THE SMARTSat REDESIGN STUDY

Submitted
to Roger Arno
NASA/Ames Research Center

to Gary Langford
SkyWatch Information Systems, Inc.

and to Professor Robert Twiggs
Department of Aeronautics and Astronautics
Stanford University

by
Richard Lu
Jeffrey Chan

In Fulfillment of the
Research Assistantship Grants for the
Fall Quarter of 1994

December 16, 1994

TABLE OF CONTENTS

(INCLUDING INDEX OF TABLES, GRAPHS)

I. INTRODUCTION

The original design of the Student Mentored Advanced Research and Technology Satellite (SMARTSat) began in January of 1994. The mission adhered to the guidelines and constraints set forth by the Student Explorer Demonstration Initiative Proposal, which was sponsored by the Universities Space Research Association.

This report represents a redesign of that original concept. The main differences include a higher orbit and a three axis stabilization system. Of course, these changes imparted significant effects upon the other subsystems. Outlined in this document are those modifications and it offers some new analyses as well.

For each subsystem there is a brief description, followed by relevant design assumptions, as summary of conclusions, and finally all pertinent supporting data and graphs. Towards the end of the report, the important issue of cost is addressed as well. Whenever possible, parametric analyses were made. This was done so that in the event the mission proceeds, certain variations in design requirements can be assessed for their overall impact upon the satellite system.

This work was performed under research assistantship grants to the authors covering the Fall quarter of 1994. Great time and effort was invested in the original proposal from January to July of 1994. As a result, the original proposal is attached at the end of this report. It serves as an addendum to answer any detailed mission or managerial questions regarding the design and implementation of SMARTSat. It also serves as a comprehensive testament to the work that has been done on the SMARTSat design over the past year.

II. SUBSYSTEMS ANALYSES

A. ORBIT

Brief Description:

The orbit of the satellite affects many of the subsystems onboard, most notably the payload, power, communication, and thermal subsystems. The environment of that orbit also affects the control subsystem, as it determines what disturbances will occur. Since the orbit influences the design of many or all of the subsystems, it should be determined first. The original design called for a 550 km circular polar orbit. That has been changed to a 750 km sun-synchronous orbit.

Assumptions:

- DASI instruments require surface daylight to obtain data, preferably in the morning.
- Pegasus launch vehicle.

Conclusions:

1) The Pegasus can inject 310 pounds into a 750 km sun-synchronous orbit. This orbit yields the following characteristics:

- 99.82 minute orbital period
- 98.39° inclination
- 35.2 minute maximum dark time
- 6.69 km/sec ground track velocity
- 0.208 km/year orbit decay rate

2) The thinner atmosphere at this altitude results in less atmospheric drag which increases the satellite's longevity.

Data and Graphs:

Period

$$\tau_p = \frac{2\pi}{\mu^{1/2}} a^{3/2}$$

$r_{earth} = 6378$ km

a (radius of orbit) = 7128 km

μ (earth's grav. constant) = 398600.5 $km^3 sec^{-2}$

τ_p=5989 sec.=99.82 min.

Inclination

$$(\Omega)_{avg} = \frac{-3\mu J_2 R^2}{2ha^3} \cos i$$

i = inclination

R = radius of the earth

h = angular momentum

$(\Omega)_{avg}$=360/365.25

i = 98.39°

Maximum Time in View

$$T = \frac{P\lambda_{max}}{180^{\circ}}$$

λ_{max} = max. earth central angle

λ_{max}(0° elevation) = 26.52°

λ_{max}(5° elevation) = 21.95°

λ_{max}(10° elevation) = 18.21°

T (0° elevation) = 14.70 min.

T (5° elevation) = 12.17 min.

T (10° elevation) = 10.10 min.

Orbit Decay

$$(km/yr) = \frac{-2\pi(C_D A/m)\rho r^2}{P}$$

$\rho = 1.24 \times 10^{-14}$

Orbit decay rate = -0.208 km/yr

Pegasus Launch Vehicle Capability For Sun-Synchronous Orbits

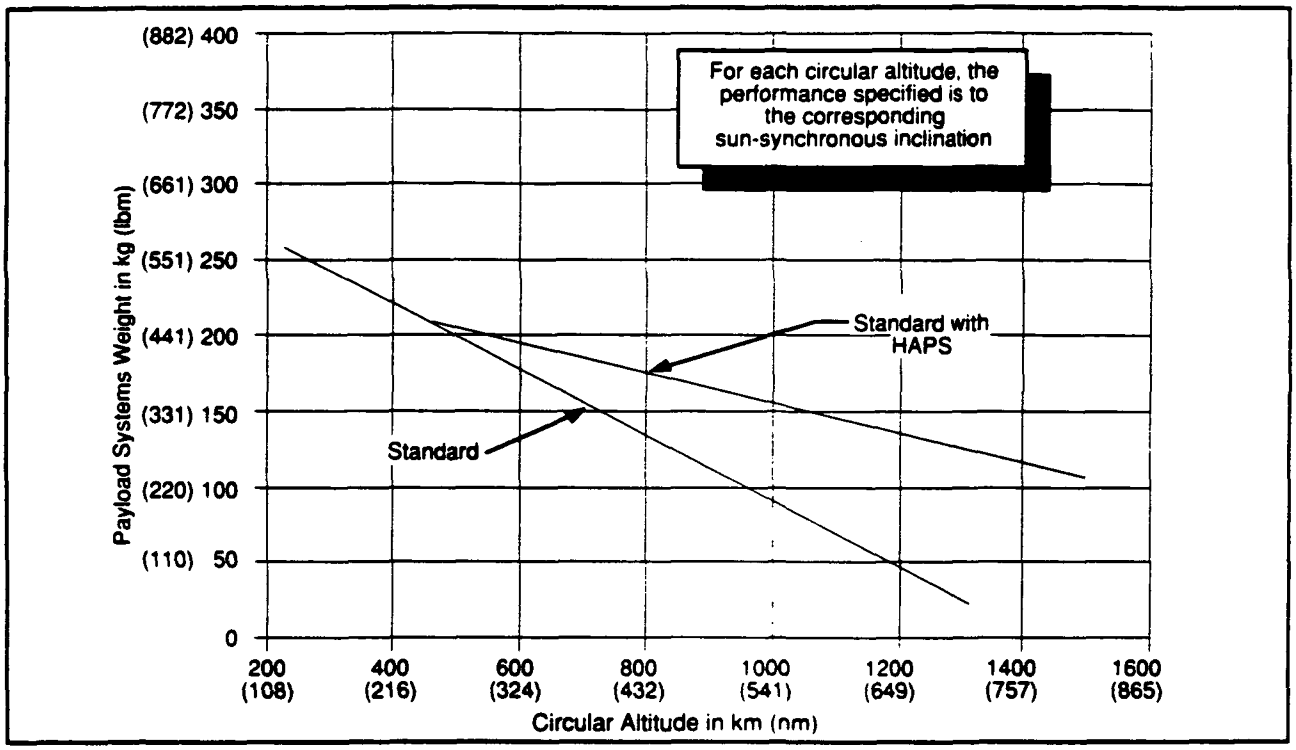


Figure 3.6. Pegasus performance capability to a circular, sun synchronous orbit from the Western Test Range.

B. POWER

Brief Description:

As satellites are remotely operated, they must be able to generate their own power in order to fulfill their mission. A common way to supply power is through the combined use of solar panels and rechargeable batteries. Operation of the DASI instrument draws the most power, and is therefore the driving requirement of the power subsystem. Since the DASI should function only in daylight, power may be drawn from the solar panels, instead of the batteries. The required number of batteries to assure a maximum allowable depth of discharge is therefore minimized.

Assumptions:

- Use of AeroAstro batteries (1.2 Amp-hr each).
- Power consumption of most components from USRA Proposal, Section I, page 10.
- Momentum wheels and torquer coils are used intermittently and therefore result in negligible load.
- 94% packing factor on solar panels.
- 14.5% solar cell efficiency BOL.
- 11.2% solar cell efficiency EOL.
- Sun-synchronous orbit ≈ 98.39° inclination orbit.
- ≈20% maximum allowable Depth of Discharge (DoD) of the batteries.
- Six solar side panels forming a regular hexagon in cross section, 36" in height, 21" in width.
- Top solar panel forming a regular hexagon, 42" point-to-point dimension.

Conclusions:

1) Three batteries are required to satisfy maximum worst-case depth of discharge. Maximum depth is 21% over a dark period of 35 minutes, with the DASI operating during that period. DoD would be worse if both DASI's operated concurrently, however this will not be planned.

2) The solar panels supply enough power to run the major components of SMARTSat at EOL, supplying 80 Watts of average power. Maximum load from SMARTSat is about 67 Watts, while a DASI is running. Extra power is available to drive the momentum wheels and the torquer coils, which were not considered in the analysis.

3) The power supplied by the solar panels increases as the sun-synchronous orbit moves from a noontime-midnight orbit to a terminator orbit.

4) From the analysis of the solar panel power output, one solar panel will not be illuminated. This fact allows us to eliminate the panel and lower the overall cost of SMARTSat.

Data and Graphs:

Available energy from 2 AeroAstro batteries

$$2.4\ \text{A-hr}\left(\frac{3600\text{s}}{\text{Hr}}\right)30\text{V} = \text{W}\cdot\text{s} = \text{J} = 259{,}200\ \text{J}$$

Available energy from 3 AeroAstro batteries

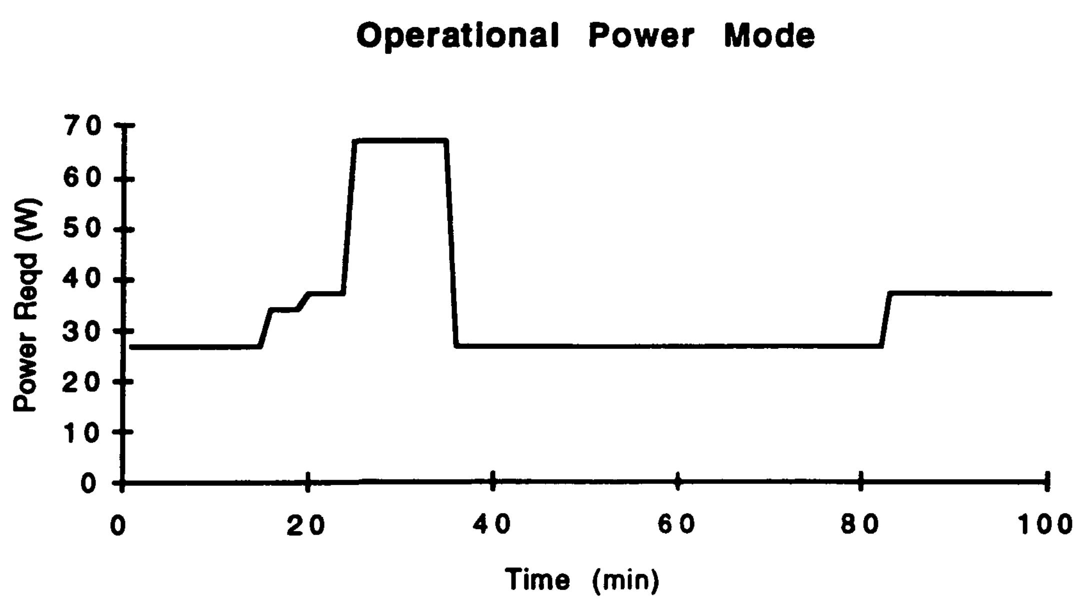
$$3.6\ \text{A-hr}\left(\frac{3600\text{s}}{\text{Hr}}\right)30\text{V} = \text{W}\cdot\text{s} = \text{J} = 388{,}800\ \text{J}$$

Operational Components During Dark Period of 35 minutes:

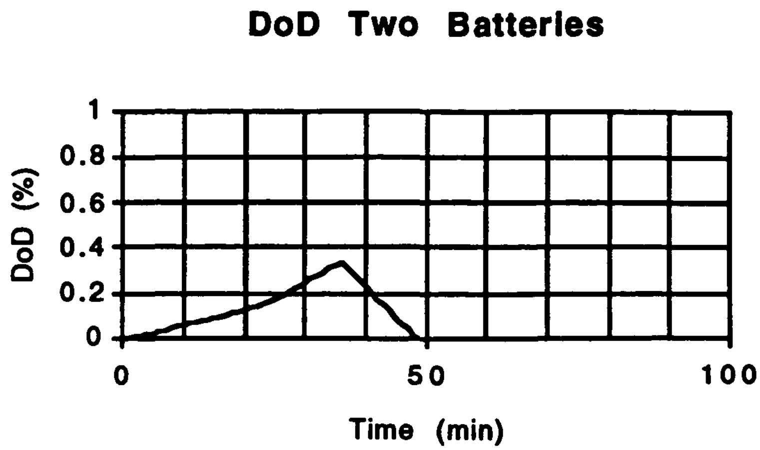
Component	Power Consumption (W)	Operational Time (min)
CPU	8.5	35
GPS	4.5	35
Sensors	6.1	35
Rx	7	35
Cooler	1	35
Tx	10	15
Tx warm-up	7	4
DASI	29.5	10
MuP23	.5	1
Total energy consumed	Σ Power * Time	75240 Joules

Power Consumption of Components:

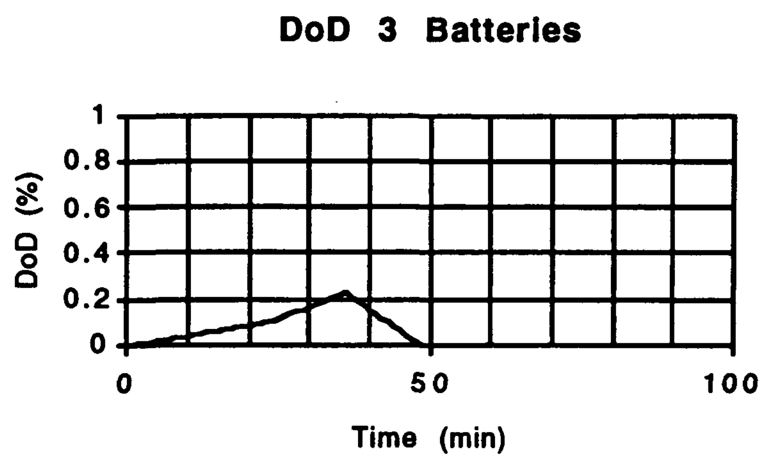
Component	Consumption (W)	
GPS	4.5	
Primary Processor	4.5	
Processor Memory	4.0	
DASI (one)	25.0	
Optics	0.5	
Research Processor	0.5	
Data Handler	5.0	
Transmitter	10.0	
Transmitter Warm-up	7.0	
Receiver	7.0	
Thermionic Cooler	1.0	
Horizon Sensor	4.0	
Sun Sensor	2.0	
Magnetometer	0.1	
Momentum Wheels	30-60	used for short times
Torquer Coils	9.0	used for short times



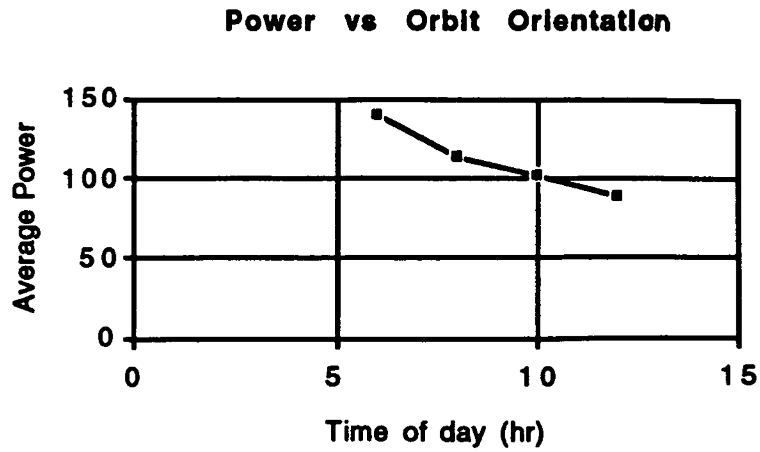
A graph of the power demands on the power subsystem for worst case DoD.



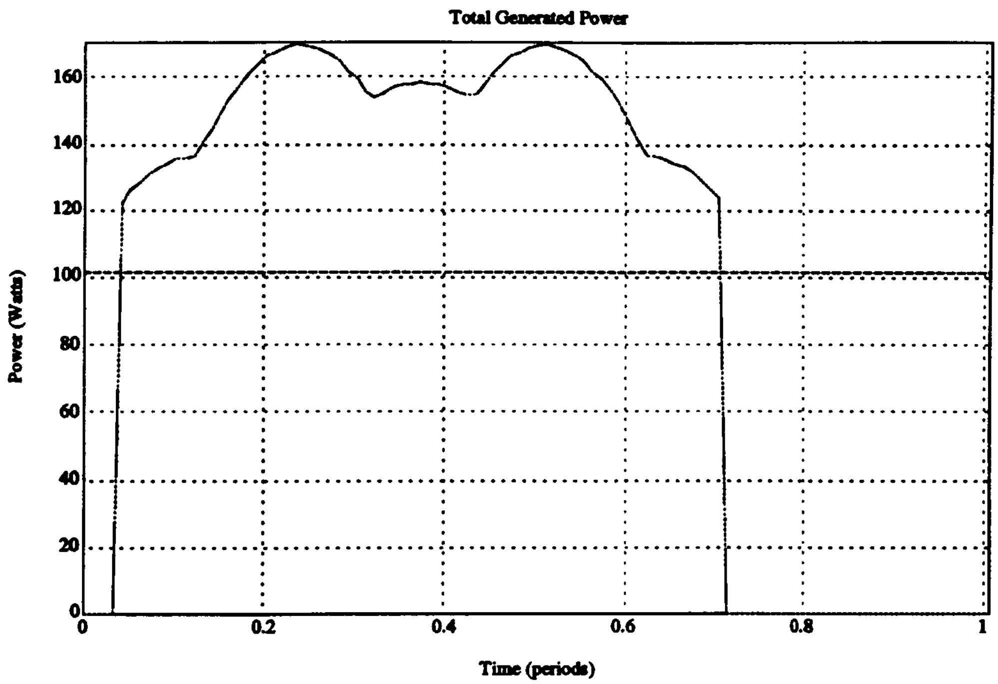
A graph of the DoD of two AeroAstro batteries based on the operational power mode above, with a 35 minute dark period starting at time zero.



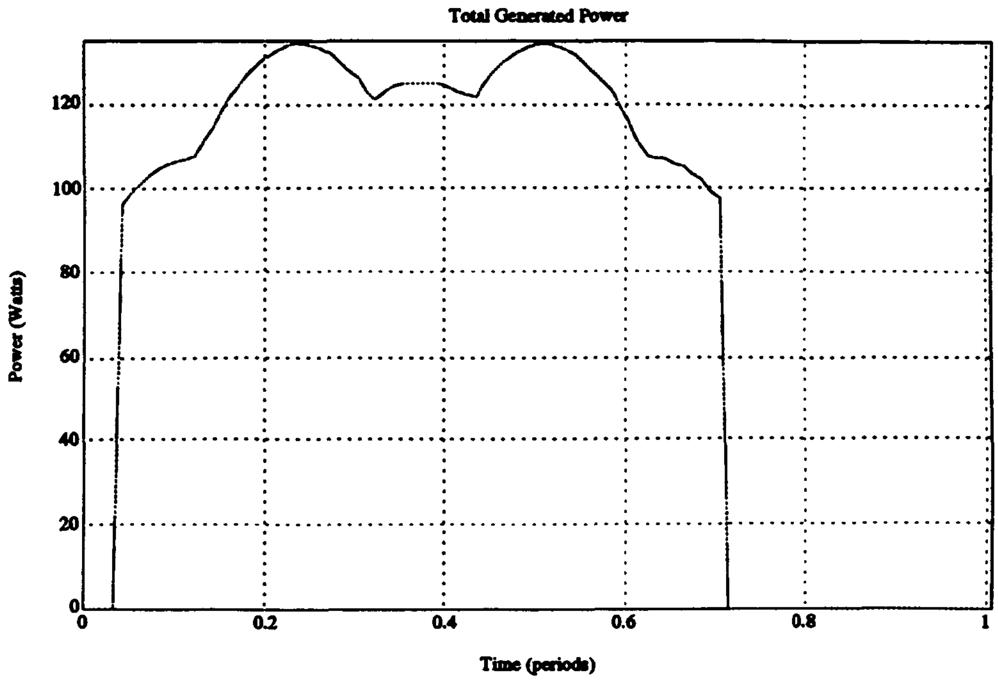
A plot fo the DoD of three AeroAstro batteries using the power operational mode above, with a 35 minute dark period starting at time zero.



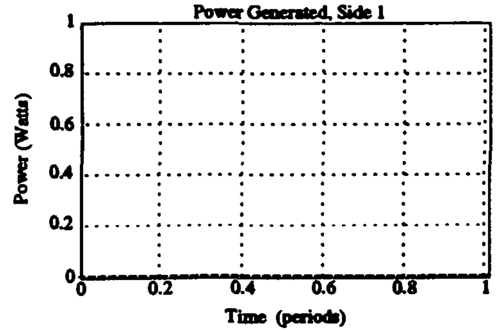
The above graph shows the behavior of the average power supplied by the solar panels as the sun-synchronous orbit changes from a noon-midnight orbit to a terminator orbit.

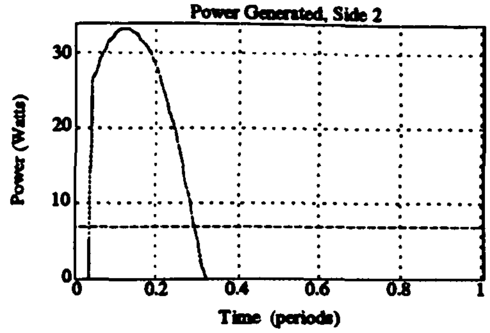


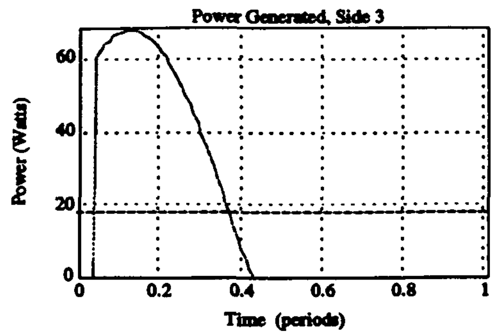
The above graph shows the power output of the solar cell panels in a 10-22 sun-synchronous orbit at BOL. The level line indicates the average power over the orbit.

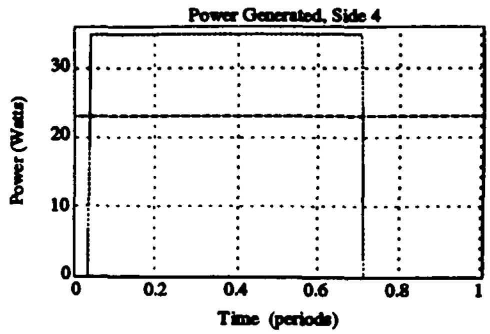


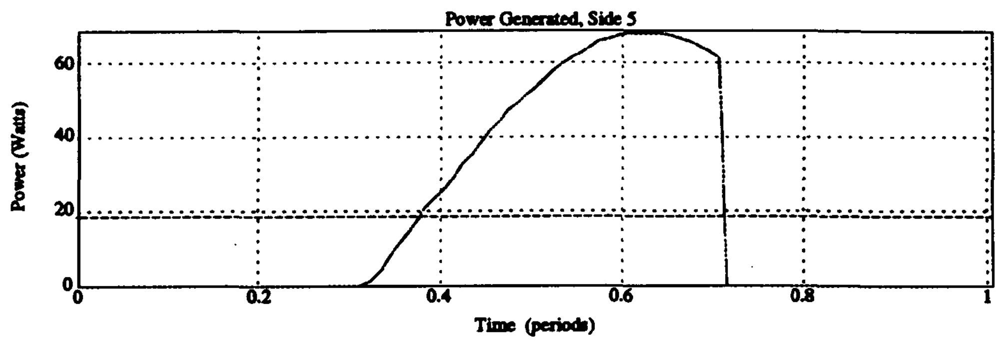
The above graph shows the same information except for the solar cell output is measured at EOL.

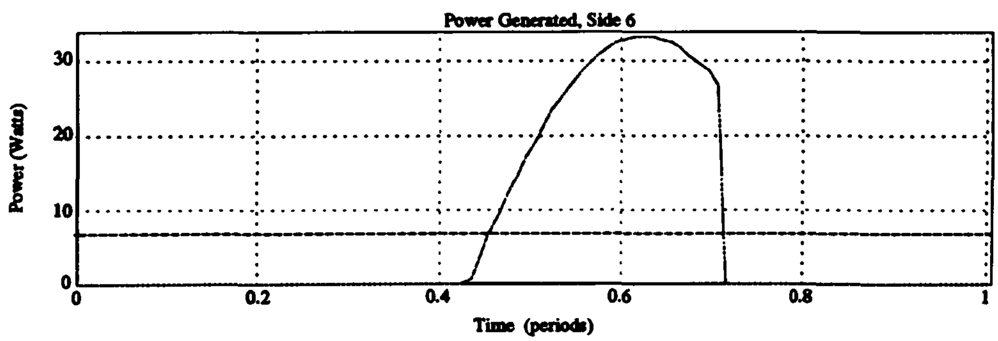

Power Generated, Side 1
Power (Watts)
Time (periods)

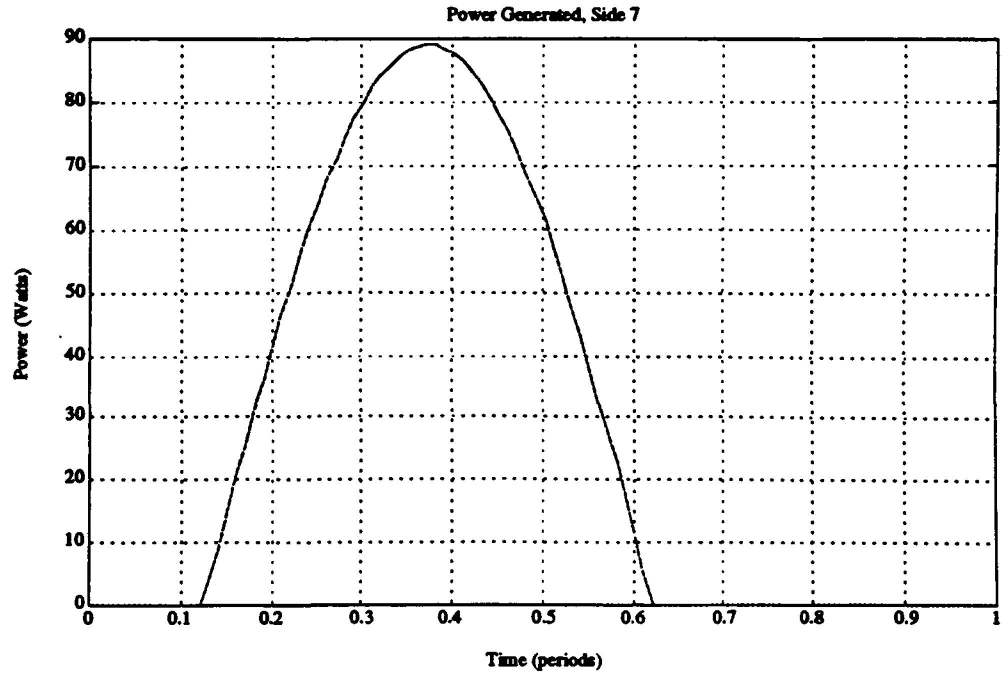

Power Generated, Side 2
Power (Watts)
Time (periods)


Power Generated, Side 3
Power (Watts)
Time (periods)

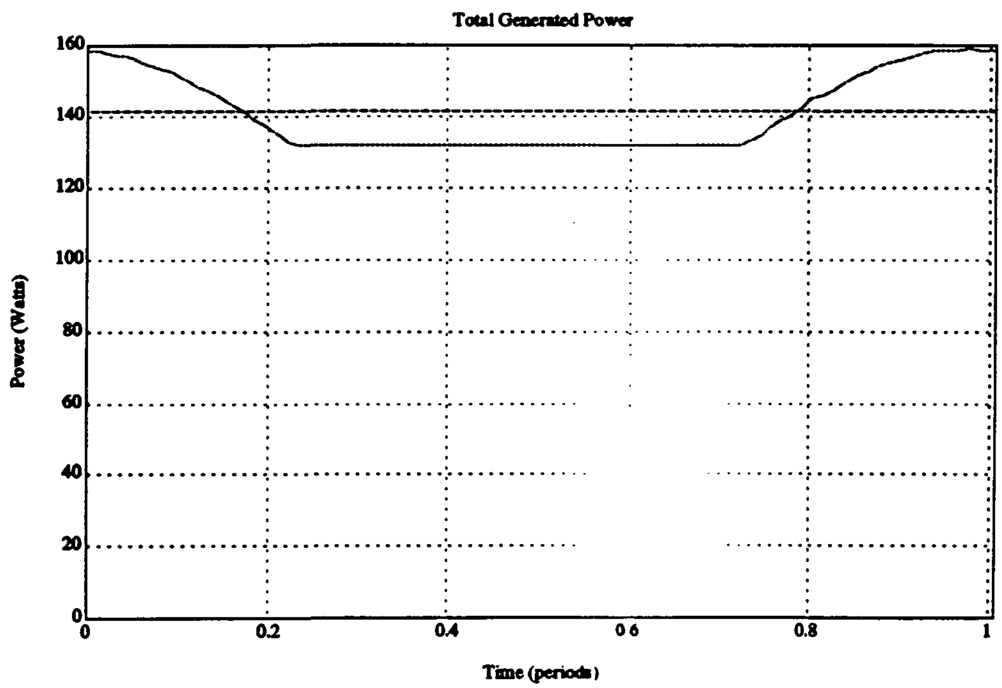

Power Generated, Side 4
Power (Watts)
Time (periods)

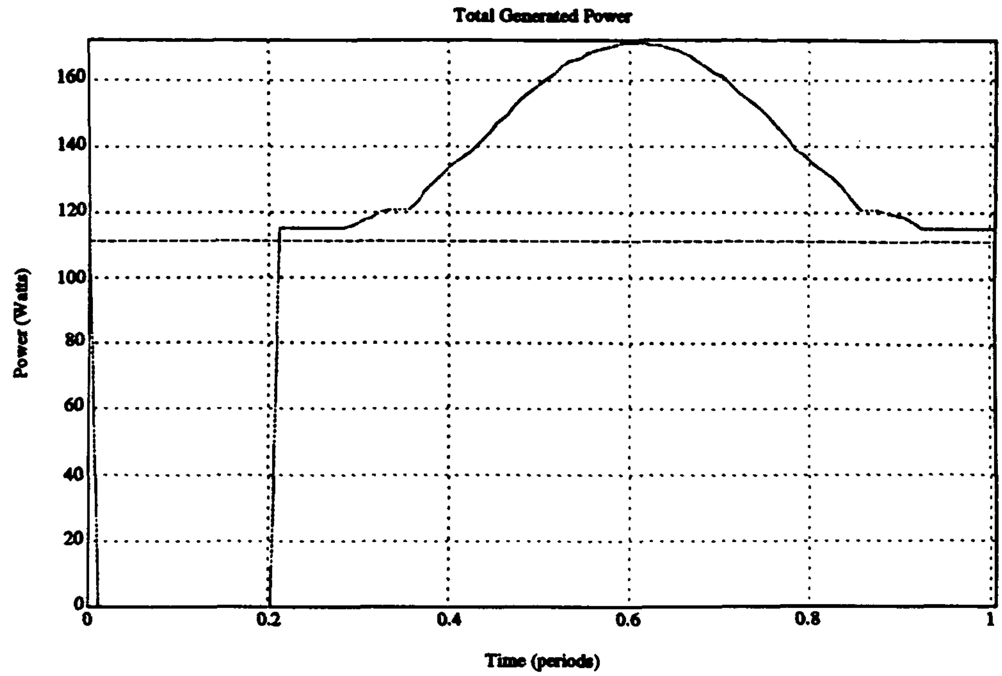

Power Generated, Side 5
Power (Watts)
Time (periods)

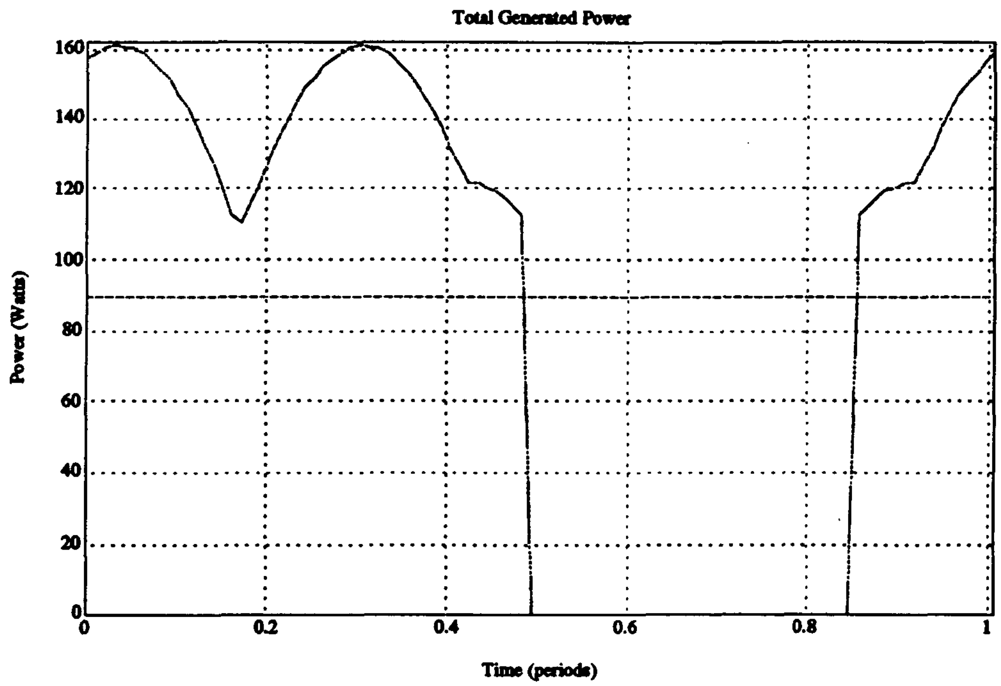

Power Generated, Side 6
Power (Watts)
Time (periods)



The above 7 plots are of the power generated by each of the solar panels (6 sides and one top), as well as the average power of each panel.







The three above plots are for the other orientation sun-synchronous orbits, and their total generated power. Respectively, they are terminator, 8-20, and noon-midnight.

C. SCIENCE

Brief Description:

The Science Subsystem will employ two Digital Array Scanning Interferometers (DASI's) to collect hyper spectral data of various global ecosystems. There will be two DASI's on SMARTSat. DASI 1 will cover the 0.4µm-0.9µm wavelength region to investigate the photosynthetic mechanisms of terrestrial ecosystems. DASI 2 will be sensitive to the infrared region of 1.0µm-2.5µm wavelengths. It will investigate coastal water regions for photoplankton biomass and other optically active organic compounds. Consequently, DASI 2 requires a thermionic cooler to enable efficient data collection of the infrared region. More detailed scientific plans can be referenced in the USRA Proposal, Section I, pages 6-8.

Assumptions:

- Data will not be taken by both DASI's at once.
- There is no skewing of the satellite during data collection which would result in a "smearing" of the pixels.
- The proper signal to noise ratio is attained by the CCD array.
- All calculations done for the 750km orbit altitude, which equates to a 6.69km/sec. ground track velocity.
- All spatial resolution options were calculated with a nominal 512 spectral samples per pixel.
- The entire observation target will be 100km x 100km in size.
- The sampling resolution per pixel is 12 bit, but for storage, these bits will be adaptively encoded to 8 bits per pixel.

Conclusions:

1) Spatial resolution will be 200m x 200m. It would be nice to get very fine spatial resolution, but the trade off is that our data handler can only take in so much data at once. Some of the cube sizes are enormous so another problem is storage of all the data that streams in. Space rated memory is very expensive and indeed cost prohibitive at a certain point. A trade off analysis for different spatial resolution options is included.

2) Total acquisition time for a 100km in track spatial dimension is 14.95 seconds.

3) This implies a frame rate of 33.45 Hz.

4) The cube size is 512 x 512 x 512 x 12 bits which when encoded to 8 bits per pixel makes 1.074 Gigabits of raw data per cube.

5) 128 MB of memory is required for each cube.

6) A higher sampling rate for DASI 2 is needed due to a higher density of waves in the infrared region. For DASI 2 the spatial resolution will be case dependent. It can be lowered to allow for higher sampling. When an actual mission is flown, then the scientific community will be queried for a case by case coastal zone resolution.

7) DASI 1 0.4μm-0.9μm
   DASI 2 1.0μm-2.5μm

8) DASI 2 needs to between +22 and +25 degrees Celsius.

Data and Graphs:

Trade Off Analysis for Spatial Resolution Options

Resolution	Spatial	Raw BPS	Frame Rate	Cube Size
Low	100km x 100km	430	0.07Hz	1x1x512
Med. Low	50km x 50km	1.6K	0.13Hz	2x2x512
Medium	6km x 6km	117K	1.12Hz	17x17x512
Med. High	1km x 1km	4.11Meg	6.69Hz	100x100x512
High	200m x 200m	105.2Meg	33.45Hz	512x512x512

Memory Sizes Needed

Low	4.096 Kbits	--->	0.5KB
Med. Low	16.384 Kbits	--->	2.0KB
Medium	1.184 Mbits	--->	144KB
Med. High	40.960 Mbits	--->	4.88MB
High	1.074 Gbits	--->	128MB

D. CONTROL

Brief Description:

The mission or payload often determines the need to control the attitude of the satellite. For the payload of SMARTSat, the DASI instrument requires accurate nadir pointing and orientation with respect to the orbital path. Inaccurate pointing or orientation of SMARTSat will result in distorted images. The information obtained from these distorted images will therefore not be as helpful, resulting in the mission being less successful. The original design called for gravity gradient stabilization, that has been re-evaluated.

Assumptions:

- DASI requires nadir pointing accuracy of $\leq 1°$.
- The diffraction slit in the DASI must be perpendicular within 1° to the orbital path of SMARTSat.

Conclusions:

1) In order to obtain less that 1° attitude control, gravity gradient stabilization is not sufficient, so 3-axis stabilization is necessary. This requires 3 momentum wheels and 2 torquer coils. The three wheels are necessary to efficiently control the attitude of SMARTSat about its three axes. Failure of one wheel will still allow 3-axis control, but not as efficient. The two torquer coils (AeroAstro) are needed to dump the momentum wheels. The torquer coils are built redundant, so having a third torquer coil serves to even further enhance the control reliability.

2) Sensors needed to operate the momentum wheels and the torquer coils include a scanning horizon sensor, a sunsensor, and a magnetometer.

Data and Graphs:

3-Axis Stabilization

3 momentum wheels (~3 lb. each)

2 torquer coils (AeroAstro)

Sensor	Accuracy
Scanning Horizon Sensor	.1° to 1°
Sun Sensor	0.005° to 3°
Magnetometer	0.5° to 3°

The momentum wheels provide about 4 Nm or torque and consume about 15-20 W of power when in use.

E. COMMUNICATIONS

Brief Description:

The Communications Subsystem will reliably transmit all the data to the ground stations. It did not undergo any major modifications for the redesign. Transmission of a full raw image cube is preferred by the scientists so that all post-processing (FFT's, Field flattenings, Correlations etc.) can be done on the ground. We will use AeroAstro's HETE communication subsystem design because it is reliable and has already been flown. It is a 2 watt, S-band transceiver with an effective data throughput of 750Kbps.

Assumptions:

- Using HETE transceiver.
- 2.092 GHz uplink, 2.272 GHz downlink, BPSK modulation.
- 750Kbps data rate, 1Mbps raw data rate, (includes headers and encoding bits).
- Ground station supplied by AeroAstro.
- From this orbit there will be 4-5 passes per day over each ground station.

Conclusions:

1) Uplink connections are not a problem due to the high power of the ground station transmissions.

2) At a five degree inclination from the horizon there is an 11.67dB margin of required Eb/No (Energy per bit/Noise density ratio) for the uplink. That is enough to statistically guarantee 99.99% uninterferred transmission due to fading losses (see Approximate Interference Fading Distribution Chart)

3) For the downlink there is only a 4.03dB margin of Eb/No at five degrees of inclination from the horizon. This statiscally correlates to 99.00% of uninterferred transmission. Of course it gets better as the satellite comes closer to the ground station.

4) The probability of error for the BPSK downlink is 6.0×10^{-4} maximum (at five degrees), this is acceptable since the industry standard is for a minimum 1.0×10^{-4} probabilty of error. (see Probability of Error Performance for BPSK Modems)

5) Assuming reliable transmission can only occur after five to ten degrees of inclination, this correlates to a maximum transmission time of 12.55 minutes for a zenith pass.

6) A full uncompressed image cube will take 23.87 minutes to transmit. With a 2X compression ratio that time gets reduced to 11.93 minutes, and a 4X ratio yields a 5.96 minute transmission time of an image cube. The scientists would prefer no compression if possible to avoid lossy effects.

7) Another possibility to reduce transmission time would be to store and forward only one side of the interferrogram per frame. But that reduces the accuracy of the data and provides no check values. It would have the virtue of cutting the data in half though.

8) With multiple ground stations, an image cube can be fully transmitted in a day. But software must be written so that the data can be sent down in variable programmable block sizes.

Data and Graphs:

SMARTSat Uplink Budget
(S-Band)

earth rad (km)	6370	*earth_rad*				
c (Gm/s)	0.3					
Eb/No at 1e-6 (dB)	14.20					
carrier freq (GHz)	2.0920					
wavelength (m)	0.14	calculated				
Tx dish diameter (m)	1.73					
Tsky (deg. K)	135	*Tsky*				
line loss to LNA(dB)	3	*line_loss*				
LNA noise temp(K)	75	*Tlna*				
system noise (K)	499	*noise_temp*				
Tx power (W)	35	*Tx_power**		*measured at input to diplexer		
PA cable loss (dB)	0.4					
Diplexer loss(uplink) ((	0.27					
bit rate (bps)	31250	*bit_rate*				
g/s ant. gain (dB)	28.6	*gs_gain*				
SC antenna gain (dBi)	0	*sc_gain*				
SC antenna G/T	-27.0	calculated				
Losses (dB)						
RF losses -g/s	3					
atmospheric loss	0.5					
polarization loss	0.5					
modulation loss	0.5					
demod loss	2					
pointing loss	0.30					
Lt (dB)	6.8	the sum				
orbit altitude (km)	750	*altitude*				
elevation (degrees)	5	10	15	30	60	89
alpha (radians)	1.10	1.08	1.04	0.89	0.46	0.02
slant range (km)	2674	2262	1934	1316	851	750
space dispersion (dB)	167.40	165.94	164.58	161.24	157.45	156.36
margin (dB)	**11.67**	**13.79**	**15.15**	**18.49**	**22.28**	**23.38**

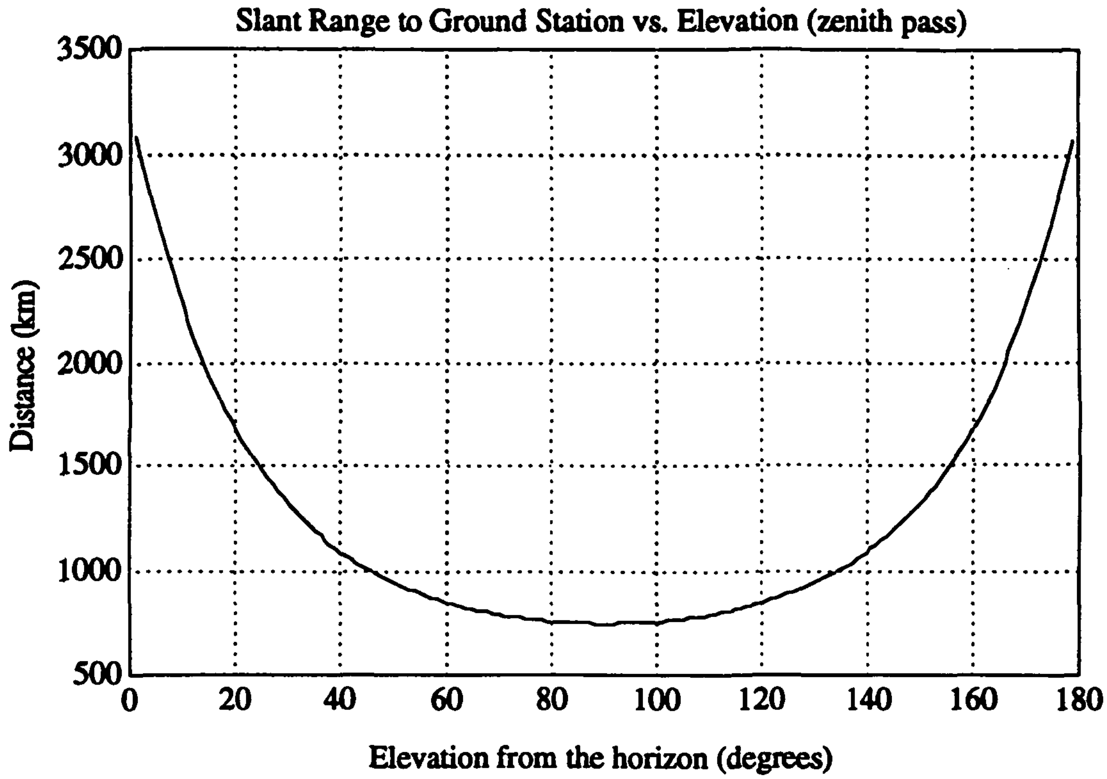
SMARTSat Downlink Budget
(S-Band)

earth rad (km)	6370	*earth_rad*	
c (Gm/s)	0.3		
Eb/No at 1e-6 (dB)	5.6	with coding	10.6 w/o coding
orbit altitude (km)	750	*altitude*	
carrier freq (GHz)	2.272		
antenna diam (m)	1.8	*ant_diam*	groundstation dish
system noise (K)	100	*noise_temp*	
Tx power (W)	2	*Tx_power*	
bit rate (bps)	750,000	*bit_rate**	
s/c ant. gain (dBi)	0	*sc_gain*	
wavelength (m)	0.13	calculated	
antenna G/T	9.68	calculated	of groundstation dish

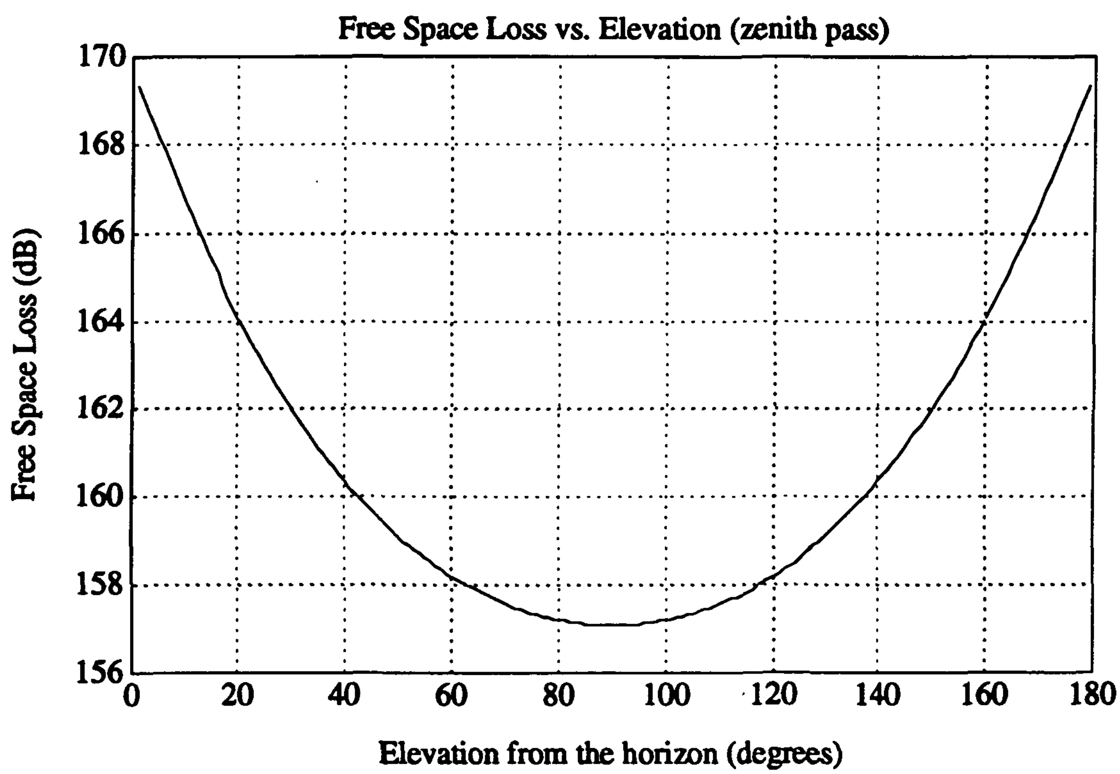
Losses (dB)		
RF losses -s/c	1	
atmospheric loss	0.5	* effective data rate is about 2% lower due to packet overhead
polarization loss	0.5	
modulation loss	0.5	
demod loss	2	
pointing loss	0.3	
Lt (dB)	4.8	the sum

elevation (degrees)	5	10	15	30	60	89
alpha (radians)	1.10	1.08	1.04	0.89	0.46	0.02
slant range (km)	2,674	2,262	1,934	1,316	851	750
space dispersion (dB)	168.11	166.66	165.30	161.96	158.17	157.07
margin (dB)	**4.03**	**5.48**	**6.84**	**10.18**	**13.97**	**15.07**

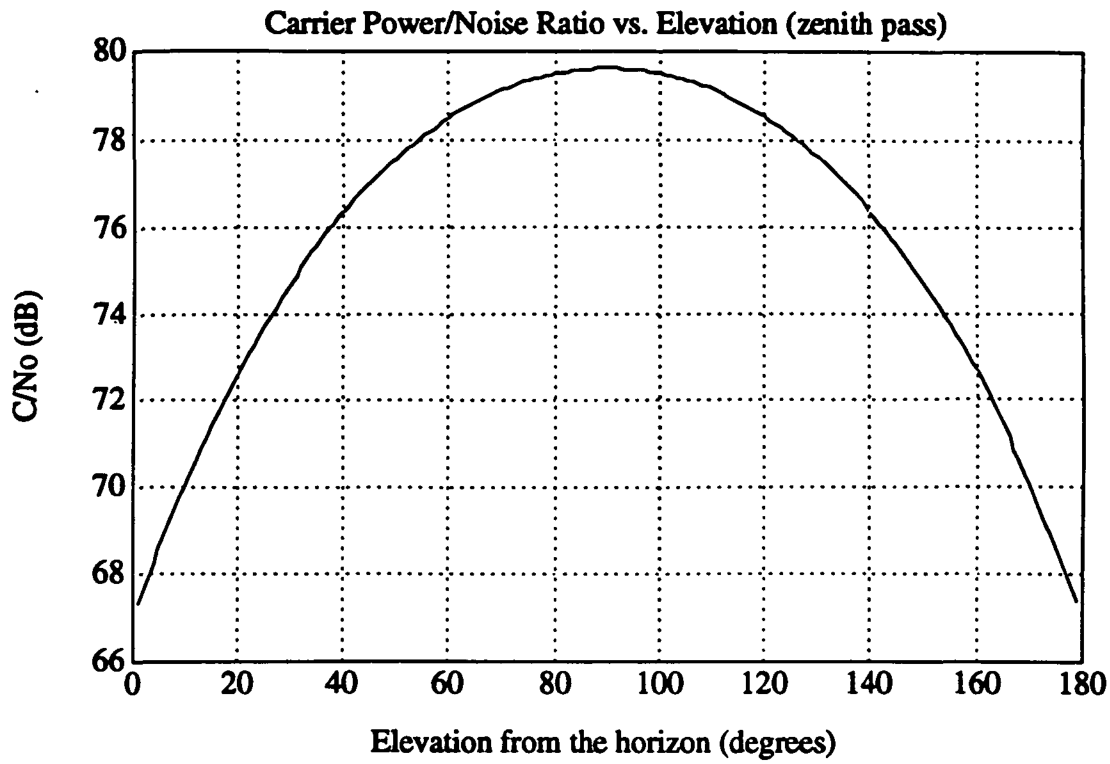
baselink (dB) 132.57 includes everything but s/c ant. gain and -20*log(range)



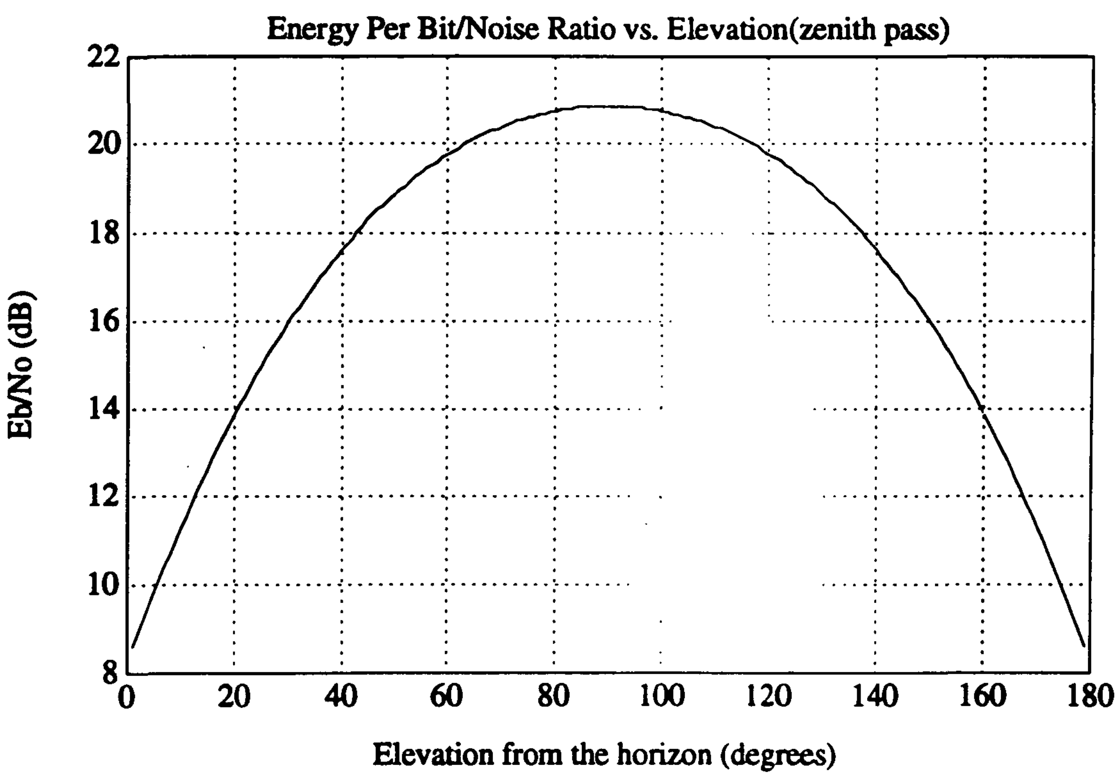
Slant Range to Ground Station vs. Elevation (zenith pass)
Distance (km)
3500
3000
2500
2000
1500
1000
500
0
20
40
60
80
100
120
140
160
180
Elevation from the horizon (degrees)



Free Space Loss vs. Elevation (zenith pass)
Free Space Loss (dB)
170
168
166
164
162
160
158
156
0
20
40
60
80
100
120
140
160
180
Elevation from the horizon (degrees)



Carrier Power/Noise Ratio vs. Elevation (zenith pass)
C/No (dB)
80
78
76
74
72
70
68
66
0
20
40
60
80
100
120
140
160
180
Elevation from the horizon (degrees)



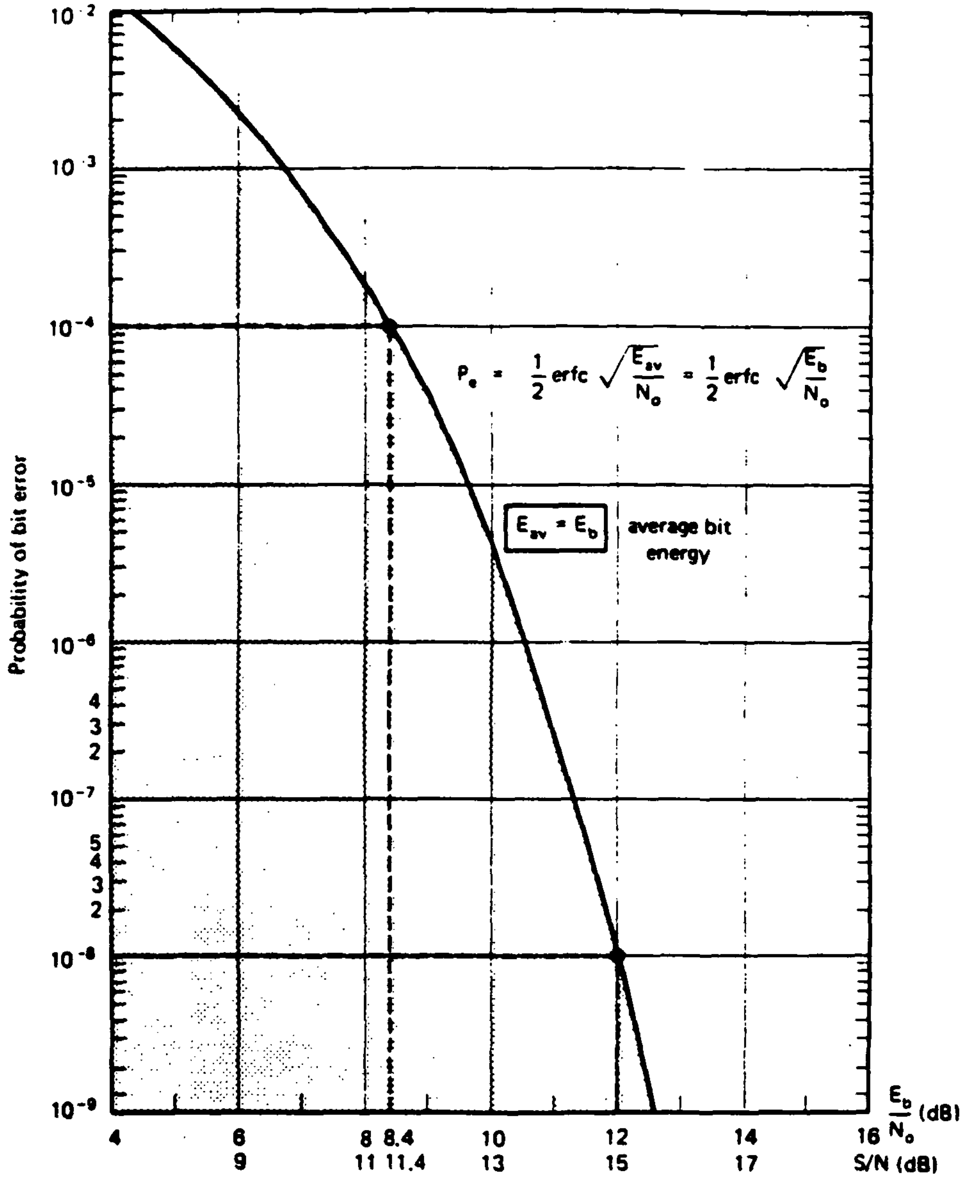
Energy Per Bit/Noise Ratio vs. Elevation(zenith pass)
Eb/No (dB)
22
20
18
16
14
12
10
8
0
20
40
60
80
100
120
140
160
180
Elevation from the horizon (degrees)

% SMARTSat COMMUNICATIONS PROGRAM (MATLAB) %

% (To calculate Eb/No, C/No, Space Loss, Slant Ranges) %

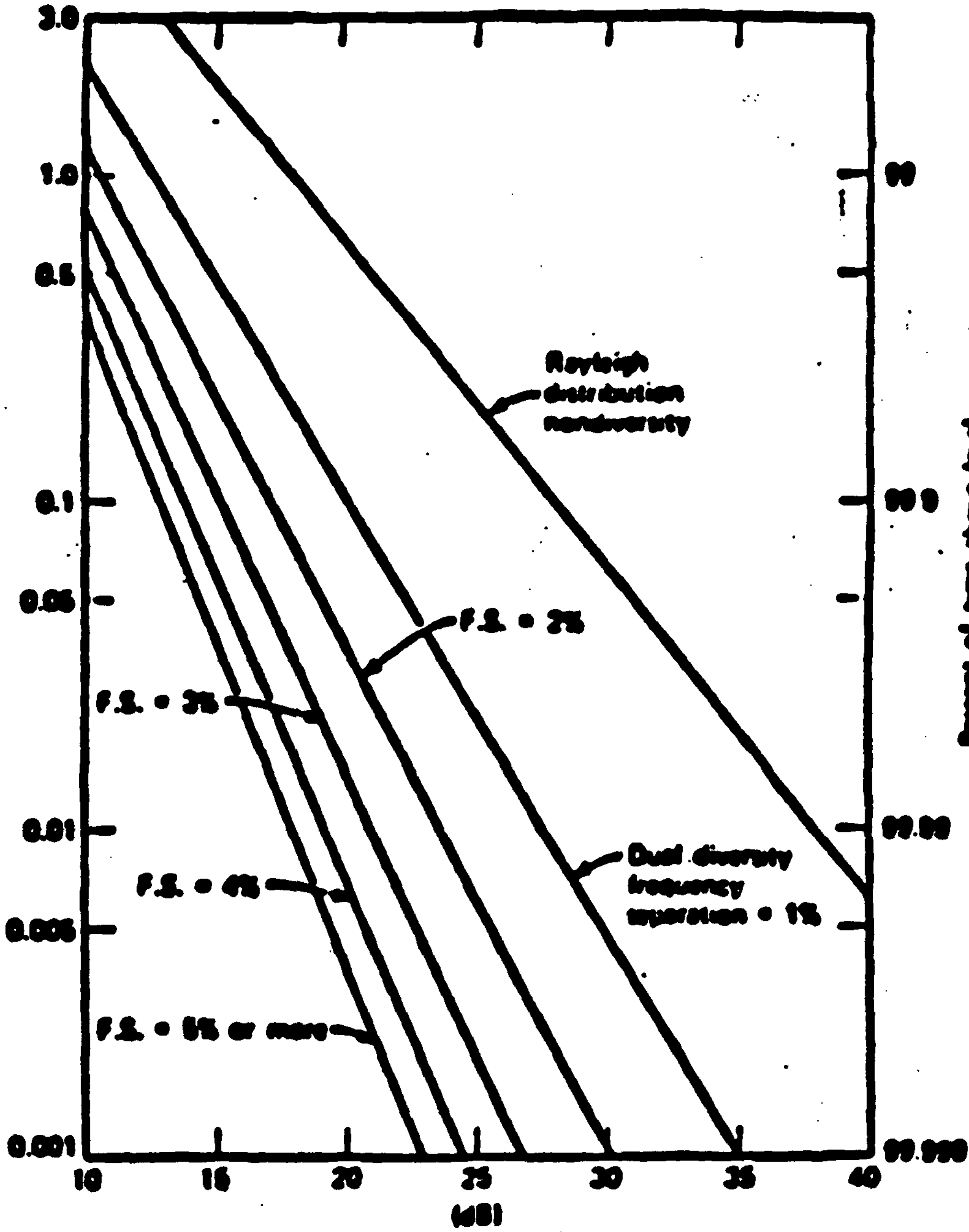
```
clear;
clg;
baud=750000;                              % Bit rate
h=750;                                    % Orbit altitude in km
r=6378;                                   % Earth radius in km
sum=h+r;
for a=1:179
  deg=a*pi/180;
  R(a)=r*cos(1.57+deg)+.5*((2*r*cos(1.57+deg))^2-4*(r^2-sum^2))^.5;
end
a=[1:1:179]';
R=1000*R';
f=2.272*10^9;                             % Frequency in Hz
L=(4*pi*f/(3*10^8))^2*R.^2;               % Path loss
Ls=10*log10(L);                           % Path loss in dB
Lt=4.8;                                   % Other losses in dB
Gs=1;                                     %Gain of satellite antenna
P=2;                                      % Satellite transmitter power in watts
boltz=-228.6;                             % Boltzmann's constant in dB
Gg=9.68;                                  % Ground station gain/temp
C=10*log10(P*Gs)-Lt+10*log10(Gg)-Ls-boltz;     % Carrier/Noise ratio
Eb=C+10*log10(1/baud);               % Bit Energy/Noise ratio
subplot(212)
plot(a,Ls)
grid
title('Free Space Loss vs. Elevation (zenith pass)')
xlabel('Elevation from the horizon (degrees)')
ylabel('Free Space Loss (dB)')
subplot(211)
R=R*.001;
plot(a,R)
grid
title('Slant Range to Ground Station vs. Elevation (zenith pass)')
xlabel('Elevation from the horizon (degrees)')
ylabel('Distance (km)')
pause
clg
subplot(212)
plot(a,Eb)
grid
title('Energy Per Bit/Noise Ratio vs. Elevation (zenith pass)')
xlabel('Elevation from the horizon (degrees)')
ylabel('Eb/No (dB)')
subplot(211)
plot(a,C)
grid
title('Carrier Power/Noise Ratio vs. Elevation (zenith pass)')
xlabel('Elevation from the horizon (degrees)')
ylabel('C/No (dB)')
```

Probability of Error Performance for Coherent BPSK Modems



Approximate Interference Fading Distribution

(This is a statistical analysis of fading losses due to atmospheric effects, multipath errors, signal refraction through clouds etc.)



F. STRUCTURE

Brief Description:

The Structure Subsystem must be able to withstand the launch loads. It must also protect and house all the components necessary for satellite operations. Therefore, all the necessary components should be selected before the structure is designed. The structure will be hexagonal, 42 inches in diameter and 36 inches in height (plus 6 inches for the adapter).

Assumptions:

- A Pegasus payload environment was chosen.
- All components are the same as the USRA Proposal.
- Added momentum wheels and torquer coils for three-axis stabilization.

Conclusions:

1) Weight is 242 pounds and the inertia tensor is diagonally dominant.

2) The Pegasus launch vehicle was chosen because it is the cheapest of the alternatives.

3) Close mounting of the major components to the baseplate is preferred to minimize the distance components are cantilevered from the plate.

4) Hexagonal dimensions of 42 inches point to point and 42 inches high, 6 for the adapter.

5) The DASI boxes provide structural attachments between the base plate and the middle deck. This provides more longitudinal stability.

6) The middle deck will provide enough space for the momentum wheels and one torquer coil. Momentum wheels will be put as close to the center of gravity as possible to simplify the control law.

7) The top deck as well as five of the sides will be solar panels, and the sixth side will act as a radiative surface for cooling.

8) Modeling the satellite on IDEAS will allow FE static load analysis, dynamic load analysis, and static thermal analysis.

Data and Graphs:

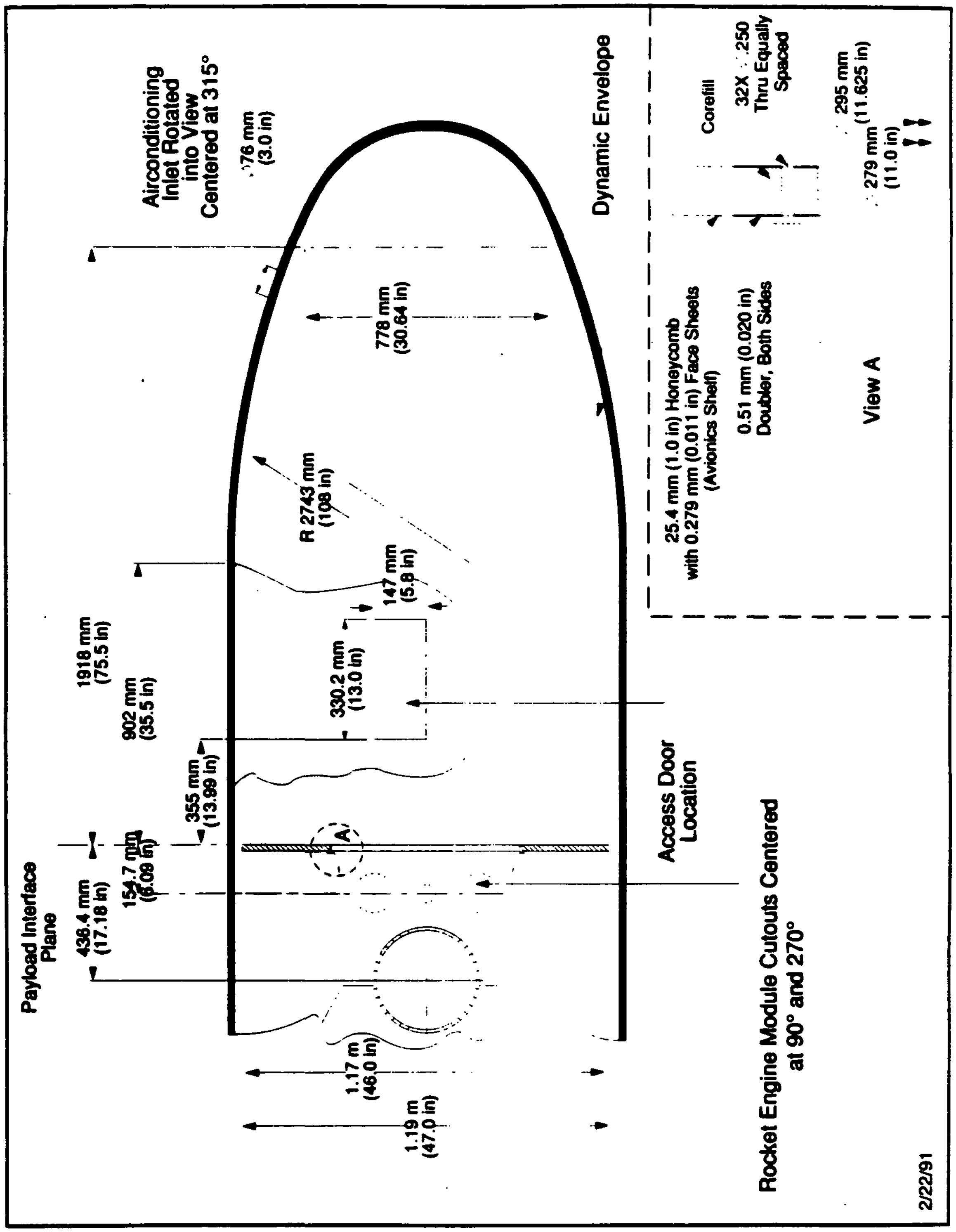
List of Components, Sizes, and Location within SMARTSat

	Dimensions of Component			Loc of Comp CG wrt the Top Base Center			
Component	length (x)	width (y)	height (z)	x	y	z	Weight
PL Data Hand	8	5	7	4	0	17.5	8
DASI	7	5	14	4	0	7	13
DASI	7	5	21	-4	0	10.5	21
Prim Comp	14	7	9	0	13.5	4.5	8
Comm Box	16	5	7	0	6.5	3.5	13.8
Power Control	14	4	5	0	-6	2.5	14.7
Battery Stack	9	3	5.5	0	-10.5	2.75	13.8
Magnetom	1	4	1.5	18	0	0.75	0.5
Magnetom	1	4	1.5	-18	0	0.75	0.5
Diff GPS	5	8	2	14	0	4	3.5
Diff GPS	5	8	2	-14	0	4	3.5
Hex	Point to Point		Height				
Alum Base	42		0.5	0	0	-0.25	20
Middle Shelf	42		0.5	0	0	21.25	20
Roof	42		0.5	0	0	35.25	20
Hex Shell	Point to Point	Thickness	Height				
Cells+Shell	42	0.5	36	0	0	17.75	50
Cylinder	Radius		Height				
Torquer Coil	11.5		2	0	0	-1.5	3.5
Torquer Coil	6		2	0	0	26.5	3.5
Mom. Wheels	8		4	0	0	24.5	9.9
Super Zip	12.5		6	0	0	-3.5	15

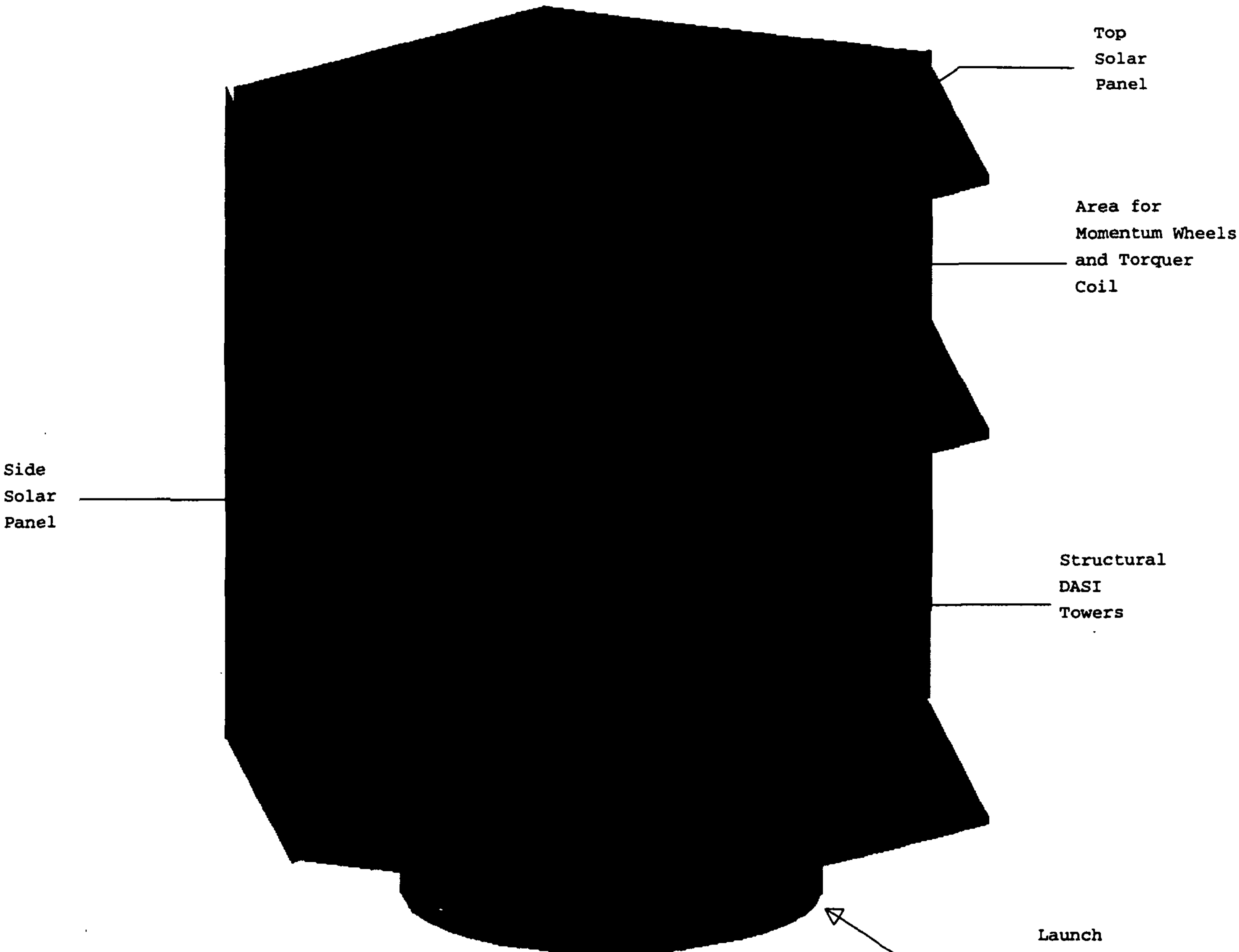
Preliminary Location of the CG and the Moment of Inertia Tensor

Total Wgt(lbs)	Location of CG wrt Top Base Center				
242.2	0	-0.1461602	12.09455		
Moment of Inertia					
XX is point to point of hexagon					
YY is along plane of hexagon					
ZZ is along the axis of the satellite					
Tensor style (lb-in2)			Tensor style (kg-m2)		
47178.5456	-1.776E-15	42	13.8063132	-5.198E-19	0.01229087
-1.776E-15	46196.9697	609.122069	-5.198E-19	13.5190652	0.17825327
42	609.122069	21998.9179	0.01229087	0.17825327	6.43775572

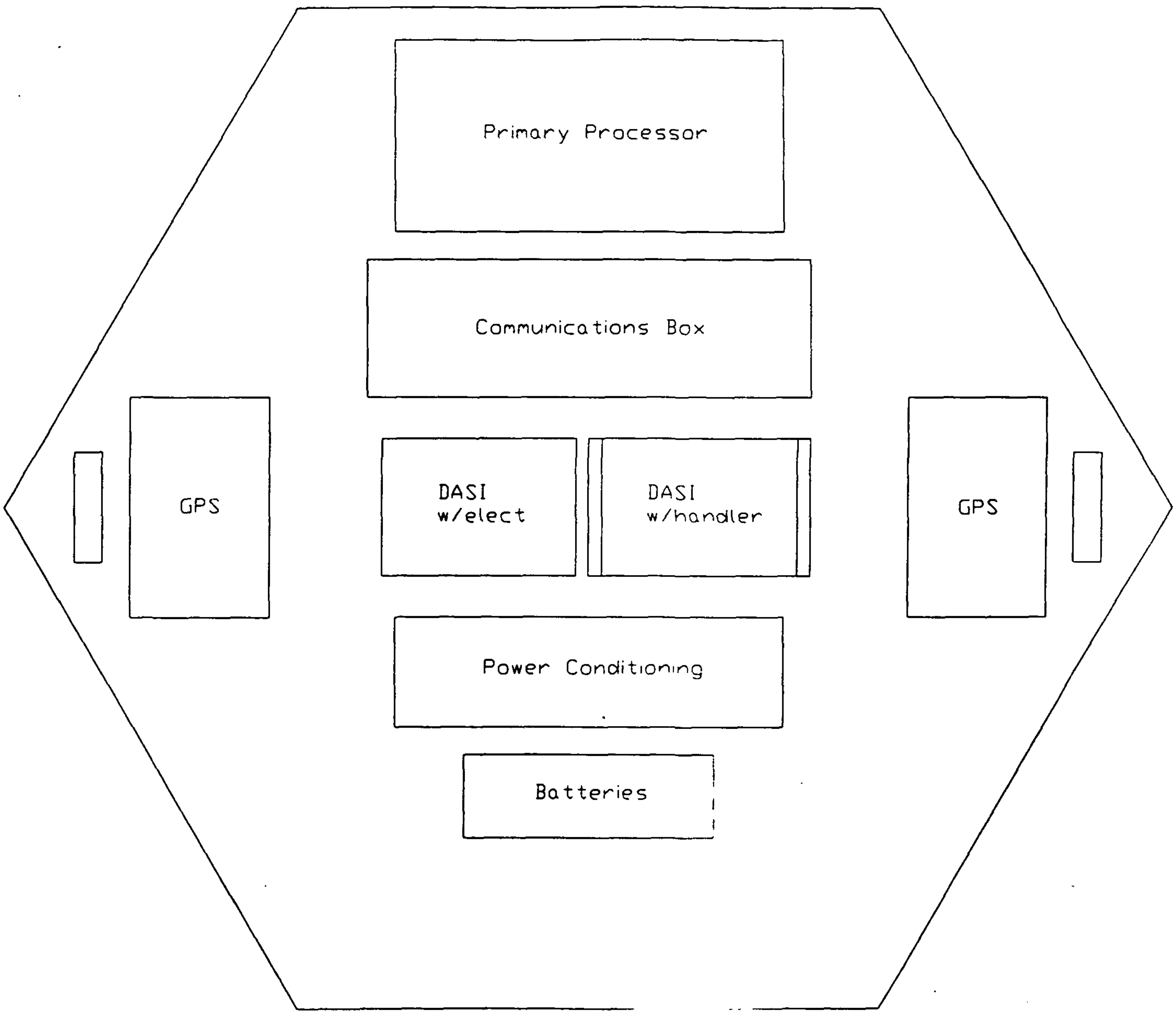
<u>Standard Pegasus Payload Fairing Dynamic Envelope</u>



3-D CAD Drawing of SMARTSat



Baseplate and Component Layout Diagram



G. THERMAL

Brief Description:

Components on a satellite require certain temperature ranges to function properly. Of the utmost importance on SMARTSat are the DASI instruments, because if the temperature is too high, the DASI's will not function adequately. To maintain proper temperature, a thermal control system should be designed, whether it is active or passive. A thermionic cooler is required to keep DASI 2 in the correct range as well.

Assumptions:

- The satellite is approximated as a sphere.
- The surface of the satellite is mostly solar cell panels, 5 sides and the top.
- The base plate is polished aluminum.
- SMARTSat has no thermal control for initial analysis.

Conclusions:

1) Worst case steady-state hot temperature without thermal control is 46° C.

2) Worst case steady-state cold temperature without thermal control is -75° C.

3) With a radiative surface of about 0.4 m^2, the worst case hot temperature is reduced to 35°C. This radiative surface is approximately the size of one of the side solar panels. Therefore one of the sides can be a radiative surface. Additionally, since one side of SMARTSat does not face the sun, this location is ideal.

5) The DASI instrument requires a lower temperature to operate, but the thermionic cooler has not been considered in the preliminary thermal design.

6) Other methods to thermally isolate the interior of SMARTSat have not been considered in this analysis.

7) To transfer heat from hot areas to the radiative panel, heat pipes should be considered as an efficient transport.

Data and Graphs:

Analysis from Wertz and Larson, Space Mission Analysis and Design.

Step 1

Components	Typical Temp Ranges °C
Electronics	0 to 40
Batteries	5 to 20
Solar Panels	-100 to 100
Structures	-45 to 65
IR detectors	-200 to -80

Step 2

5681.75 in^2 = Surface area of the satellite
Equivalent radius of a sphere with equivalent surface area = 0.54 meters

Step 3

Radiative and Absorptive properties of SMARTSat

Solar cell surface
$a_t = a = 0.805$
$e_t = e_{ir} = 0.825$

Polished aluminum base plate

$a_B = 0.2$

$e_B = 0.031$

Step 4

$$T_{max} = \left[\frac{\frac{G_s\alpha}{4} + \frac{q_I\varepsilon(1-\cos\rho)}{2} + \frac{G_s a\alpha K_a(1-\cos\rho)}{2} + \frac{Q_w}{\pi D^2}}{\sigma\varepsilon} \right]^{1/4} = 46^{\circ}C$$

$$T_{min} = \left[\frac{\frac{q_I\varepsilon(1-\cos\rho)}{2} + \frac{Q_w}{\pi D^2}}{\sigma\varepsilon} \right]^{1/4} = -75.5^{\circ}C$$

Constants used for T_{max} calculation:

$Gs = 1363\ W/m^2$

$a = 35\%$

$q_I = 258$

$Q_w = 170\ W$

Constants used for Tmin calculation:

$q_I = 216$

$a = 25\%$

$Q_w = 80\ W$

Constants used for both calculations:

$r = 63.48^{\circ}$

$K_a = 0.992$

$a = 0.805$

$e = 0.825$

$s = 5.67 \times 10^{-8}$

Step 5

46° C is hotter than desired maximum temperature
-75° C is colder that desired minimum temperature

Step 6

$$A_r = \frac{Q_w}{\sigma \varepsilon T^4}$$

Assume that T is about 35°C, which results in $A_r = 0.40\ m^2$.

Step 7

T_{low} = -17°C
with Q_w = 80 Watts

T_{low} is the lowest temperature of the radiator surface during worst case cold conditions.

H. ENVIRONMENT

Brief Description:

The hazards of the space environment must be investigated to determine any possible risk factors. The two main concerns with the low earth orbit environment are radiation and orbital debris. Extensive radiation studies have been done to estimate the total dosage as a function of time and mils of aluminum shielding. Also, there have been studies performed to estimate the average flux on a satellite due to space object populations. The results of both such studies, in relation to SMARTSat, are presented here.

Assumptions:

For the radiation dosage calculations...

- The solar event is a one time event over the course of a year.
- 98 degree inclination orbit at an altitude of 780km.

For the average flux of orbital debris calculations...

- The variation of flux with altitude is based on the smoothed altitude space distribution (see Flux Resulting from U.S. Space Command Data, page 42).
- The inclination distribution of the space object population is the same at all altitudes and is the same as the 1985 Norad Data Base distribution.
- The effective area of impact was taken to be 0.8445 square meters at an orientation angle perpendicular to the orbit vector.

Conclusions:

1) With 50 mils of aluminum shielding between the sensitive computer chips and space, the total radiation dosage is 1.75 Krads/Si/Yr. With a solar flare event, that number rises to 2.25 Krads/Si/Yr. This is a relatively benign environment (see Natural Space Environment, page 41).

2) The South Atlantic Anomaly effects have been smoothed out by the total yearly dose calculations, but it is not considered to be a problem in a per orbit basis either.

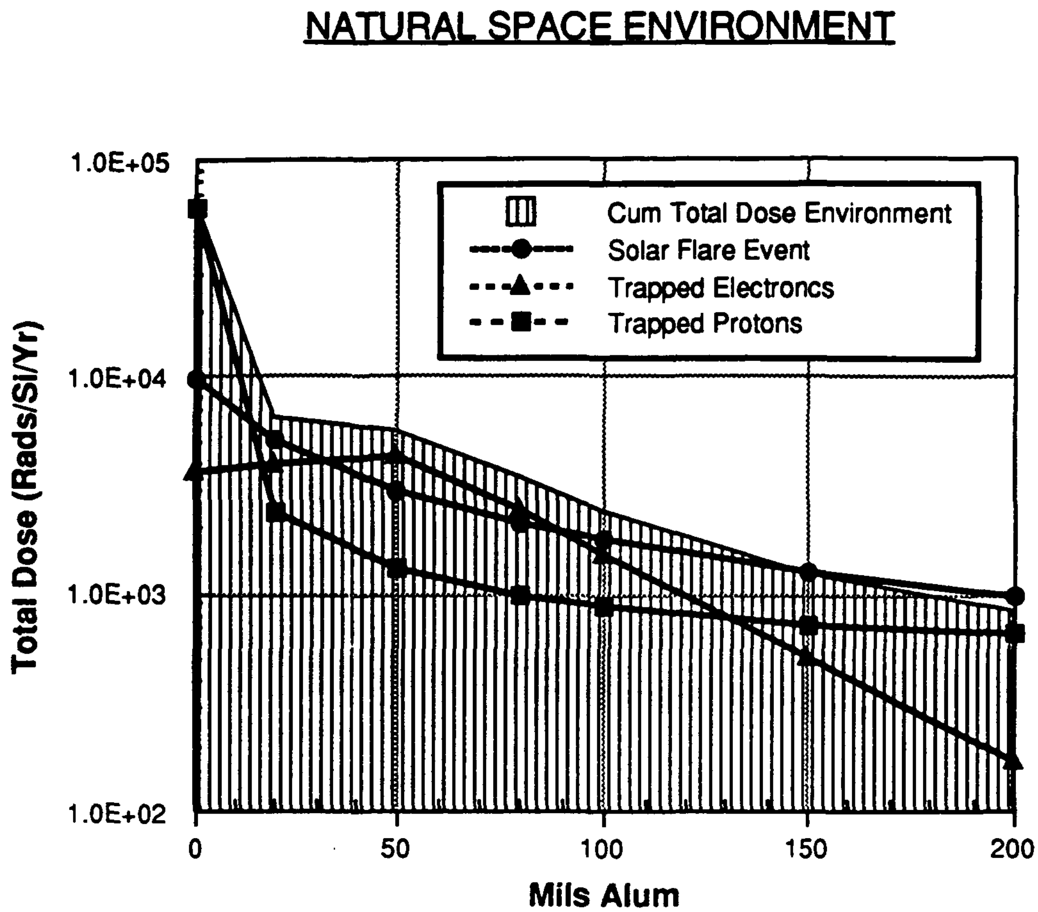
3) With radiation hardened parts, there should be minimal SEU's and an extremely low probability of burn out.

4) The variation of flux over one orbit increases with increasing inclination because there is more variation of the orientation of the surface being impacted as the inclination increases.

5) The flux is greater at higher altitudes because there are more objects in that region. The population of debris less than 10 cm in diameter is larger at higher altitudes.

6) The probability is 3.121×10^{-6} impacts per year of objects greater than 10 cm in diameter, this is acceptable. A collision by such an object would be mission ending.

Data and Graphs:

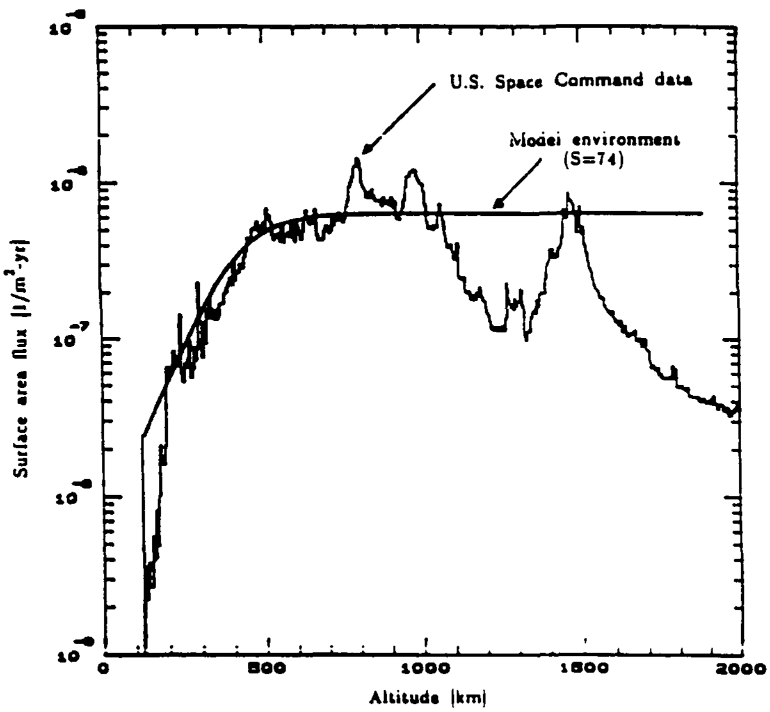


80° Inclination at 780 km orbit
The shaded areas represent the cumulative total dose per year
The solar flare event is a one time event
The solar flare total dose contribution must be added to the proton and electron total dose

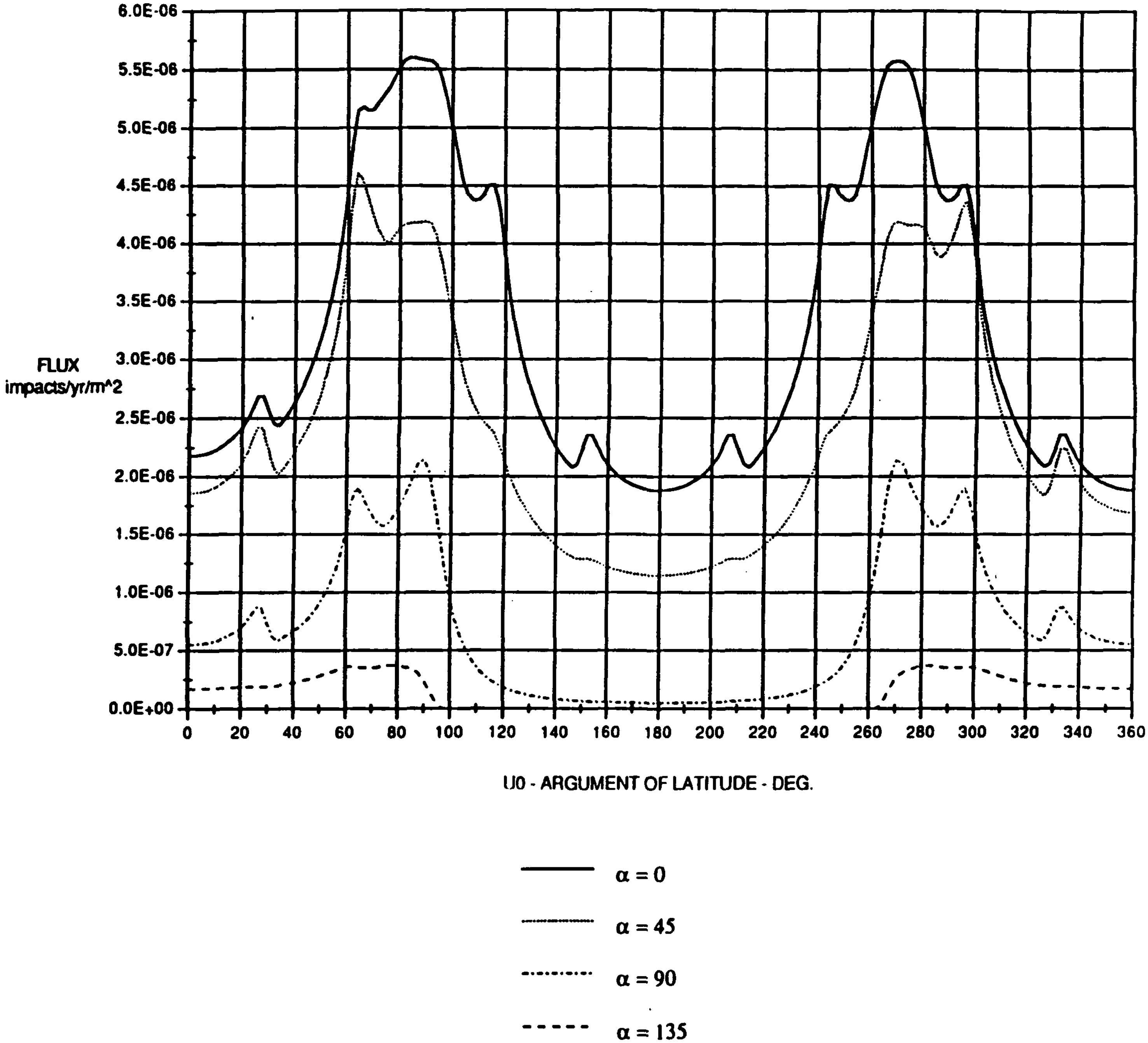
FLUX RESULTING FROM ALL OBJECTS TRACKED BY THE U.S. SPACE COMMAND

(Plotted against the model environment used in all subsequent analyses)

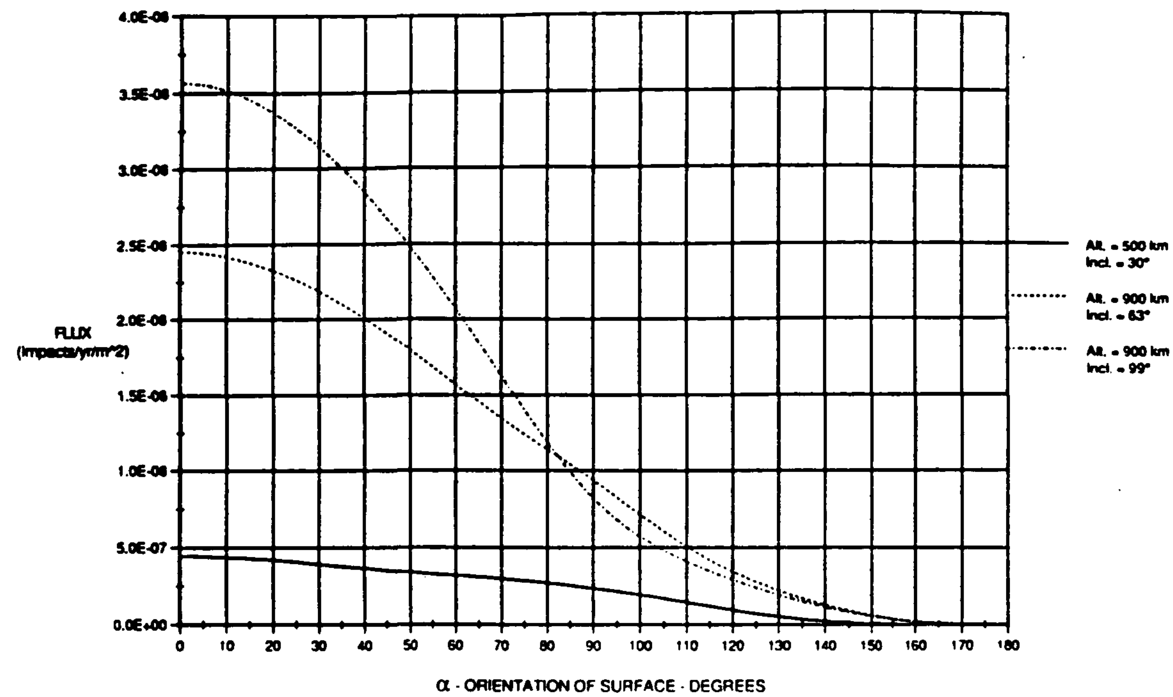
(This is for 10 cm or greater objects)



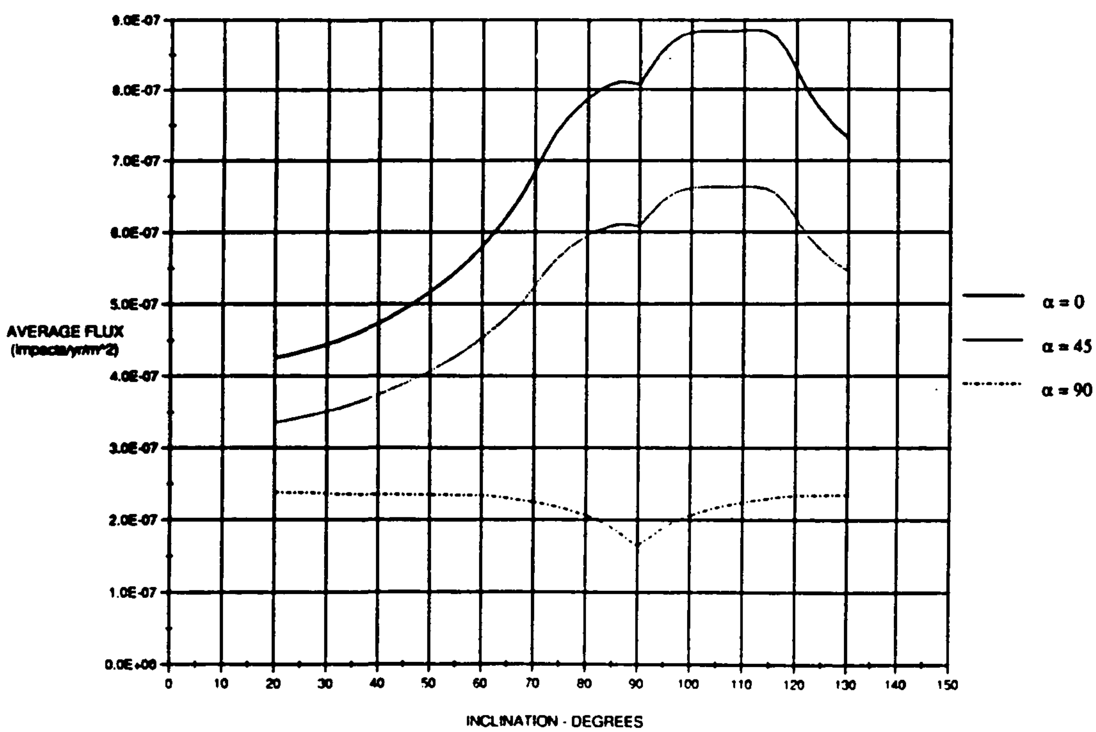
FLUX VARIATION WITH ORBIT POSITION
(Altitude = 800 km, Inclination = 99°)



Flux Variation with Surface Orientation



Average Flux as a Function of Inclination



III. COST

The cost estimates were mainly taken from the USRA Proposal, Section II, page 56. This represents Phase 2 parts and labor costs. The redesign will be slightly more expensive than the original due to the addition of three axis control and a thermionic cooler. Bottom line, the total cost of this design is $3.34 million. This represents a 17 % increase over the original $2.86 million.

What follows is the costs of parts and labor. This does not represent any possible cost overuns nor does it account for possible delays or setbacks. A nominal percentage of the total cost should be added to properly attain a feasible cost estimate for this design.

It should also be mentioned that these costs account for a mission lifetime of 1-2 years. But at the new altitude of 750km, the orbit lifetime is greatly increased and thus the mission can go on for much longer. Operations costs are fixed costs and so any additional mission lifetime would result in a simple addition of the cost per year, times some inflation percentage of course.

A reserve fund of $500,000 shouls be set aside to reduce the risk of schedule shortfalls. This value also has not been added to the total of $3.34 million, because it is quite variable and only there as a risk reduction factor.

Cost Item Space Segment	Estimate Labor (costs) $1,626,580	 Parts (costs) $1,713,016	Basis of Cost Estimate
Payload Instrument	$ 247,131	$ 230,116	Space Qual, Pluto Flyby Design
•telescope assembly		•$15,616	Tinsey Lab Quotation
•DASI optical train		•$12,000	Tinsey Lab Quotation
•Optics enclosure		•$19,500	Tinsey Lab Quotation
•CCD detectors		•$26,000	Photonic Detectors Quotation
•Thermionic Cooler		•$45,000	
•DASI electronics		•$112,000	Ball Aerospace Quotation
Research Processor	$ 85,170	$ 85,000	Harris RTX2000 space qual. cpu
Spacecraft Structure	$ 79,235	$ 12,400	Yankee Grey Experience
Payload Data Handler	$ 94,585	$ 62,000	Hubble Tel. Data Handler exp.
Primary Processor Interface	$ 70,440	$ 120,000	AeroAstro Quotation
Primary Processor	$ 49,142	$ 280,000	AeroAstro Quotation
Primary Memory		$ 90,000	AeroAstro Quotation
On-board Satellite Software	$ 149,200		AeroAstro Quotation
Communications	$ 32,575	$ 152,000	AeroAstro Quotation
•transmitter		•$50,000	AeroAstro Quotation
•receiver		•$62,000	AeroAstro Quotation
•antennae		•$10,000	AeroAstro Quotation
•telemetry interface		•$30,000	AeroAstro Quotation
Power	$ 36,675	$180,000	AeroAstro Quotation
•solar cells		•$90,000	SpectroLabs Quotation
•conditioning circuits		•$90,000	AeroAstro Quotation
Attitude Determination	$ 151,698	$ 130,000	AeroAstro Quotation
•attitude GPS (2)		•$10,000	Trimble Navigation Quotation
•magnetometer (2)		•$20,000	Southwest Rsch Inst. Quotation
•horizon sensor (2)		•$60,000	AeroAstro Quotation
•sun sensor (2)		•$40,000	AeroAstro Quotation
Attitude Control	$ 36,483	$ 336,000	AeroAstro Quotation
•torquer coils (2)		•$66,000	AeroAstro Quotation
•momentum wheels (3)		•$270,000	AeroAstro Quotation
Gnd. Support Equipment	$ 20,000	$ 16,000	AeroAstro Quotation
Integration	$ 123,188		Exp. with satellite Yankee Gray
Testing	$ 277,162		Exp. with satellite Yankee Gray
POCC (hw,sw,training,...)	$ 8,530	$ 16,000	Exp. with ALEXIS, Pioneer
Mission Design Mission Plan	$ 10,000		Exp. with ALEXIS, Pioneer
Science Data Processing	$ 8,000	$ 14,000	Exp. with air flight data proc.

Science Data Analysis	$ 8,444	$ 8,500	3 yr. exp. with DASI instrument
Management	$ 39,500		Yankee Grey program mgmt.
Reliability	$ 36,675		Exp. with ALEXIS, Yankee Grey
Quality Assurance	$ 44,967		Exp. with ALEXIS, Yankee Grey
Launch Operations	$ 17,780		Exp. with ALEXIS, Yankee Grey

Total Estimated Cost for the SMARTSat: $ 3,340,000

IV. RESEARCHERS

Jeffrey Chan received his Masters degree in Aero/Astro from Stanford University in September of 1994, and is currently pursuing an Engineers' degree in Aero/Astro, specializing in spacecraft and satellite design. While at Stanford, he has actively studied spacecraft design, taking courses such as orbital mechanics, spacecraft design, rocket propulsion, and participation in the submission of a small satellite proposal to USRA. From the summer of 94 to the present, he has been the payload manager for the microsatellite SAPPHIRE, being designed and built by the Stanford Satellite Development Laboratory, and the lead engineer for the onboard digital camera. Currently, he is searching for an Engineer's project related to satellite or spacecraft design, so that his studies may continue.

Richard Lu received his Masters degree in Aero/Astro Engineering from Stanford University in December of 1994. In addition, he holds a Bachelors degree in Electrical Engineering from UCLA. He presently is involved in the satellite activities of the Stanford Satellite Systems Development Laboratories. With his background in electrical engineering he assumed the role of communications subsystem manager for the SAPPHIRE satellite. In this program he has designed and built the entire satellite transceiver. His career interests involve satellite research and development, especially the managerial aspects of such an endeavor. In response to his desire for a technical leadership position, he has applied to the Engineering Management Department of Stanford for an additional Masters degree. He also holds the position of Vice-President of the local chapter of AIAA, and President of the Stanford Technology Management Society.

Jeffrey Y. Chan

Home Address
47 Countryside Lane,
Williamsville, NY 14221
(716) 688-2732

Current Address
425 Grant Avenue, Apt. 20
Palo Alto, CA 94306
(415) 322-7709

Objective

A position in composites and/or satellite structures design

Education

9/93 - present

Stanford University, Stanford, California
Engineer's Degree focusing on composites and spacecraft design
Expected graduation date: 6/96
M.S. Aeronautics and Astronautics, 9/94 G.P.A.: 3.83/4.00

Relevant Coursework
Spacecraft Design, Composites Design, Orbital Mechanics, Rocket Propulsion, Aerospace Structures, Compressible Flow, Dynamics

8/89 - 6/93

State University of New York at Buffalo
B.S. Aerospace Engineering G.P.A.: 3.75/4.00
Graduated Summa Cum Laude

Work Experience

6/94 - present

Payload Manager
Stanford University, California
Overlook the payload subsystem of the SAPPHIRE satellite being designed by Stanford Aero/Astro SSDL. Engineering lead of the on-board digital camera.

9/94 - present

Research Assistant
NASA/Ames, Mountain View, California
Baseline design of a 3-axis stabilized satellite platform for a scanning interferometer, used for agricultural purposes.

9/93 - 9/94

Research Assistant
Stanford University, California
Design of composite satellite bus for USRA proposal.

6/92 - 8/92

Research Assistant
Syracuse University, Syracuse, NY
Implementation of CFD code on parallel architecture supercomputer

Honors & Activities

David Sen-Lin Lee Fellow 1993-94
Assistant Principal Second Violin
of Stanford Symphony Orchestra 1994-95
Avid Downhill Skier
NDSEG Fellowship Honorable Mention
Member of Tau Beta Pi
Member of Sigma Gamma Tau
Honors Scholar at State University of New York at Buffalo

References

Will be furnished on request

RICHARD ADAM LU

Escondido Village #143C	1524 Bridges Court
Stanford, CA 94305	Fremont, CA 94536
(415) 497-5366	(510) 794-5274
alucneat@leland.stanford.edu	

OBJECTIVE: A position involving space systems engineering; satellite communications.

EDUCATION:

Expected: 12/94 — **STANFORD UNIVERSITY**, Stanford, CA.
M.S. Aero/Astronautical Engineering

9/88-12/92 — **UNIVERSITY OF CALIFORNIA AT LOS ANGELES**, Los Angeles, CA.
B.S. Electrical Engineering

EXPERIENCE:

6/94-present — **NASA AMES RESEARCH CENTER**, Moffett Field, CA.
Research Assistant, Satellite Subsystem Manager - GPS Payload:
- Involved in the design of the SMARTSAT satellite, system integrations, and organization.
- Evaluated engineering trade-offs with the technical, cost, and scheduling constraints.
- Helped to write the initial government procurement proposal.

1/94-present — **SATELLITE SYSTEMS DEVELOPMENT LABORATORIES**, Stanford University.
Satellite Subsystem Manager - Communications:
- Subsystem work scheduling, budget management, component acquisitions, and system integration.
- Design, fabrication, and testing of the entire satellite UHF communication system.
- SAPPHIRE Project, satellite systems engineering and design experience.

6/94-present — **SKYWATCH INFORMATION SYSTEMS INC.**, Sunnyvale, CA.
Technology Commercialization Intern:
- Cellular communications research, global trend forecasting for the purpose of a new product.
- Ames Technology Commercialization Center related projects.

1/93-9/93 — **SPACE PROJECTS GROUP**, University of California at Los Angeles.
Electrical Engineer:
- Designed, fabricated, and tested a pressure and thermal diagnostic system for rocket propulsion tests.
- Altius Project, rocket sub-system interaction and integrations.

10/92-9/93 — **INSTITUTE OF PLASMA AND FUSION RESEARCH**, Los Angeles, CA.
Engineering Aide, Machinist:
- Fabrication of electronic circuits for data acquisition systems, power system design.
- Set up experiments with the Tokamak high energy plasma reactor, microwave diagnostics.
- Machining of materials and parts, metal shop technician.

9/90-9/92 — **PLASMA DIAGNOSTICS GROUP**, University of California at Los Angeles.
Laboratory Technician:
- Involved in the design, fabrication , testing, and packaging of solid state electronics.
- High frequency filter design and construction.
- Project cost assessments and component acquisition.
- Microwave reflectometry research, co-authored technical paper.
- Machining of materials.

HONORS:

Eta Kappa Nu National Electrical Engineering Honor Society
Tau Beta Pi National Engineering Honor Society
San Francisco Bay Area Engineers' Scholarship
Newpark Scholastic Scholarship

TECHNICAL SKILLS: Computer experience: PASCAL, PSPICE, NOVA, ORCAD, MATLAB software design tools, MS-DOS, UNIX operating systems; Proficiency with electronics laboratory equipment, circuit fabrication techniques, and machining.

ACTIVITIES/ INTERESTS: AIAA Vice President, Stanford; STEMS Social Committee; HAM radio operator, KE6FZO.
Interests include: Football, chess, astronomy, billiards, scuba diving.