 1999 03:40:49.38	545.336
11	23 May 1999 05:03:35.68	23 May 1999 05:12:36.58	540.900
12	23 May 1999 15:24:27.54	23 May 1999 15:34:26.96	599.422
13	23 May 1999 16:58:10.39	23 May 1999 17:05:01.26	410.870
14	24 May 1999 02:45:26.36	24 May 1999 02:47:47.41	141.046
15	24 May 1999 04:13:54.75	24 May 1999 04:24:06.41	611.656
16	24 May 1999 05:47:54.48	24 May 1999 05:52:42.13	287.647
17	24 May 1999 14:36:44.13	24 May 1999 14:44:16.08	451.949
18	24 May 1999 16:07:06.75	24 May 1999 16:17:01.59	594.836
19	25 May 1999 03:25:13.93	25 May 1999 03:33:59.99	526.060
20	25 May 1999 04:56:51.77	25 May 1999 05:06:08.75	556.983
21	25 May 1999 15:17:54.57	25 May 1999 15:27:46.27	591.703
22	25 May 1999 16:51:11.50	25 May 1999 16:58:38.80	447.295
23	26 May 1999 04:07:17.48	26 May 1999 04:17:27.76	610.279
24	26 May 1999 05:40:54.01	26 May 1999 05:46:35.08	341.069
25	26 May 1999 14:30:25.43	26 May 1999 14:37:25.61	420.183
26	26 May 1999 16:00:24.53	26 May 1999 16:10:26.54	602.010
27	27 May 1999 03:18:45.25	27 May 1999 03:27:08.73	503.478
28	27 May 1999 04:50:08.77	27 May 1999 04:59:39.54	570.771
29	27 May 1999 15:11:22.82	27 May 1999 15:21:04.85	582.028
30	27 May 1999 16:44:15.66	27 May 1999 16:52:13.76	478.092

Global Statistics

Min Duration	14	24 May 1999 02:45:26.36	24 May 1999 02:47:47.41	141.046
Max Duration	15	24 May 1999 04:13:54.75	24 May 1999 04:24:06.41	611.656
Mean Duration			487.116	
Total Duration			14613.474	

CAL POLY-104 DEGREES INCLINATION-400 KM ALTIDUDE
18 May 1999 20:33:58
Satellite-Satellite2-Sensor-onmi1-To-Facility-CalPoly-Sensor-CPreceiver: Access Summary Report

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	21 May 1999 03:39:19.16	21 May 1999 03:47:11.58	472.421
2	21 May 1999 05:09:36.48	21 May 1999 05:19:20.50	584.021
3	21 May 1999 14:02:05.98	21 May 1999 14:08:21.97	375.992
4	21 May 1999 15:30:46.55	21 May 1999 15:40:51.97	605.428
5	21 May 1999 17:06:41.58	21 May 1999 17:08:04.75	83.168
6	22 May 1999 04:20:27.67	22 May 1999 04:30:16.18	588.505
7	22 May 1999 05:52:11.68	22 May 1999 06:00:09.16	477.485
8	22 May 1999 14:42:18.41	22 May 1999 14:51:21.53	543.119
9	22 May 1999 16:13:17.47	22 May 1999 16:22:24.56	547.096
10	23 May 1999 03:32:48.43	23 May 1999 03:39:04.37	375.934
11	23 May 1999 05:02:18.09	23 May 1999 05:12:19.43	601.335
12	23 May 1999 06:36:03.64	23 May 1999 06:39:16.60	192.957
13	23 May 1999 13:56:09.87	23 May 1999 14:00:16.34	246.470
14	23 May 1999 15:23:35.54	23 May 1999 15:33:39.43	603.881
15	23 May 1999 16:57:12.23	23 May 1999 17:02:52.39	340.160
16	24 May 1999 04:13:27.56	24 May 1999 04:22:45.82	558.255

APPENDIX E
STK SIMULATIONS (CONT.)

17	24 May 1999 05:44:39.39	24 May 1999 05:53:30.85	531.461
18	24 May 1999 14:35:37.23	24 May 1999 14:43:51.11	493.875
19	24 May 1999 16:05:44.41	24 May 1999 16:15:26.35	581.939
20	25 May 1999 03:26:57.87	25 May 1999 03:30:14.03	196.156
21	25 May 1999 04:55:04.29	25 May 1999 05:05:10.53	606.247
22	25 May 1999 06:27:46.11	25 May 1999 06:33:33.48	347.372
23	25 May 1999 15:16:32.34	25 May 1999 15:26:22.28	589.931
24	25 May 1999 16:48:57.63	25 May 1999 16:56:28.25	450.624
25	26 May 1999 04:06:35.36	26 May 1999 04:15:04.60	509.241
26	26 May 1999 05:37:13.36	26 May 1999 05:46:42.66	569.300
27	26 May 1999 14:29:07.76	26 May 1999 14:36:12.71	424.954
28	26 May 1999 15:58:20.78	26 May 1999 16:08:21.69	600.919
29	27 May 1999 04:47:55.28	27 May 1999 04:57:53.72	598.442
30	27 May 1999 06:19:57.57	27 May 1999 06:27:17.32	439.745
31	27 May 1999 15:09:37.15	27 May 1999 15:19:00.44	563.289
32	27 May 1999 16:41:04.98	27 May 1999 16:49:45.11	520.136

Global Statistics

Min Duration	5	21 May 1999 17:06:41.58	21 May 1999 17:08:04.75	83.168
Max Duration	21	25 May 1999 04:55:04.29	25 May 1999 05:05:10.53	606.247
Mean Duration			475.620	
Total Duration			15219.856	

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