SUNSTROKE

May 2nd, 2000

The Proposal was prepared by:

Joseph Hackel – PROJECT MANAGER/POWER
1494 Greenbriar Blvd.
Boulder, CO 80303
(303) 499-5784
JoeHackel@netscape.net

Joseph Hackel

David Waller – SYSTEM ENGINEER/ COMMAND AND DATA HANDLING
677 Walden Circle
Boulder, CO 80309
(720) 304-8836
david.waller@colorado.edu

David Waller

Brian Corff – STRUCTURES
4840 Meredith Way #201
Boulder, CO 80303
(720) 5654-0488
corff@colorado.edu

Brian Corff

Frank Friedl – PROPULSION
2905 Aurora Apt 231
Boulder, CO 80303
(303) 245-8837
friedlf@hotmail.com

Frank Friedl

Mark Hegge – ATTITUDE DETERMINTATION AND CONTROL SYSTEM
512 Buckthorn Way,
Louisville, Colorado, 80027
(303) 939-5614
mhegge@ball.com

Mark Hegge

Gretchen England – THERMAL
4494 Hamilton Ct
Boulder, CO 80302
(720) 839-8632
England@colorado.edu

Gretchen England

Reto Zingg – TELECOMMUNICATIONS
2905 E. Aurora Ave #328
Boulder, CO 80303
(303) 444-0661
reto.zingg@pop.ch

Reto Zingg

Corissa Young – MISSION DESIGN
2900 East College Avenue Apt. #14
Boulder, CO 80303
(303) 417-1019
youngca@ucsub.colorado.edu

Corissa Young

1.0 Spacecraft System Summary

1.1 Summary of Spacecraft Characteristics

The Sunstroke spacecraft will investigate the Ultraviolet (UV) characteristics of the sun at Mars for a minimum of two years. The mission shall not exceed $75 million in FY 2000 dollars and shall be launched on a target date of April 1, 2003 on a Delta II-7925.

The primary payload consists of two UV spectrometers that require both sun pointing and star pointing. Sunstroke shall be pointed at the sun during all daylight operations for direct UV measurements with data being taken for a minimum of 15 minutes. During eclipse, both spectrometers shall be pointed at a predetermined set of stars for calibration for a minimum of 20 minutes. Pointing accuracy shall be met through a 3-axis stabilized system. Four reaction wheels and 12 thrusters shall provide the necessary stability, momentum dumping, and slew rates. Two star trackers, two Inertial Measurement Units (IMU), four Coarse Sun Sensors (CSS), and one Fine Sun Sensor (FSS) shall provide attitude knowledge.

Power shall be provided by two 18 Amp-hr Eagle Pitcher batteries and three solar array assemblies using TECSTAR triple junction $GaInP^2/GaAs/Ge$ cells. A Spectrum Astro Charge Control Board (CCB) and Power Control Board (PCB) shall be used for charging and power distribution.

Propulsion shall consist of a bi-propellant blowdown system. This system will use positive expulsion, reversing diaphragm tanks for the fuel. Twelve thrusters will be used to control the spacecraft through three axis in two directions each, with coupled thrusters to provide balanced moments. The main engine on which design was based is a 490 Newton engine provided by Kaiser-Marquardt.

The Sunstroke structure shall consist of an eight sided stringer design with aluminum honeycomb closeout panels and composite honeycomb decks. All components shall be internally mounted, excluding the instruments requiring outside field of views such as the ADCS sensors and science payload. The primary structure shall be mated to a tapered cylindrical propulsion module consisting of the main engine, ADCS thrusters, and propellant tank assemblies. The propulsion module will also act as the interface to the launch vehicle. Three orthogonal fixed solar array assemblies shall be mated to the topside of the structure with a deployable antenna boom completing the fourth side.

Multi Layer Insulation (MLI) shall be primarily employed to thermally control the spacecraft. All high power internal components shall be mounted with at least one side attached to a radiative surface. Additionally thermistors and heater strips shall be employed to regulate the temperatures of the batteries, propellant tanks, thrusters, and electronic equipment.

Commanding shall be performed by a bus style architecture using VME format. The main processor shall be a Harris Labs Standard Spacecraft Processor Module capable of

31 MIPS at 20 MHz. Attitude control shall use the Spectrum Astro Payload and Attitude Control Interface (PACI) VME card. Additionally, the Spectrum Astro Charge Control Board and Power Control Board cards shall be utilized. A VME memory module consisting of EDAC compliant DRAM and payload interfaces shall be contracted though Harris Labs.

Telecommunications will be provided by two low gain hemispheric antennas and one high gain 1.5 meter parabolic dish. The high gain antenna shall be articulated using a 2-axis gimbal provided by Moog. Two X-band Motorola transponders shall be used with two 50 W travelling wave tube amplifiers. The transponders provide Reed-Solomon encoding and decoding.

1.2 System Block Diagram

Figure 1.1 depicts the system interfaces and connections.

Page left blank for system foldout diagram

1.3 System Drawings

1.3.1 Envelope Dimensions
Figure 1.2 shows the overall envelope dimensions of the spacecraft.

1.3.2 Instrument Mechanical Interfaces
Figure 1.3 shows the placement and field of view of the UV spectrometer payload. It should be noted that the fine sun sensor and hemispherical antenna are mounted on the same bracket for field of view purposes.

1.3.4 Launch Vehicle Bus Interface
The Sunstroke spacecraft shall interface to the Delta II third stage through the 3712A Payload Attach Fitting (PAF). The clamp assembly and four separation spring actuators shall be connected to the lower portion of the propulsion module. Figure 1.4 shows the spacecraft stowed within the 2.9 meter fairing. The fairing access door location will be specified upon further refinement of the design. The umbilical location shall be defined upon further refinement of the design. It shall provide battery trickle charge power, serial telemetry, and launch vehicle S/C separation sense.

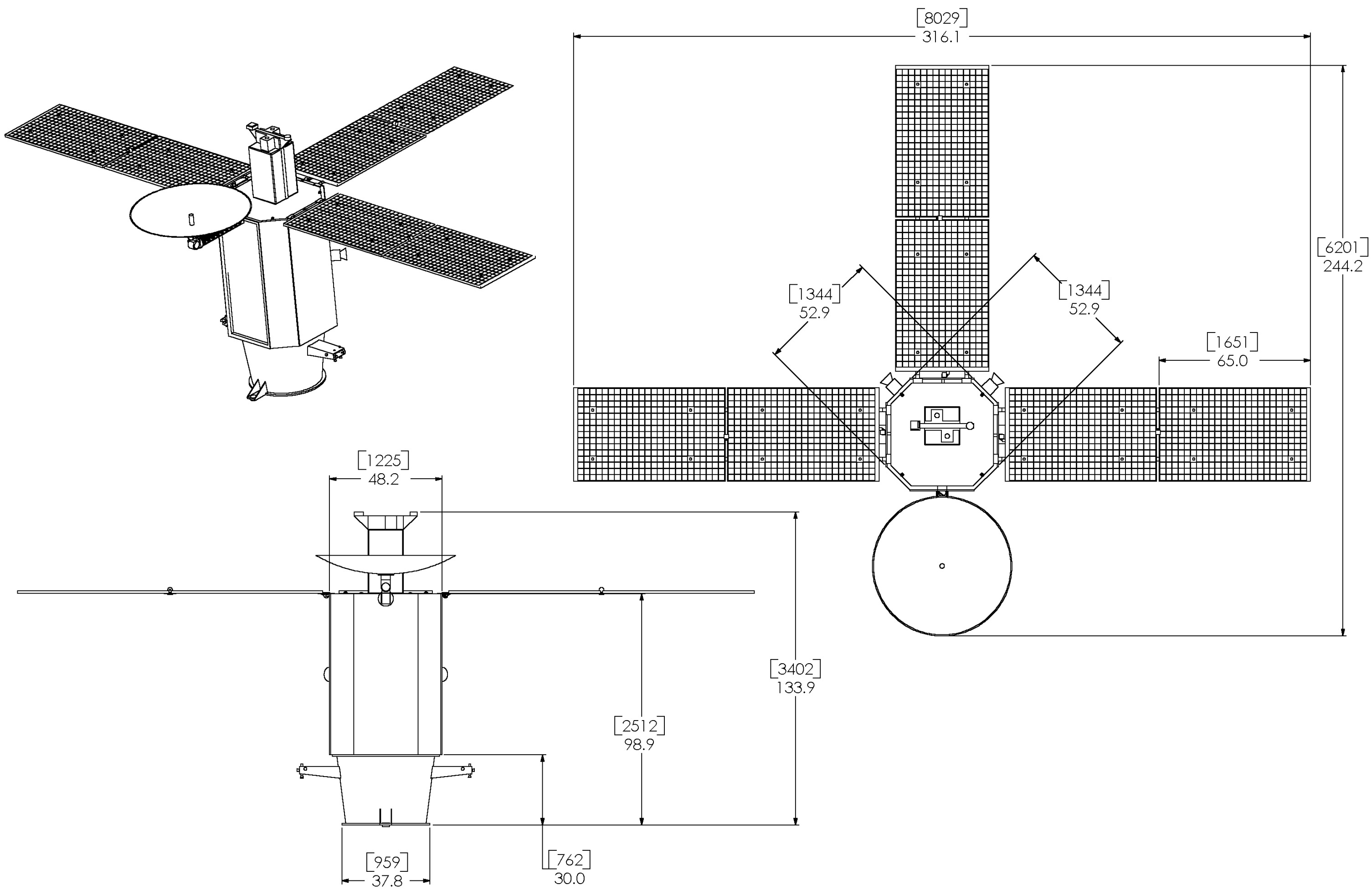


Fig 1.2 Sunstroke Envelope Drawing

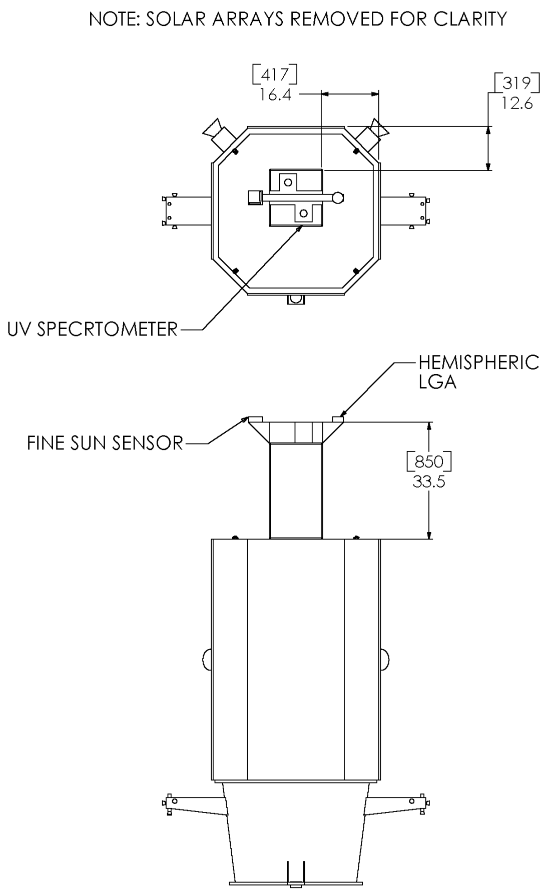


Fig 1.3 Instrument Mechanical Interface

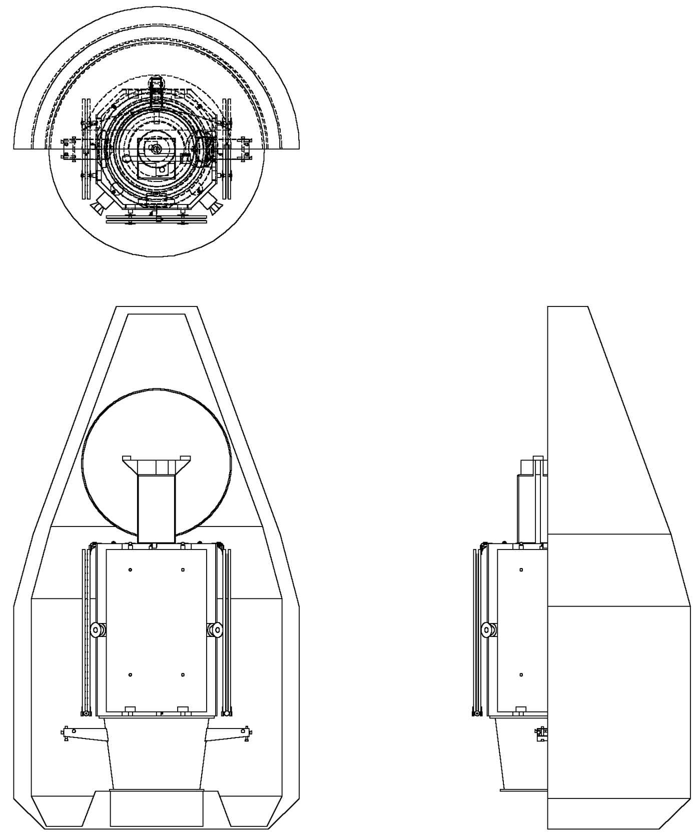


Fig 1.4 Launch Vehicle Interface

1.4 Power Profile

The power profile is broken into the launch phase, cruise phase, and orbital phase. The detailed listing of power requirements can be found in section 6.0 of this proposal.

1.4.1 Launch Power Profile

The spacecraft will not be powered upon launch. Telemetry switches will signal the spacecraft upon separation from the second stage. At this point a considerable amount of one time power will be needed to activate the solar array deployment mechanisms. The table below outlines the power needs during this phase.

Table 1.5: Launch Power

HOURS UNTIL SOLAR CELL DEPLOYMENT	8	HR
Battery Energy Removed in launch	48.5	Watts
Battery Depth of Discharge	46%	
Power use during cruise phase	152.5	Watts
Battery Energy Removed in Eclipse	215.7	Watts
Battery Depth of Discharge	17%	

1.4.2 Cruise Power Profile

During cruise, all equipment shall be functionally tested periodically. It has not been determined if science will be taken at this phase in the mission. The solar arrays shall be sun pointed for a majority of the time and will easily accommodate the power required for communications, orbital correction maneuvers, and attitude control.

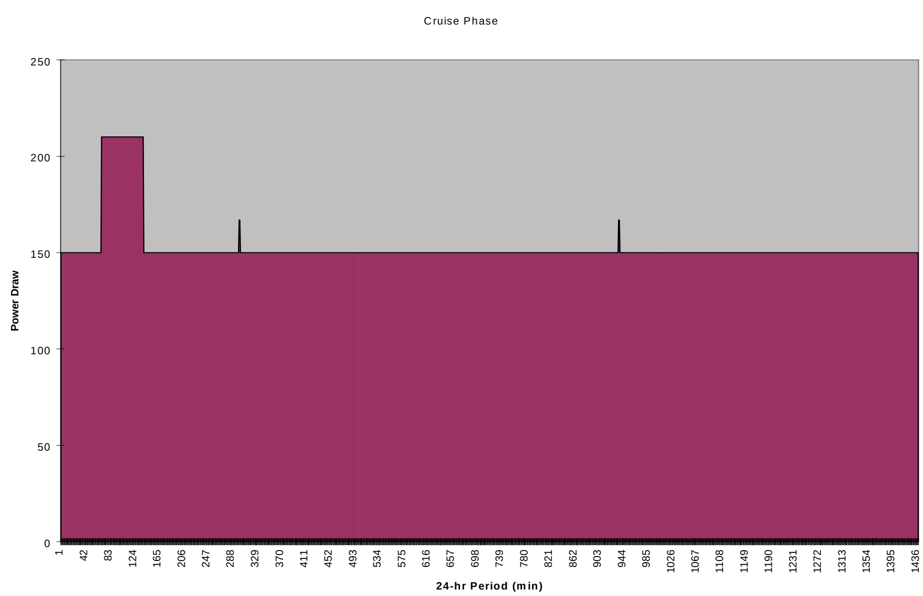


Fig 1.5 Power Profile during Cruise

1.4.3 Orbital Power Profile

In Mars orbit, the science payload will be operating for 35 minutes per orbit. The profile in Figure 1.6 is calculated at maximum distance.

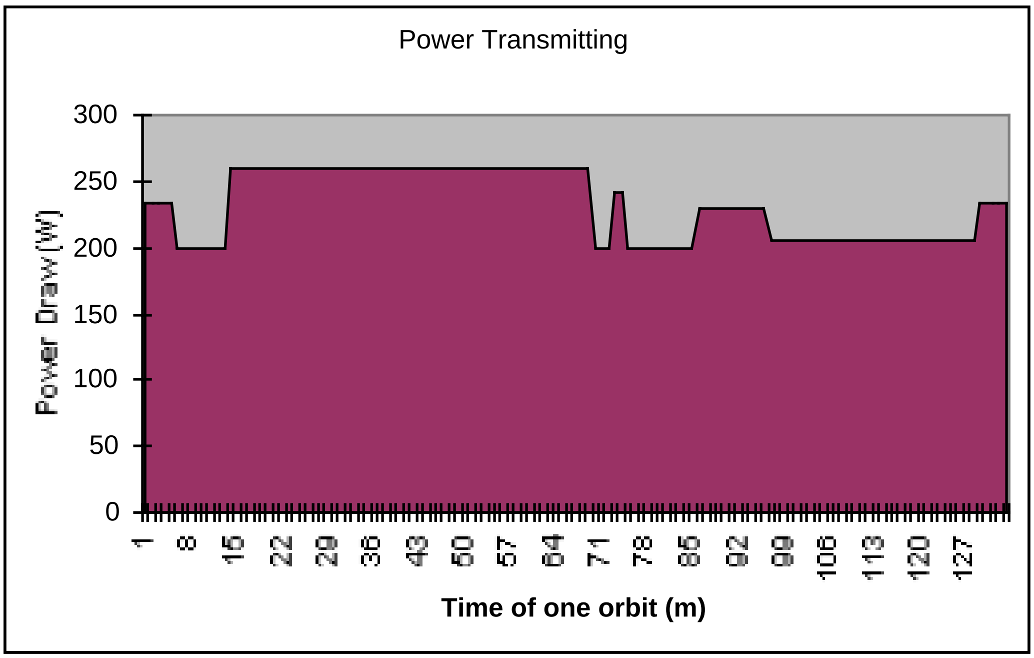


Figure 1.6. Power Profile during standard orbit

At maximum distance, a full downlink will be required approximately every 7 days. The power profile for this scenario is shown below in Figure 1.7.

1.5 Mass Statement

The following Figure outlines the mass for the entire spacecraft including 25% margin.

Subsystem and Component	Quantity	Size (mm)	Individual Mass (kg)	Total Mass (kg)	Idle Power (W)	Operating Power (W)
Payload						
SOLSTICE UV Spectrometer	2	183 x 387 x 846	18	36	N/A	16.6
SOLSTICE GCI	2	81 x 183 x 203	1.3	2.6	0	0
Command & Data Handling						
Electronics Module (Computer)	1	292.1 x 203.2 x 203.2	7.5	7.5	13.5	33
Electrical Power						
Battery	2	128.5 dia. x 563.9	9	18	0	0
Solar Array	3	(10.3 m^2 total)	N/A	12.8	0	0
Power Electronics Board	2	81 x 183 x 203	0.6	1.2	6	9
Cabling	N/A	N/A	N/A	20	0	0
Attitude Control						
Reaction Wheel	4	316 dia. x 159	7	28	7.5	29
Star Tracker	2	152.4 dia. x 203.2	5.5	11	30	30
Coarse Sun Detector	4	28 dia. x 12	0.011	0.044	0	0
Fine Sun Sensor	1	100 x 100 x 40	0.29	0.29	0	0
Fine Sun Sensor Electronics	1	90 x 140 x 60	0.64	0.64	2.8	2.8
Inertial Measurment Unit	2	203.2 dia. x 152.4	0.75	1.5	24	24
Thermal Control						
Heat Pipes	6	5 dia. x 300	0.18	1.08	0	0
Insulation and Coatings	N/A	N/A	N/A	1.53	0	0
Thermostatic Heater	8	N/A	0.05	0.4	16	16
Temperature Gauge	60	N/A	0.01	0.6	0	1
Communications						
High-gain Antenna (Parabolic)	1	1500 dia. x 700	8	8	0	0
Himispheric Antenna	2	80 dia. x 40	0.25	0.5	0	0
TX Amplifier TWTA	2	300 x 100 x 80	2.3	4.6	20	50
Transponder	2	113 x 174 x 134	3.1	6.2	8	13
Switches & Diplexer	1	100 x 100 x 20	0.5	0.5	0	0
Wiring	1	N/A	3	3	0	0
Structures and Mechanisms						
Launch-vehicle Adapter	1	1270 dia. x 762	13.5	13.5	0	0
Spacecraft Structure*	1	1200 x 1200 x 1750	106.2	106.2	0	0
Two-axis Gimbal for Antenna	1	N/A	5.44	5.44	0	10
Antenna Boom Deployment Mechanism**	1	N/A	0.5	0.5	0	0
Solar Array Deployment Mechanism**	3	N/A	1.3	3.9	0	0
Propulsion						
Thrusters	12	32 dia. x 146	0.33	3.96	0	8.5
Main Engine	1	380 dia. x 700	3.6	3.6	0	12
Propulsion Fuel	1	N/A	553	553	0	0
MMH Storage Tank	1	460 dia. sphere	2	2	0	0
N_2O_4 Storage Tank	1	366 dia. Sphere	2	2	0	0
Mass and Power without Growth				860.1	127.8	254.9
25% Mass-growth Allowance#				215.0		
20% Power-growth Allowance#					25.6	51.0
Totals with Growth				**1075.1**	**153.4**	**305.9**

*"Spacecraft structure" includes the body structure, equipment-support panels, payload mounting structure, appendage booms and restraint structure, solar array substrate, and miscellaneous tertiary structures.

**The deployment mechanisms listed above operate one time only, so they won't contribute to the spacecraft's power demands once operational capability has been achieved.

#The growth-allowance percentages are historical averages.

Fig 1.7 Mass and Power Statement

1.6 Summary of Key Trade Studies

1.6.1 Mission Design

The key trade study for mission design was determining launch dates and available C3 from the Delta launch vehicle. Additionally, aerobraking was evaluated as a means to lessen the ΔV and C3 requirements. Because of the short mission lifetime and small payload, aerobraking was deemed unnecessary.

1.6.2 Structures

Mass constraints and material selection make up the key trade studies in the structures subsystem. Aluminum, aluminum honeycomb, and composite honeycomb were selected because of availability and flight proven performance.

1.6.3 Propulsion

The type of propellant system employed was the key trade study in propulsion. Because of the necessity to have a main engine capable of a large ΔV and also a thruster system for slews and momentum dumping, a bi-propellant system was chosen.

1.6.4 Attitude Control System

Because of high pointing requirements, a 3 axis stabilized system was chosen. As well, because of recent Mars mission failures, a redundant reaction wheel, star tracker, and IMU were evaluated and added.

1.6.5 Power

Key trade studies were performed on battery cell types and solar cell types. The batteries were sized from mission constraints developed and then evaluated on a weight to power basis. The solar cells were evaluated based on efficiency.

1.6.6 Thermal

A passive system was traded for an active system due to power constraints and the ability to use thermally emissive paints and MLI.

1.6.7 Command and Data Handling

Trades were performed on the type of computer architecture to use. A VME style architecture was chosen because of flight heritage, ease of system upgrade, and readily available commercial products.

1.6.8 Telecommunications

A major trade established the use of X-band frequencies for both uplink and downlink. Additionally a travelling wave tube amplifier was traded for a solid state device due to power constraints.

2.0 Mission Design

2.1 Summary of Requirements

The Request for Proposal (RFP) for the Solar Irradiance Monitoring Mission at Mars outlines several requirements specific to the mission design portion of the project. This particular mission involves inter-planetary travel from Earth to Mars as well as final orbit insertion at Mars. The following list of requirements were specified in the RFP:

- Launch Date no later than September 30, 2004
- Launch within 35 months from start of design/development phase (phase C/D)
- Launch will be provided by NASA for medium class (Delta II 7925) or smaller expendable launch vehicle of U.S. manufacture
- Launch Operations must include:
  - Launch from Earth
  - Put spacecraft in orbit around Mars
  - Choose launch window
  - Schedule 3 targeting delta V adjustments enroute to Mars
- Orbit Characteristics
  - Perform solar observations during 15 minutes of each Mars orbit
  - Observe 1 target star for 20 minutes during eclipse
- Mission Life - 2 years

Additional requirements were placed on the mission design by other subsystems of the satellite including telecommunications, thermal, and power.

- **Telecommunications**: needed a certain orbit period around Mars in order to have enough time to transmit science and engineering data back to Earth
- **Thermal**: wanted the shortest time possible in eclipse
- **Power**: needed sufficient time in the sun to collect solar energy with the solar arrays

2.2 Earth Orbit - Phase I

The initial phase of the mission design sequence is launch from Earth and establishment of a parking orbit around Earth. After reviewing the location of both Earth and Mars with respect to each other and the Sun, a launch date of April 1, 2003 was chosen (Fig 2.1). The blue orbital path is representative of Mars, the red orbital path is representative of Earth, the black line is the trajectory of the spacecraft, the pink point is the point that the spacecraft intercepts Mars, and the sun is located in the middle of the figure.

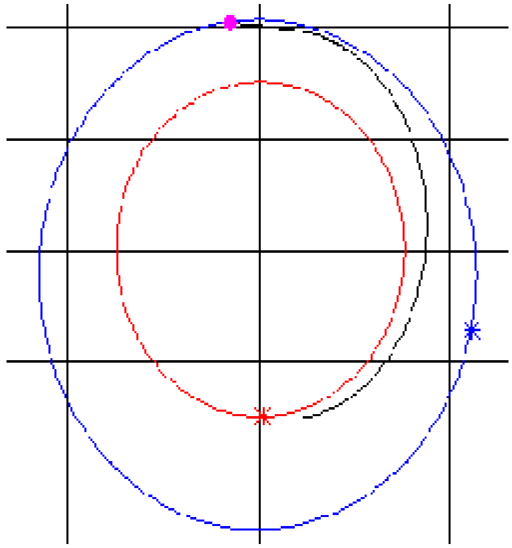


Figure 2.1**: Earth and Mars Positions on April 1, 2003 and Trajectory of Spacecraft**

2.2.1 Launch Vehicle Characteristics

The Delta II- 7925 expendable launch vehicle posseses several characteristics that made it an appealing option for this mission. Most importantly, the Delta II-7925 is a 3-stage rocket that utilizes its last stage for initiation of an inter-planetary trajectory. Below is a list of other important Delta II-7925 characteristics:

- Eastern Range launch location
- 2.9-m Fairing diameter
- Maximum Perigee Velocity = 11.5 km/sec (based on spacecraft mass of 1000-kg)
- Launch Energy (C3) = 14 km^2/sec^2 (based on spacecraft mass of 1000-kg)
- Spacecraft mass capabilities up to 1400-kg
- Variable orbital parameter capabilities including: inclination, azimuth angle, radius of apogee, and radius of perigee

2.2.2 Earth Orbital Parameters

Table 2.1 shows the essential orbital elements that define the Earth orbit of the satellite. All equations used for calculating the values in Table 2.1 are shown in Appendix: A2.

Table 2.1**: Earth Orbital Parameters**

Orbital Elements	Value	Units
Semi-major axis (a)	800	km
Inclination (i)	0	deg
Eccentricity (e)	0	NA
Period (P)	101.1	minutes
Radius of Perigee (r_p)	800	km
Radius of Apogee (r_a)	800	km
Right Ascension of Ascending Node (Ω)	86.5	deg
Argument of Perigee (ω)	0	deg
True Anomaly (υ)	159.9	deg
Velocity	7450.44	m/sec
C3 Needed	11	km^2/sec^2

During the 10 days spent in Earth orbit, the spacecraft will go thought a series of functional tests and the solar panels and antennas will also be deployed. Testing is being done in Earth orbit to ensure that the satellite is completely operational before progressing to the next mission design phase, and in order for the satellite to be completely functional the solar panels and antennas must be deployed. After the spacecraft design team is satisfied with the functionality of the satellite, a burn will be made using the 3rd stage of the Delta II-7925 to begin the transfer orbit sequence.

2.3 Transfer Orbit - Phase II

The second phase of the mission design sequence involves projecting the satellite from Earth orbit to Mars orbit and performing the necessary ΔV maneuvers to compensate for the difference in flight path angles of Earth and Mars. This phase utilizes the patched conic approach to trajectory calculations outlined in the Spacecraft Mission Design Book, 2nd Edition. A detailed discussion of the patch conic approach along with equations and calculations can be found in Appendix: A2

Figure 2.2 shows the position of the Earth and Mars when the spacecraft intercepts Mars, and the figure has the same configuration as Figure 2.1.

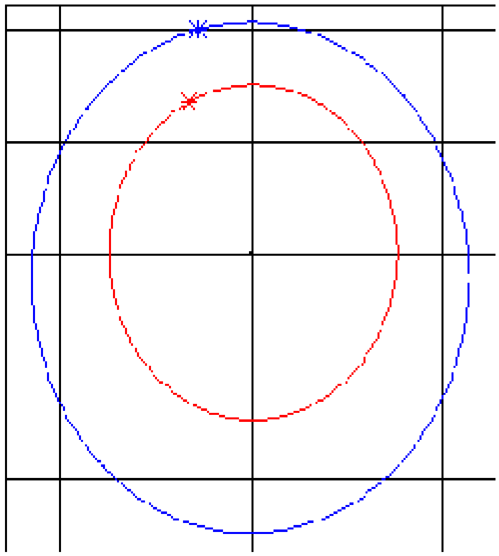


Figure 2.2: Earth/Mars position at spacecraft interception October 1, 2003

Table 2.2 below shows the calculated values for all of the hyperbolic orbital parameters discussed in Appendix: A2.

Table 2.2**: Trajectory Orbital Parameters**

Orbital Element	Value	Units
Semi-major axis (a)	178,439,043	km
Radius of perigee (r_p)	144,744,684	km
Eccentricity (e)	0.188	NA
Inclination (i)	2.6	deg
V_{HE}	3.31	km/sec
Period (P)	182.9	days
$V_{infinity}$	8.98	km/sec
ΔV at Mars	3.375	km/sec

Three ΔV maneuvers will be performed enroute to Mars to correct for the following:

- Flight path angle (relative to the sun) difference between Earth and Mars which is 10.07°
- β angle changes of 21.4° and 18.4° (please refer to Appendix: A2 for calculation of β angles)
- Correction for small orbital disturbances during the trajectory flight

ΔV values of $10^{m}/_{sec}$, $30^{m}/_{sec}$, and $50^{m}/_{sec}$ will be used to make the above mentioned changes in angle. It will be determined during the trajectory flight the best time to make these corrections before arriving at Mars.

2.4 Mars Orbit

The orbit around Mars will begin on October 1, 2003 and will propagate until October 1, 2005. Primary drivers of the Martian orbital parameters were the science payloads, the power sub-system, the thermal subsystem, and the telecommunications sub-system including:

- Science Payloads
  - 15 minutes of solar observation each orbit
  - 20 minutes of star tracking each orbit during eclipse
- TeleComm
  - 20 minutes to transmit data back to Earth
- Thermal
  - Minimal time in eclipse = 40 minutes/orbit
- Power
  - Maximum time in the sun = 92.7 minutes/orbit

Meeting the above requirements resulted in an orbit with the characteristics outlined in Table 2.3. All of the equations and constants used to calculate the values in Table 2.3 are in Appendix: A2.

Table 2.3**: Mars Orbital Parameters**

Orbital Elements	Value	Units
Semi-major axis (a)	700	km
Inclination (i)	5	deg
Eccentricity (e)	0.43	NA
Period (P)	132.7	minutes
Time in Eclipse	40	minutes
Radius of Perigee (r_p)	400	km
Radius of Apogee (r_a)	1000	km
Right Ascension of Ascending Node (Ω)	49.58	deg
Argument of Perigee (ω)	175	deg
True Anomaly (υ)	323.5	deg
Velocity at Perigee V_p	12.36	km/sec
Velocity at Apogee V_a	4.95	km/sec

The initial orientation of the Earth and Mars relative to the sun is shown in Figure 2.3. During the two years that will be spent in the Martian orbit, Earth and Mars will move into positions that are located on opposite sides of the sun. This occurs in August of the year 2004 and lasts for approximately 1 month. At both the beginning and end of the mission, Earth and Mars will be at their closest point to each other, please refer to Figure 2.2 and Figure 2.4 for clarification.

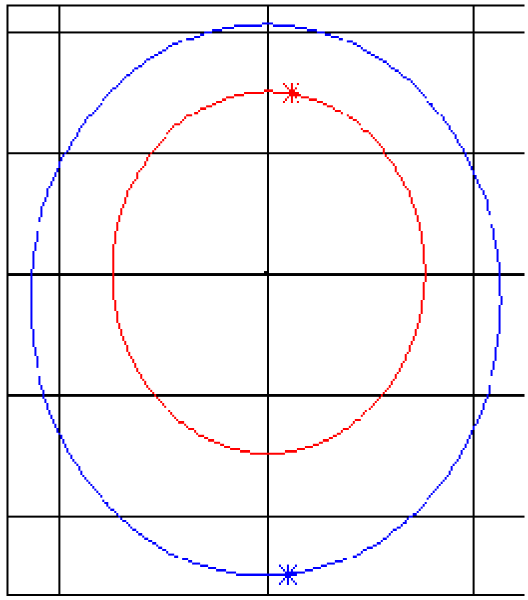


Figure 2.3**: Earth/Mars location on August 22, 2004**

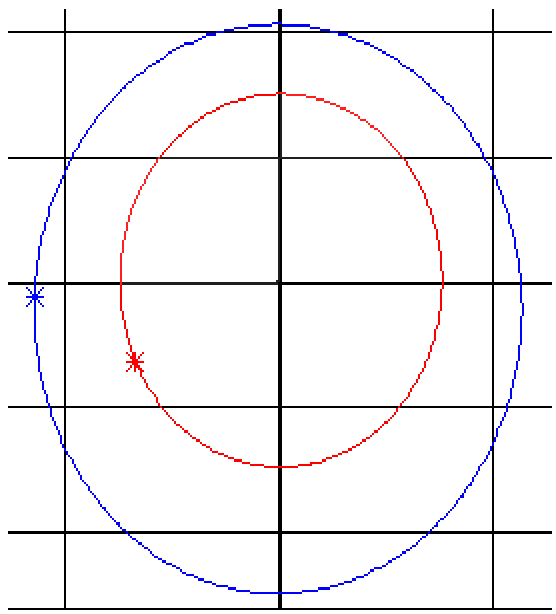


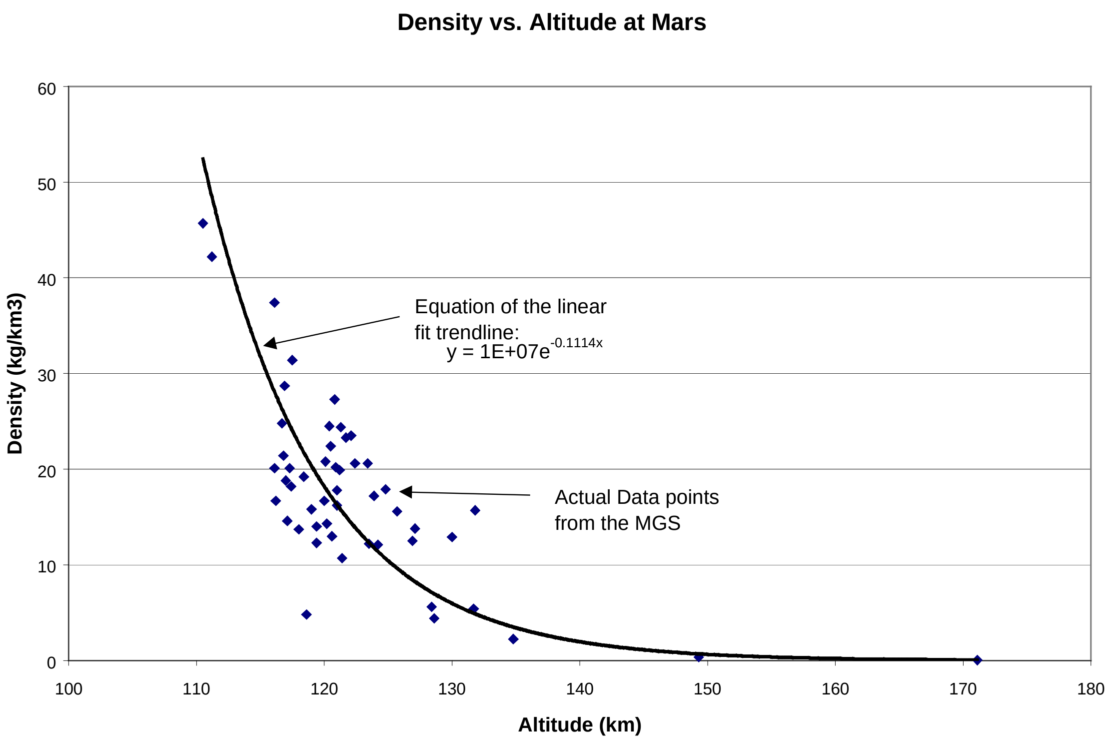
Figure 2.4**: Earth/Mars position at the end of mission December 25, 2005**

2.4 Orbit Degradation

There are several space environmental factors that will affect this Martian orbit during its two-year duration. These perturbations are induced primarily by aerodynamic drag due to the Martian atmosphere. Another source of disturbance to the spacecraft is solar radiation. Aerodynamic drag will affect the spacecraft at the radius of perigee much more than it will at the radius of apogee. The aerodynamic drag acts as a negative acceleration on the spacecraft causing the velocity to decrease. As the velocity decreases, so does the orbit altitude resulting in an orbit closer to the surface of Mars.

2.4.1 Aerodynamic Drag

Data collected from the Mars Global Surveyor was used to determine the atmospheric density as a function of altitude for Mars. This data was then used to calculate the aerodynamic drag using the equation defined in Appendix: A2. The resulting aerodynamic drag as a function of height is shown in Graph 2.1 below. Translating this to a decreased velocity and height resulted in the final orbit parameters for Mars defined in Table 2.5. The amount of decrease in both velocity and altitude (as a function of initial orbit altitude) due to aerodynamic drag is shown in Graph 2.2.



Graph 2.1**: Atmospheric Density vs. Altitude at Mars**

Graph 2.1 reveals the exponential nature of the atmospheric density near Mars. The density decreases exponentially with an increase in altitude above Mars. An exponential trendline was fit to the data, so that the equation for the line could be derived.

$$y = 1 * 10^{7} e^{-0.1114 * x} \qquad (1)$$

This equation was then used to calculate the density of Mars at altitudes ranging from 400-km to 1000-km.

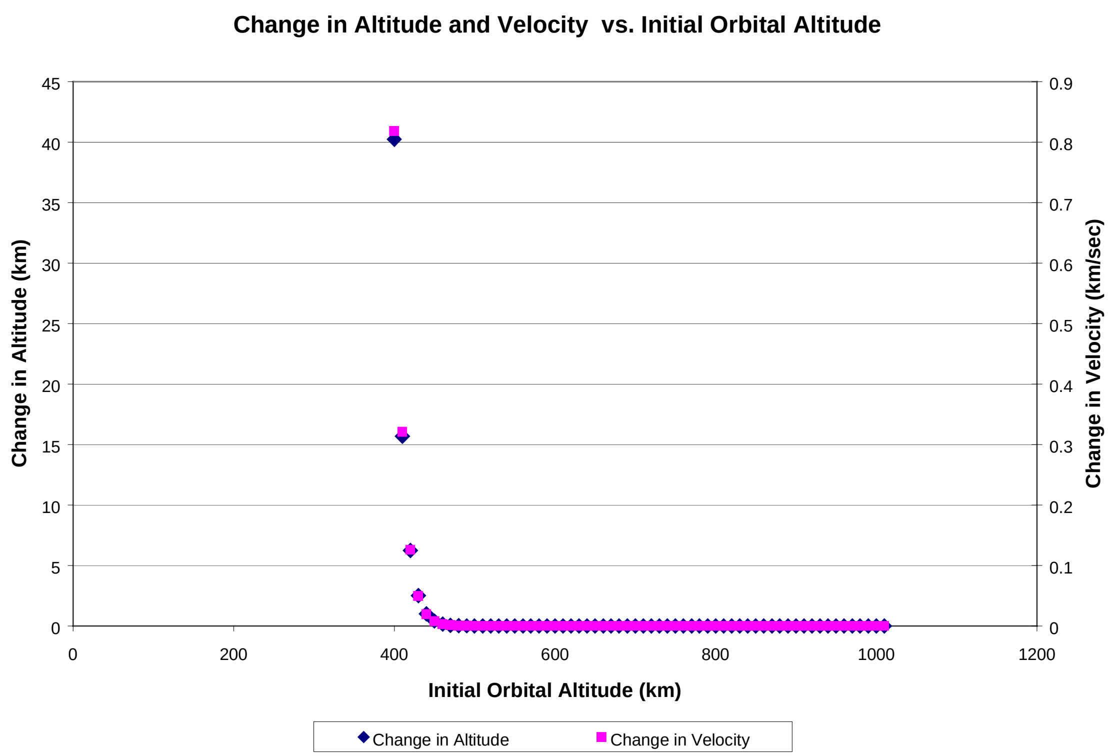


Figure 2.2**: Effects of Aerodynamic Drag on Altitude and Velocity**

Graph 2.2 shows the direct relationship between the change in altitude and velocity as a function of initial orbit altitude. It is observed that atmospheric drag has the greatest effect on the perigee radius, reducing it by 40-km during the two-year mission. There is virtually no effect on either the altitude or velocity at initial orbital heights greater than 450-km. This change in perigee altitude will not have a devastating effect on the other orbital parameters, which are shown in Table 2.5.

Table 2.5**: Change in Orbital Parameters due to Environmental Affects**

Orbital Parameter	Amount of Change from Initial	New Value	Units
Semi-major axis (a)	20	680	km
Eccentricity (e)	0.04	0.47	NA
Period (P)	0.973	131.73	minutes
Radius of Perigee (r_p)	40	360	km
Velocity at Perigee (V_p)	0.82	11.54	km/sec

2.4.2 Other Environmental Disturbances

The other sources of environmental disturbances that the spacecraft will incur during its two-year mission are gravity gradient torques and solar perturbations. Both of these parameters will be corrected for with the ADCS system and will not greatly affect the orbital parameters. The torque due to solar radiation pressure was calculated to be negative 8.57143E-10 m^2/kg, which does not produce a significant amount of disturbance on the spacecraft. All of the torques created by the gravity gradient of Mars are shown in Appendix: A5.

3.0 Structural Subsystem

3.1 Summary of Requirements

3.1.1 Functional Requirements

- Mechanical and electrical accommodation for SOLSTICE payload and associated electronics.
- Mounting attachments and support for subsystems.
  - Clear line of sight for payload optics and communication antennae.
  - Adequate sun-pointed surface area for solar arrays and thermal control panels.
  - Structural support and proper isolation for propulsion and ADC systems.
  - Insulate electronics, fuels, payload, batteries, etc. for optimal thermal environment.
- Easy integration between subsystems (i.e. wiring).
- Minimize vibration.
- Mission life of 2 years plus transfer time.
- Fail-safe design: no single failure can cause the loss of more than 50% of the scientific data, or loss of the minimum engineering data, or loss of command capability.
- Minimize outgassing of materials.
- Low cost.

3.1.2 Operational Requirements

- Payload
  - 2π steradian clear field of view.
  - Keep optics field-of-view aligned within ±1.5° of calibrated orientation.
- Communications
  - ±45° field-of-view, swept 360° for high-gain parabolic antenna.
  - ±180° field-of-view, swept 360° with few obstacles for hemispherical antennae.
- Withstand temperature extremes from worst-case hot to worst-case cold.

3.1.3 Constraints

- Required by launch vehicle: Delta 7925 with standard 2.9 m diameter fairing
  - Maximum spacecraft mass: 1500 kg
  - Maximum diameter: 2482 mm
  - Static envelope: 58 mm from maximum diameter
  - Stiffness to provide fundamental frequencies above 35 Hz in the thrust axis and 15 Hz in the lateral axis.
  - Center of gravity within 1.3 mm of the centerline.
  - Principal axis misalignment of less than 0.25°.
- Required by Propulsion Subsystem
  - Maximum spacecraft mass: 1050 kg

3.2 Spacecraft General Arrangement

Figures 3.1 through 3.3 illustrate the basic layout of Sunstroke in its stowed and deployed configurations. Figure 3.3 is a layout of the internal packaging arrangement for the components.

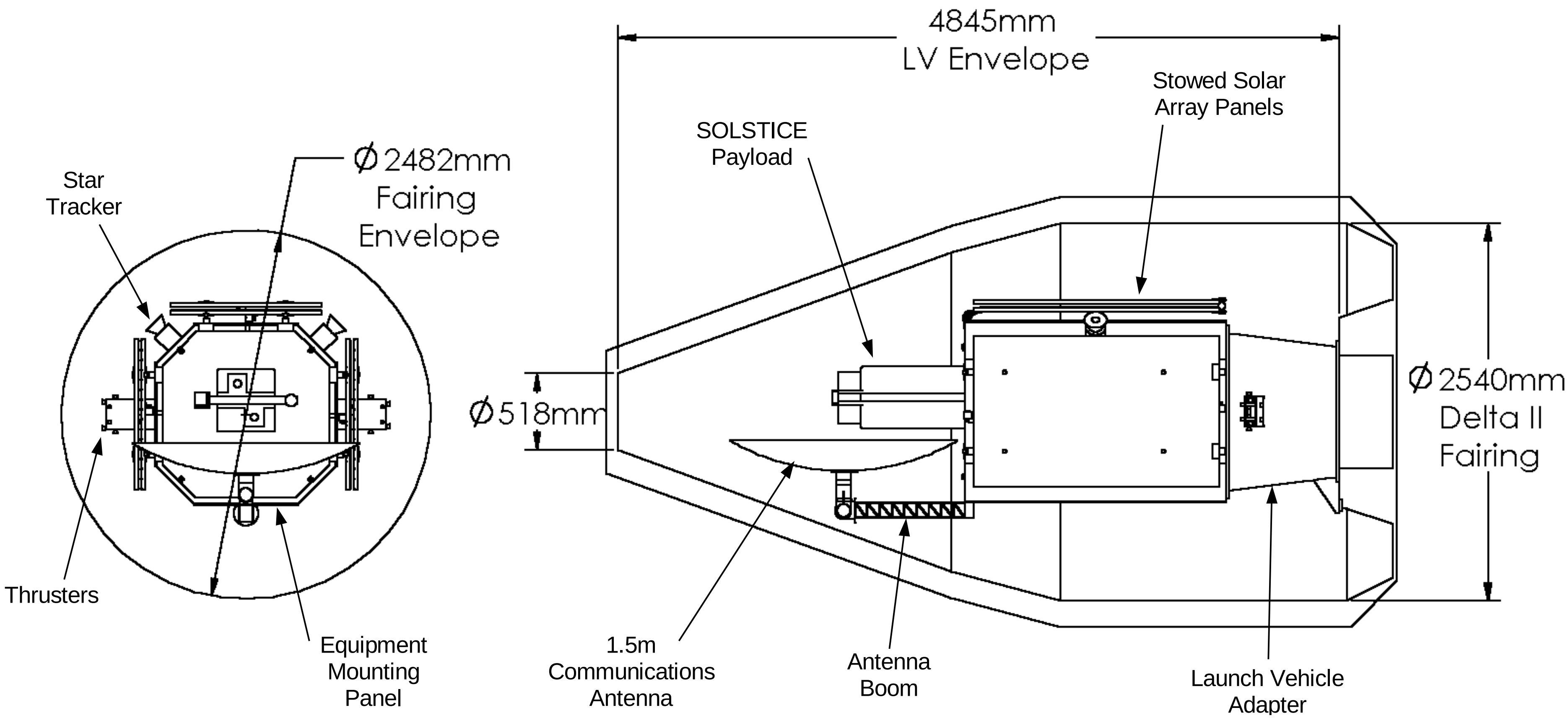


Figure 3.1: Stowed Configuration of Sunstroke

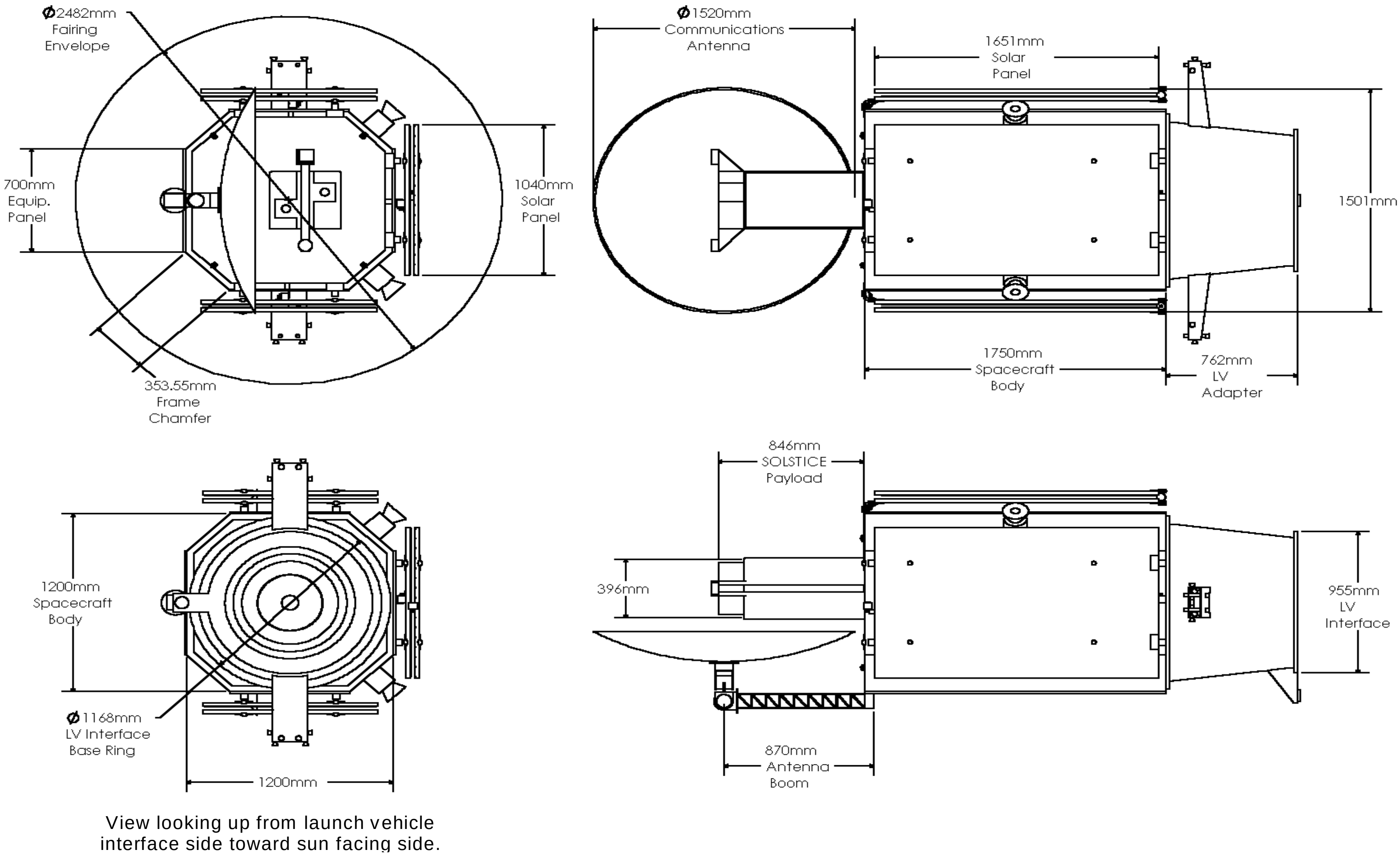


Figure 3.2: Stowed Configuration of Sunstroke – Additional Views

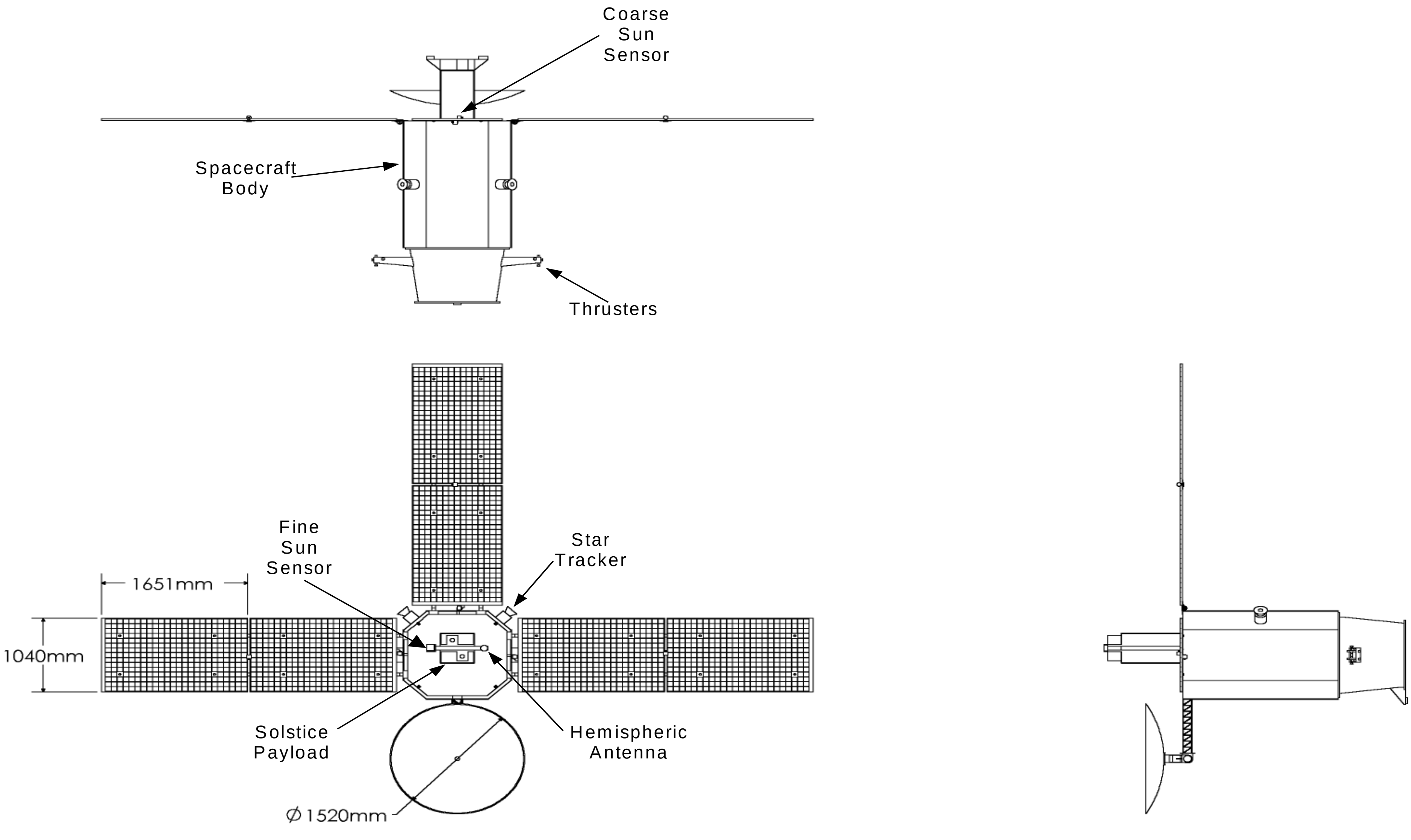
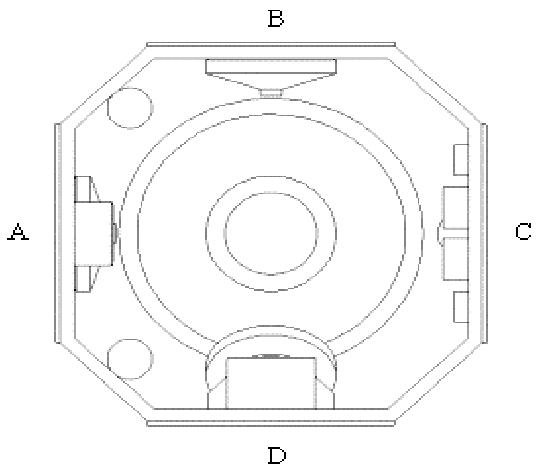
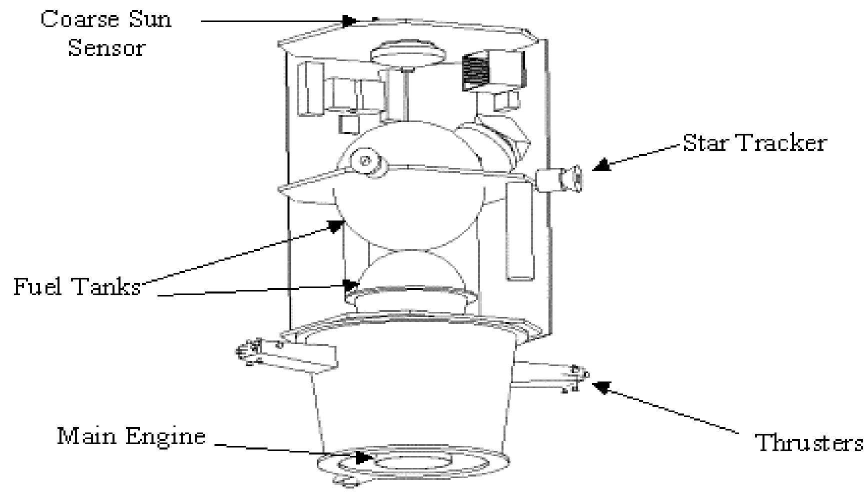


Figure 3.3: Deployed Configuration of Sunstroke



Orientation View Looking From Sun-Facing Side Toward Launch Vehicle Interface



Panels A and B hidden to show interior packaging of propulsion system and decks..

Equipment Identification
1 - Inertial Measurment Unit
2 - Reaction Wheel
3 - TWTA Amplifier
4 - Transponder
5 - Diplexer
6 - Battery
7 - Electronics Module
8 - Fine Sun Sensor Electronics

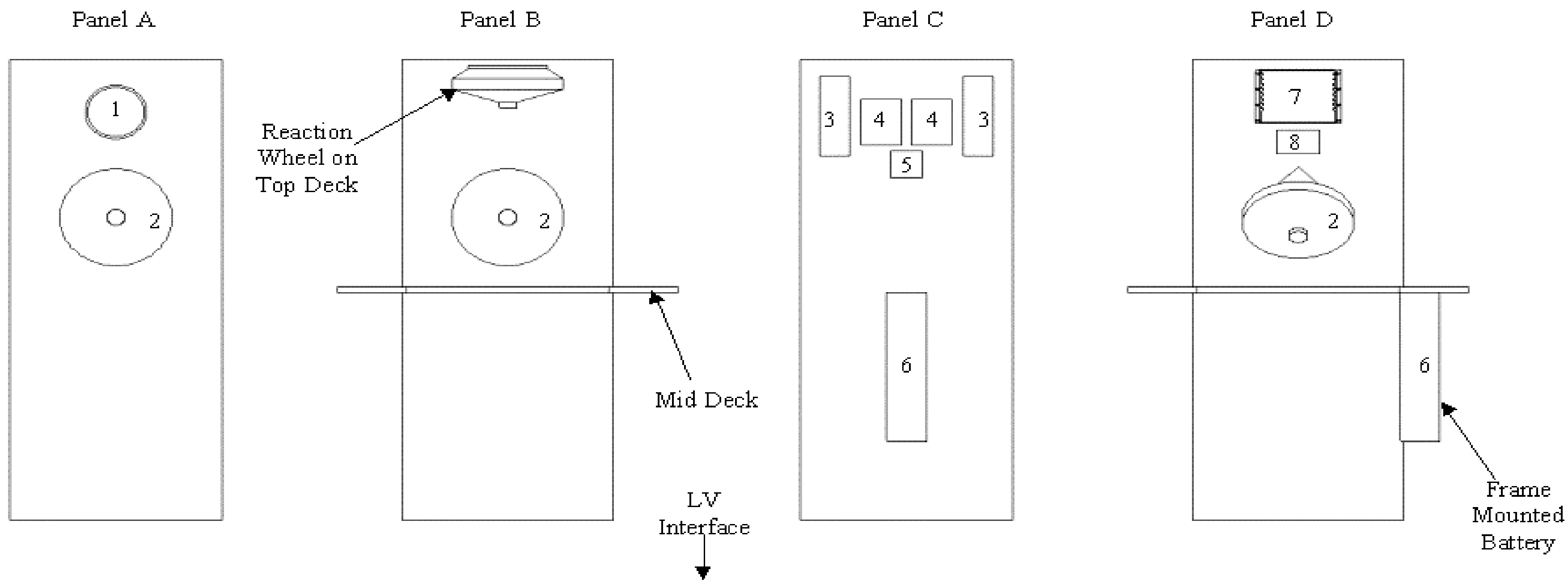


Figure 3.4: Layout of Sunstroke's Internal Packaging Arrangement

3.2.1 Spacecraft Bus

Because bus volume is limited, a closed architecture will be used to maximize packaging efficiency. A closed architecture will also provide greater bending stiffness due to its wider cross section.

Sunstroke will have a skin-frame type structure. The frame will have a nearly octagonal cross-section, as seen in Figure 3.5, and will be constructed with extruded aluminum tubes. This modified square was chosen as the optimal shape to accommodate the solar arrays and communications antenna, evenly distribute loads, integrate with the launch vehicle, and provide flat mounting surfaces for components.

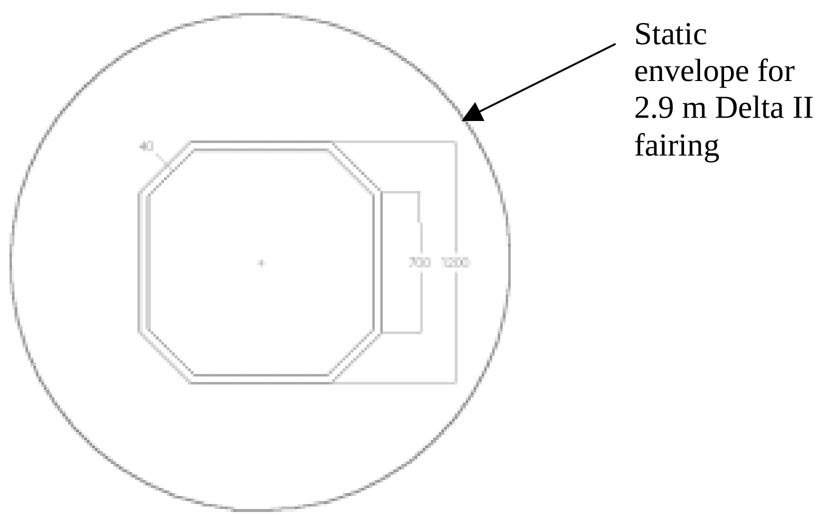


Figure 3.5: Modified square cross-section used for Sunstroke.

The skin will be constructed with 12.7-mm thick aluminum honeycomb sandwich panels. These panels will help carry compressive loads imposed on the spacecraft and will serve as mounting surfaces for the majority of the internal components. Although all of the surface area will not be usable due to interference, panel mounting provides better thermal control because the panels will serve as radiators. A single deck will also be included in the middle of the body length for structural support and overall rigidity of the spacecraft. The primary fuel tank will be mounted to this deck. Mounting diagrams for each of the panels are shown in Figure 3.4.

There will also be several externally mounted components. Three symmetrical solar arrays will be balanced with communications antenna on the sun facing side of Solstice to provide adequate solar array area, proper field of regard for the antenna, and to minimize environmental disturbances or torque. Two star trackers will be mounted on two different frame panels with a 90° offset. Two hemispherical antennas will be mounted opposite one another on the fore and aft decks for full 360° coverage. A fine sun sensor and four coarse sun sensors will also be mounted on the sun-facing deck.

The launch vehicle interface will remain attached to the main bus. It will serve as housing for the main engine and part of the small propulsion tank. It will also act as a mounting structure for all twelve thrusters.

3.2.2 Solar Arrays

Fixed solar arrays will be used because the payload requires the spacecraft to be pointed directly at the sun when not in eclipse, and the pointing requirements for the payload are more than adequate to keep the solar arrays aligned. This configuration requires 10.3 m^2 of solar array area.

Solar array area requirements necessitate a common hinged-stacked solar array arrangement that is one panel wide and 2 panels deep. Thus, there are a total of six panels, each of which must have an area of approximately 1.72 m^2. The final dimensions of each solar panel are 1.04 m x 1.65 m for a total area of 10.3 m^2. The substrate for the solar arrays is a 12.7-mm thick aluminum honeycomb sandwich panel with composite facesheets.

3.3 Mechanical System Description

A primary goal for Sunstroke is to minimize the number and complexity of on-board mechanisms. The required mechanisms include a launch vehicle separation interface, solar array release mechanisms, hinges, and dampers, a boom deployment mechanism, and an unlocking gimbal mechanism.

Launch vehicle separation will be provided by the separation ring supplied with the selection of the Delta II launch vehicle.

Launch restraints for the solar arrays will consist of four hard points or through-bolts for each array. These are simple devices with no moving parts that will maintain proper stiffness to avoid cracking the solar panels during launch. To release the arrays, a flight-proven “off-the-shelf” type explosive bolt cutter will be used. A capture plate will be included on the outside of the spacecraft to catch the debris from the severed bolts.

The hinge mechanism for the solar arrays will be a simple carpenter’s tape type hinge. This is a simple device with few total parts. There are no sliding parts, so there is no friction in the system and there will be no looseness when deployed. Thus, there will be no linear motion and the panels will be able to withstand higher vibration frequencies. For deployment, a damper mechanism will be included to ensure that the panels deploy slowly so that they do not cause the spacecraft to spin out of control. This will be easier for the attitude control system to counteract. Both the hinges and the damper can be bought “off-the-shelf” with a flight-tested design.

Release and deployment mechanisms of the communications antenna and boom will use very similar components. For pointing, a two-axis gimbal platform and a deployable boom will be employed. Again, designs will be used that have been previously qualified for space flight.

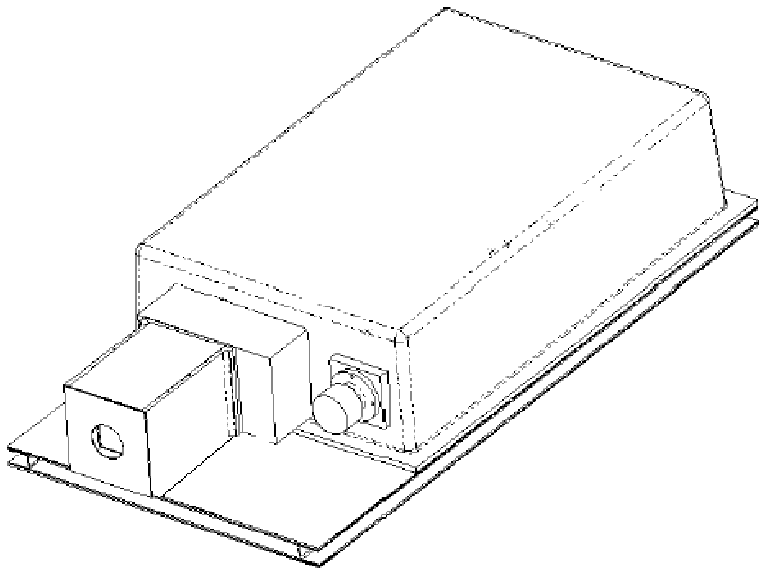
3.4 Mechanical Equipment List

Table 3.1: Equipment List for Structural Subsystem

Component	Quantity	Size (mm)	Individual Mass (kg)	Total Mass (kg)
Primary Structure				
Frame	1	1200 x 1200 x 1750	33.5	33.5
Side Panels	4	700 x 700 x 1750	3.6	14.4
Top Deck	1	1120 x 1120 x 25.4	6.8	6.8
Mid Deck	1	1120 x 1120 x 25.4	4.2	4.2
Bottom Deck	1	1120 x 1120 x 25.4	2.0	2.0
Launch Vehicle Interface	1	1270 dia. x 762	13.5	13.5
Solar Panel Substrate	6	1040 x 1651	5.0	30.0
Antenna Boom	1	101.6 x 101.6 x 793	2.6	2.6
Two-axis Gimbal Platform	1	N/A	5.44	5.44
Secondary Structure				
Mounitng Plate for Solstice	1	846 x 367 x 50	2	2
Solar Array Hinges	6	N/A	0.2	1.2
Solar Array Damper	6	N/A	0.45	2.7
Antenna Deployment Mechanism	1	N/A	0.5	0.5
Bolt Cutters	14	N/A	0.05	0.7
Mounting Brackets	N/A	Misc.	N/A	10
Total Mass of Structural Subsystem				**129.5**

3.5 Payload Accommodation Issues

To better isolate it from pyrotechnic shock and structural distortions during launch, the SOLSTICE payload will be mounted at the forward end of the spacecraft instead of the aft end (launch-vehicle interface). This payload consists of two UV spectrometers like the one in Figure 3.6 that both must be sun-pointed. They will be mounted externally on the sun facing deck with a specially designed mounting bracket. This mounting configuration easily satisfies the ±1.5° swept 360° field of view



requirements for the SOLSTICE experiment.

Figure 3.6: Solstice UV Spectrometer

3.6 Mass and Power Statement

The structural subsystem for Sunstroke will have a mass of 129.54 kg. This includes the launch vehicle interface, solar panel substrate, solar array release and deployment mechanisms, antenna boom, antenna release and deployment mechanisms, gimbal platform, frame, skin panels, mounting brackets for components, and miscellaneous tertiary components. Power requirements for the structural subsystem, excluding the gimbal platform, will be negligible once the solar arrays and antenna have been deployed. The gimbal platform will have a maximum power draw of 10 watts.

4.0 Propulsion Subsystem

4.1 Summary of Requirements

Mission Phase Requirements

There is 3 main phases for the propulsion subsystem.

- cruise phase
- orbit insertion
- station keeping

Initially, the spacecraft will be inserted into a parking orbit around the Earth by a Delta II-7925 launch vehicle. This rocket will also supply a third stage that will be used to put the spacecraft into the proper Earth-Mars transfer orbit (cruise phase).

Cruise Phase

Once in the cruise phase of the mission, the spacecraft must be able to perform up to three trajectory-correction maneuvers (TCMs). These maneuvers are necessary to ensure the spacecraft's proper orientation with respect to Mars' equator, to provide course corrections, and to deal with other factors that may come into play during the mission. The spacecraft is designed to perform TCMs of 50, 30, and 10 m/s. Actual values of the TCMs will be determined during the mission. The main engine will perform these maneuvers unless a plane change is required, in which case the attitude control thrusters will be used to change the orientation of the spacecraft and the main engine will provide the necessary ΔV.

Orbit Insertion

When the spacecraft reaches Mars, it must be decelerated in order to be captured by Mars' gravity (orbit insertion). This will be accomplished by a single burn of the main engine. The ΔV corresponding to this maneuver is defined by the mission design to be 3375 m/s. This burn will provide the necessary ΔV for the final Mars orbit as well.

Station Keeping

While in the final Mars orbit, there must be enough propellant remaining for orbit corrections (OCMs) and station keeping. Actual values of OCMs will be determined during the mission. Additionally, the spacecraft must perform one minute worth of momentum dumping each orbit for attitude control.

ΔV Requirements

Table 4.1 lists all known ΔV requirements for each mission phase. The TCMs are dictated by the mission RFP, the orbit insertion is determined from the mission design, and station keeping maneuvers are defined by attitude determination and control. The main engine performs all ΔVs, with the exception of momentum dumping and orbit corrections. Because maneuvers using thrusters will either be pulsed (for station keeping) or have short durations (for TCMs), gravity losses are neglected. Likewise, the main engine also neglects gravity losses because of the large thrust it provides.

Table 4.1 - Mission ΔV requirements

Maneuver	V (m/s)
Cruise Phase	
TCM 1	50
TCM 2	30
TCM 3	10
Insertion	3375
Orbit Maintenance (over mission life)	~25

4.2 Propulsion System Selection

A trade study was performed to determine the appropriate type of propulsion system for this mission. Table 4.2 summarizes this process and Appendix A8 describes each system in greater detail. Each system was ranked relative to each other in five categories. Ties indicate an equal level of performance. For example, each system that uses monopropellant thrusters has the same rank, with the exception of the all-bipropellant system, which uses bipropellant thrusters.

Table 4.2- System Trade Study

			Thruster		*Insertion*	
	power	*complexity*	*performance*	*mass*	*performance*	*I*
ALL MONO	1	1	1	5	4	26
HYBRID	4	3	1	4	2	45
ALL BI	2	2	5	2	1	38
DUAL	3	5	1	3	1	45
ELECTRIC	5	4	1	1	5	51
IMPORTANCE	5	4	3	2	1	

Each of the categories was then rated in order of importance. For example, the propulsion subsystem requires less power compared with other subsystems, therefore power requirements were not given as high a rank of importance as orbit insertion performance (based upon specific impulse) which dictates the greatest fuel demands.

The preceding method led to an equation to evaluate each system and all categories together:

$$I \text{ (Index)} = \sum_{\text{all categories}} \text{category rank} * \text{rank of system for that category}$$

With this equation, a lower I indicates a better system. From this, it was found that the all-monopropellant was the best performing system overall. However, in the two most important categories, Mars insertion and mass requirements, this system

An additional factor in determining the proper system was which types have been used for similar missions. Specifically, the Mars Global Surveyor, which most closely resembles this mission, was used as a guide. This mission, designed similarly to the Cassini mission, uses a dual-mode propulsion system. This system provides a high specific impulse for large ΔVs while still using monopropellant attitude control thrusters. This reduces the overall mass of fuel, in addition to providing greater flexibility. This system was therefore chosen for this mission.

Trade Study

Five types of propulsion systems were considered for this mission. An all-monopropellant system would use a large monopropellant engine burning Hydrazine for insertion and small monopropellant thrusters for the other maneuvers. A “hybrid” system would use a solid rocket booster for insertion and monopropellant Hydrazine thrusters for TCMs and OCMs. An all-bipropellant system would be identical to the all-monopropellant system except that an oxidizer would be used in addition to the Hydrazine. A dual-mode system would use a bipropellant main engine and use additional Hydrazine for monopropellant thrusters. Finally, an electric propulsion system would charge the fuel and would be used for the main engine while TCMs and OCMs would use monopropellant Hydrazine.

Rankings for power are based upon requirements during usage. Complexity describes the overall system in terms of number of valves, amount of fuel lines, etc. Thruster performance and insertion performance are based upon average specific impulses for a particular system. Mass rankings are based upon the required masses for fuel, engines, lines, and valves.

4.3 System Schematic and Description

The dual-mode propulsion system is shown in Figure 4.3. This system will use positive expulsion, reversing diaphragm tanks for the fuel. These tanks have advantages in weight and efficiency.

There will be two tanks each for the oxidizer and the hydrazine. This is a result of the available sizes of fuel tanks, but also provides a balanced layout within the spacecraft keeping the center of mass close to the center of pressure. The hydrazine tanks will each have a radius of 13 in. while the oxidizer tanks will be 11 in. in radius**.** The tanks will be aluminum to reduce weight. These tanks are designed based upon ones supplied by Atlantic Research Corporation which have flown on numerous missions such as the Clementine moon mission. Appendix A8 gives a description of these tanks.

The main engine will be used for TCMs and orbit insertion. The main engine will need to provide a thrust on the order of 400-500 Newtons. This level is based upon requirements determined by mission design coupled with information from prior Mars missions. The engine on which design was based is a 490 Newton engine provided by Kaiser-Marquardt (see Appendix A4 for specs). This engine will be pressure fed and radiation-cooled. Internally, the engine is designed to keep itself cool and also contains valves to prevent premature mixing of the fuels and to ensure only one direction of flow.

While the Kaiser-Marquardt engine was used as a basis for design, it only supplies a specific impulse of 315 seconds. This value requires too much propellant, causing the spacecraft to be heavier than the launch vehicle allows. Therefore, a bipropellant engine with a specific impulse of 350 seconds is used for mass calculations.

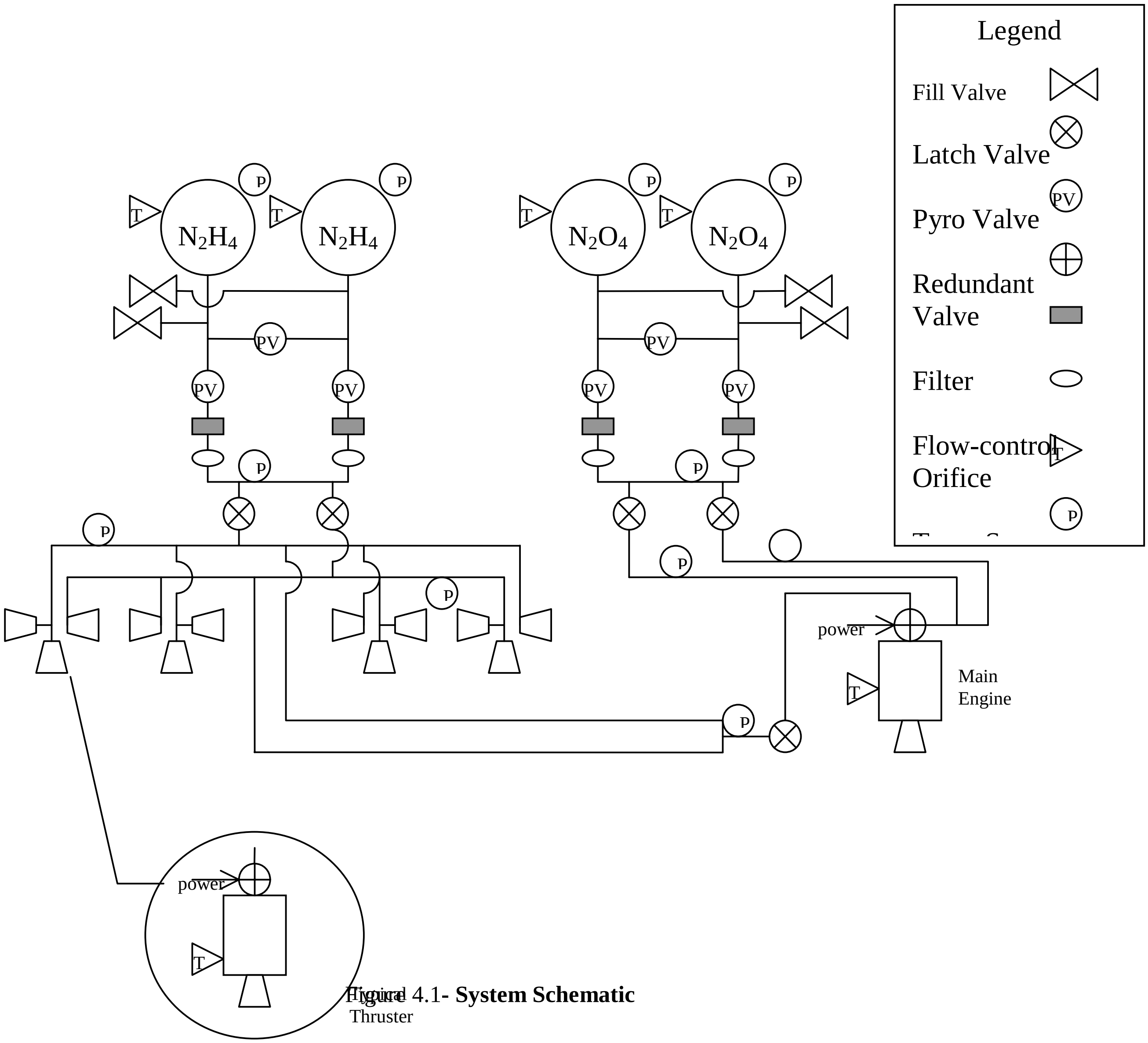


Figure 4.1- **System Schematic**

Twelve thrusters will be used to control the spacecraft through three axis in two directions each, with coupled thrusters to provide balanced moments. The thrusters must provide a nominal thrust of 1 Newton. This level was determined based upon momentum dumping procedures conducted throughout the mission.

The thrusters will be pulsed for these maneuvers, thus requiring a high pulse life (3.0E+5 from attitude determination design). The thrusters chosen for design are based upon ones supplied by Primex which provide a thrust of 1.2 Newtons with a pulse life of up to 745,000 pulses. Appendix A4 provides a description of these thrusters. It should be noted that for the TCMs, thrusters can be used to rotate the orientation of the spacecraft if necessary, while the main engine will provide the required ΔV. This is because the thrusters are not powerful enough for the TCMs dictated by the RFP. Using the thrusters for these maneuvers would require long burn times (1-2 hours) which is unreasonable.

The feed system is designed to be fault tolerant. Should any valve not operate properly, there is enough redundancy to ensure proper operation. Also, valves will be placed such that fuel can be delivered from any tank, regardless of any failures. Again, both the main engine and the thrusters have internal valves, adding more redundancy.

Filters are placed following each tank to ensure pure mixing of fuel. There will be two lines sending monopropellant to six thrusters each and two lines to the main engine for redundancy. Additionally, there will be two lines providing the oxidizer to the main engine. In case of failure to any valves in either of the thruster or main engine lines, latch valves will close that line to prevent fuel from getting to the engines. These latch valves can then be opened if those engines are needed. The thruster lines are arranged such that there will still be control in each axis, even in the case of failure to a safety latch valve. However, coupled moments would not be available in this case. The safety latch valves are open throughout the mission.

Pressure and temperature sensors will be placed throughout the system to monitor performance.

4.4 Equipment List of Pyrotechnic Functions and Devices

Because the hydrazine and oxidizer are hypergolic, they combust instantly upon contact with each other. Therefore, there are no pyrotechnic devices required for the main engine. The thrusters internally contain catalyst beds for the hydrazine. All pyrotechnics for the thrusters are therefore internal. Apart from the engines, there will be pyrotechnic valves that isolate the system during launch. These valves are included to ensure that the engines are able to receive fuel from any of the tanks, even in case of failure to any other valves. There will be six pyrotechnic valves total and can be seen in the system schematic (Figure 4.1).

4.5 Mass and Power Estimates

The propellant mass was estimated based upon initial estimates of the dry weight of the spacecraft plus all required propellant. This process was therefore iterative, since propellant mass changed depending upon total mass. A proper solution required that the final mass without propellant equaled the initial dry mass guess. The equation used for these calculations is derived from the rocket equation as:

$$M_p=M_i(1-\exp(-\Delta V/g_c \cdot I_{sp})$$

where M_p is the propellant mass, M_i is the mass prior to burn, and g_c is the gravitational constant to keep units consistent. This equation assumes negligible gravity losses, as explained earlier. The mass prior to burn changed for each phase of the mission and was found by simply subtracting the propellant mass from the previous initial mass. A code was developed to perform these calculations and is included in Appendix A4 following component spec sheets. It should be noted that the specific impulse also varied with the phase of the mission. This too, was taken into account in the included code.

The size of the fuel tanks is based upon the total required mass of the fuel. This volume does not include ullage volume because the tanks being used eliminate this loss.

The mass of each thruster is based on off-the-shelf thrusters provided by Primex. Each thruster has a mass of .33 kg. The mass of the main engine is based on an off- the-shelf engine provided by Kaiser-Marquardt, which has a total mass of 3.6 kg. The masses of additional components are estimated based upon prior missions. The feed lines are 10.5 meters of .25 in. x .020 in. stainless steel tubing, based upon the Mars Global Surveyor Mission. Masses for these components are given in Table 4.7.

Table 4.7- System Component Masses

Item	Qty.	Mass (kg)	Total Mass (kg)
Tanks	**4**	**~4**	**16**
Main Eng.	**1**	**3.6**	**3.6**
Thrusters	**12**	**0.33**	**3.96**
Hydrazine	**-**	**145.72**	**145.72**
N2O4	**-**	**408.06**	**408.06**
Valves	**28**	**0.5**	**14**
Temperature Sensors	**17**	**0.045**	**0.765**
Pressure Transducers	**10**	**0.15**	**1.5**
Lines	**-**	**0.8**	**0.8**
Fittings	**-**	**1**	**1.00**
Total Dry Mass (kg)		**41.625**	
Total Propellant Mass (kg)		**664.78**	

The propulsion subsystem has minimal power requirements. No power is required unless the system is in operation. This does not include thermal control. Power is only required to operate valves and engines as they are being used. Power requirements for the thrusters and main engine are given by the manufacturer while the power required by each individual valve is considered negligible. Power requirements are summarized in Table 4.8.

Table 4.8**- System Power Requirements**

Main Engine	~12 W
Thrusters	8.25 W each
Valves	negligible

5.0 Attitude Control System

5.1 Summary of Requirements

The spacecraft coordinate system shall be as shown in figure 5.1. The X-axis runs down the centerline of the spacecraft and is defined by the nominal centerline of the launch vehicle/spacecraft interface. The X=0 plane is defined by the vehicle/spacecraft interface plane (the coordinate system is not shown as X=0 in Figure 5.1)

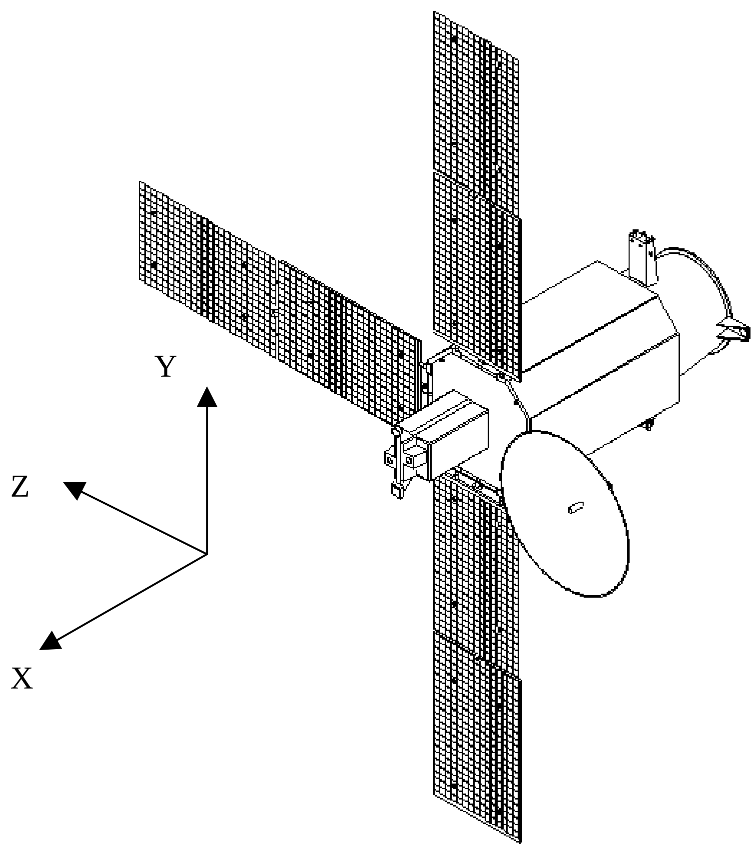


Figure 5.1. Spacecraft Coordinate System.

All ADCS requirements are listed below

- The attitude control system (ACS) shall be capable of performing all operations for the entire mission autonomously.

- The ACS shall determine the spacecraft orientation and spin rate and then despin, inertially fix and dump all momentum prior to launch vehicle separation after the 3rd stage of the launch vehicle has burned out.

- The ACS shall determine the spacecraft orientation and spin rate upon separation from the launch vehicle. It shall orient the spacecraft to point the solar arrays at the sun and orient the communications dish to point toward Earth. It shall provide attitude maintenance to hold this orientation.

- The ACS shall not allow the spacecraft to perform maneuvers that disrupt the Earth pointing orientation of the communications dish at any time during the mission. This dish orientation shall be autonomously maintained throughout the mission after the dish is deployed.

- The ACS shall dump momentum as necessary while pointing the solar arrays at the sun in the transfer ellipse. See Appendix A5 for calculations.

- The ACS shall orient the spacecraft for all delta V burns in the transfer ellipse. Momentum shall be dumped prior to each burn.

- The ACS shall orient the spacecraft to point the solar arrays at the sun after each delta V burn is completed.

- The ACS shall orient the spacecraft for the Mars orbit insertion burn. Momentum shall be dumped prior to this burn.

- The ACS shall orient the spacecraft to point the instrument and solar arrays at the sun after the Mars orbit insertion burn is completed. (The UV spectrometers will take solar observations for a minimum of 15 minutes in each orbit while in this orientation.) This inertially fixed spacecraft orientation shall be maintained by the ACS at all times while in Mars orbit with the exception of requirement 5.1.11.

- The ACS shall orient the spacecraft such that the UV spectrometers can take observations of one target star outside the ecliptic plane for a minimum of 20 minutes while in Martian eclipse in each orbit.

- The ACS shall not dump momentum during instrument observation unless it is an emergency as defined by requirement 5.1.13

- The ACS shall immediately dump momentum if at any time a reaction wheel reaches 75% peak operating speed.

- All components of the ACS shall have a designed life of 3 years on orbit (1 year to get to Mars plus 2 years of science acquisition).

- No single failures shall cause the loss of ACS command capability. For the ACS this means full redundancy.

The pointing requirements for the mission are listed below in Table 5.1

Table 5.1. **ACS Pointing Requirements**

	Instrument	Solar Arrays	Communications Dish	Engine Burns
Transfer Ellipse	N/A	+/- 5 degrees	+/- 0.25 degrees	+/- 0.5 degrees
Mars Orbit Insertion	N/A	N/A	N/A	+/- 0.5 degrees
Solar Observation in Mars Orbit	+/- 0.1 degrees knowledge with 0.05 degree stability in a 2 minute period	+/- 5 degrees	+/- 0.25 degrees	N/A
Target Star Observation in Mars Orbit	+/- 0.1 degrees knowledge with 0.05 degree stability in a 2 minute period	+/- 5 degrees	+/- 0.25 degrees	N/A

5.2 ACS Selection and Trade Study

5.2.1 Attitude Control Method Trade Study for Meeting Pointing Requirements

Table 5.2**. System Trade Study**

ACS Type	Will it meet the 0.1 deg pointing requirement?
Gravity Gradient	No
Gravity Gradient and Momentum Bias Wheel	No
Passive Magnetic	N/A for Mars
Pure Spin Stabilization	No
Dual-Spin Stabilization	No
Bias Momentum (1 wheel)	No
Zero Momentum (Thrusters only)	No
Zero Momentum (3 wheels)	Yes
Zero Momentum CMG	Yes

Three-axis stabilization is necessary to meet the 0.1degree pointing accuracy. Reaction wheels weigh and cost less than a CMG system and a CMG system is more complex to control. Therefore a zero momentum set of reaction wheels is the best approach for this application.

5.2.2 Attitude Control Method Trade Study for Meeting Spacecraft Slew Rate

The use of thrusters was considered for execution of the 90-degree slew maneuvers, which are required in each Mars orbit. The trade is between the extra mass necessary in the momentum wheels to exceed the required momentum for the slew versus the mass of propellant required for the cumulative slewing over the mission life. The conclusion of the trade was that it is more efficient to size the momentum wheels large enough to execute the large slew maneuver than to use thrusters for slewing in combination with smaller momentum wheels for disturbance torque compensation during accurate pointing of the spacecraft. See Appendix A5 for calculations.

5.2.3 ACS Redundancy

The ACS has no single point failures. The ACS is fully redundant to meet the requirements. To achieve 100% redundancy for attitude determination, two star trackers and two Inertial Measurement Units (IMUs) are included in the design. One star tracker used in conjunction with one IMU gives a complete attitude solution, which includes attitude and absolute position of the spacecraft. These components are cross-strapped together such that either of the two star trackers can be used in combination with either of the IMUs.

To achieve 100% redundancy for attitude control, four reaction wheel assemblies are included in the design, one of which is cold stand-by oriented in an isometric plane. The wheels are sized such that in the event of an orthogonal wheel failure, the redundant wheel is capable of maintaining spacecraft attitude within the instrument observation pointing requirements in the disturbance torque environment of the Mars orbit. Only large slews in specific directions, which can and will be avoided, will no longer be available to the spacecraft upon the failure of one orthogonal wheel. Thrusters used for momentum dumping will also be fully redundant (see section 4.0 of this document).

5.2.4 Other ACS Design Features

One fine sun sensor and 4 coarse sun detectors are included in the ACS design to supply points of reference for verifying that the attitude determination system is operating and performing properly.

One 2-axis gimbal will be used for pointing the communications antenna. The gimbal will keep the antenna inertially fixed independent of the spacecraft while the spacecraft performs slewing maneuvers.

5.3 Expected Disturbance Torques

5.3.1 Disturbance Torques in Transfer Ellipse en route to Mars

The only expected disturbance torque while en route to Mars is due to solar pressure. It will be necessary to dump Momentum approximately once every 12 hours while in the transfer ellipse en route to Mars. See Appendix A5 for calculations.

Table 5.3. **Expected Disturbance Torques in Transfer Eclipse**

Source	Torque (const)	Momentum/en route
Solar Pressure	1.64×10^{-4} Nm	3402 Nms

5.3.3 Disturbance Torques in Mars Orbit

Disturbance torques that will influence the spacecraft in Mars orbit are expected to be due to gravity gradient, solar pressure, and aerodynamic drag. Magnetic disturbances are negligible. Momentum will be dumped once per orbit prior to entering the Martian eclipse. See Appendix A5 for calculations.

Table 5.4. **Expected Disturbance Torques in Mars Orbit**

Source	Torque/Orbit	Momentum/Orbit
Gravity Gradient	5.718×10^{-5} Nm	0.08 Nms
Solar Pressure	8.594×10^{-5} Nm	0.51 Nms
Aerodynamic Drag	2.531×10^{-7} Nm	0.002 Nms
Magnetic	Neglect	Neglect
Totals	1.434×10^{-4} Nm	0.592 Nms

5.4 System Block Diagram

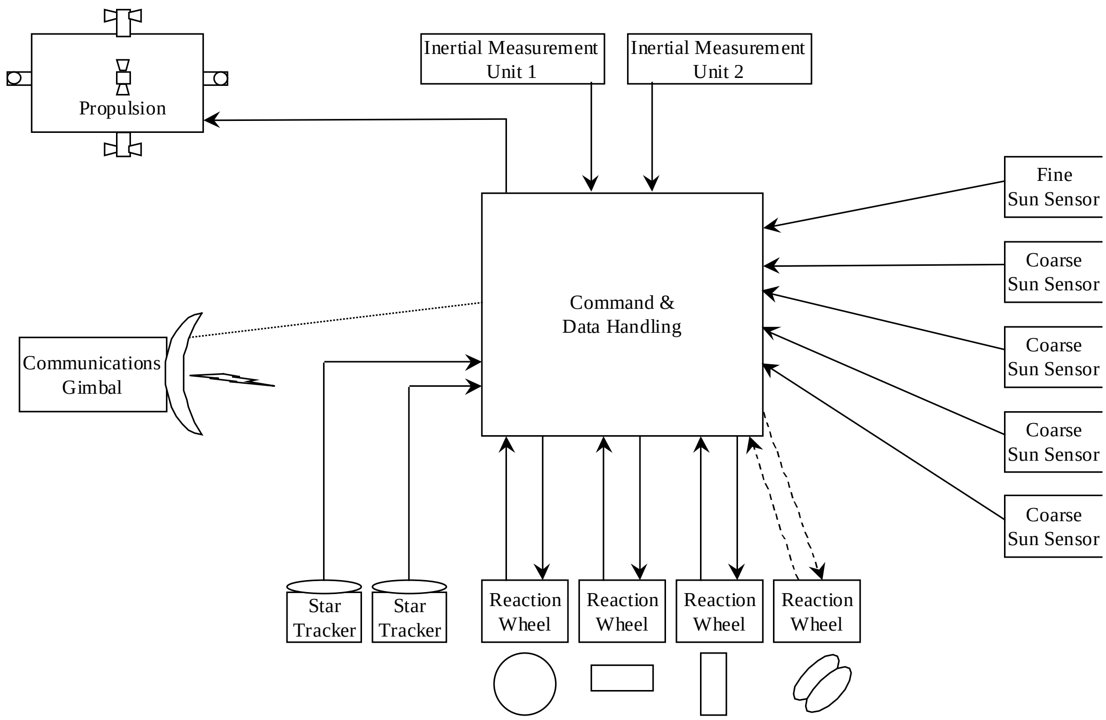


Figure 5.2. ACS System Block Diagram.

5.5 Equipment List

The ACS components are listed below. Thrusters necessary for momentum dumping are included in the propulsion section of this document (Section 4).

- 4 Zero Momentum Reaction Wheels (20 Nms storage capacity in each wheels). . See Appendix A5 for calculations.
- 2 Star Trackers
- 4 Course Sun Detectors
- 1 Fine Sun Sensor
- 2 Inertial Measurement Units

5.6 Mass and Power Estimates

Cabling, mounting brackets and hardware are not included in these estimates. . See Appendix A5 for the basis of these mass and power estimates.

Table 5.5 **ACS Mass and Power Estimates**

		Mass		Power	
Component	Qty	Mass per (kg)	total (kg)	Avg. Power (W)	Peak Pwr (W)
Reaction Wheel	4	7	28	7.5	29
Star Tracker	2	5.5	11	30	30
Course Sun Detectors	4	0.011	0.044	0	0
Fine Sun Sensor	1	0.29	0.29	0	0
Fine Sun Sensor Electronics	1	0.64	0.64	2.8	2.8
Inertial Measurement Unit	2	0.75	1.5	24	24
		Totals	**41.5**	**64.3**	**85.8**

6.0 POWER SYSTEM

6.1 Power Requirements

1) Functional Requirements
   a) The mission life is 2 years plus a transfer phase of 7 months
      i) The solar arrays and batteries must be sized to support this mission life
      ii) Solar viewing time is equal to the shortest satellite day
   b) Mission operations will be supported by factoring in EOL power generation at Mars orbital position and battery degradation, with a 20% margin built in
   c) Must accommodate the payload

2) Operational Requirements
   a) All power generated is provided for the spacecraft bus
      i) Power generation will be 28 volts (and not fall below 22 volts)
      ii) The spacecraft bus will protected from under/over voltage
   b) All power components should withstand temperature extremes
   c) Batteries should be placed so that it is accessible on launch vehicle
   d) Provide telemetry on status of the system
   e) Strict administration of satellite modes

3) Constraints
   a) Cost is a consideration, and tradeoffs will be made
   b) The solar arrays have to fold in such a way that they will fit in the launch vehicle
   c) Minimize EMI to ADCS, Payload, etc.
   d) Political circumstances dictate that failure is not an option
      i) Redundancy needs to be built in

6.1 System Block Diagram and Description

The power subsystem was designed with off-the-shelf equipment being used whenever possible. The subsystem can be divided into 4 main parts: Solar Panels, Batteries, Charge Control Board, and Power Control Board. The power system relies on three deployed GaAs solar panels. The solar panels have a total area of approximately 10 m^2. The battery section consists of two 15 A-hr NiH batteries supplied by Eagle-Picher. The batteries and solar panels are all tied to the Charge Control Board (CCB), which regulates the power from the panels, and helps to distribute the power to the batteries or the Power Control Board (PCB). The PCB distributes the power to DC-DC power converters for the critical and non-critical buses as well as spacecraft components and payload. The CCB and PCB are described in detail in the section 8.0 of this document

The four main components of the power system can be easily seen, as well as lines out to C&DH.

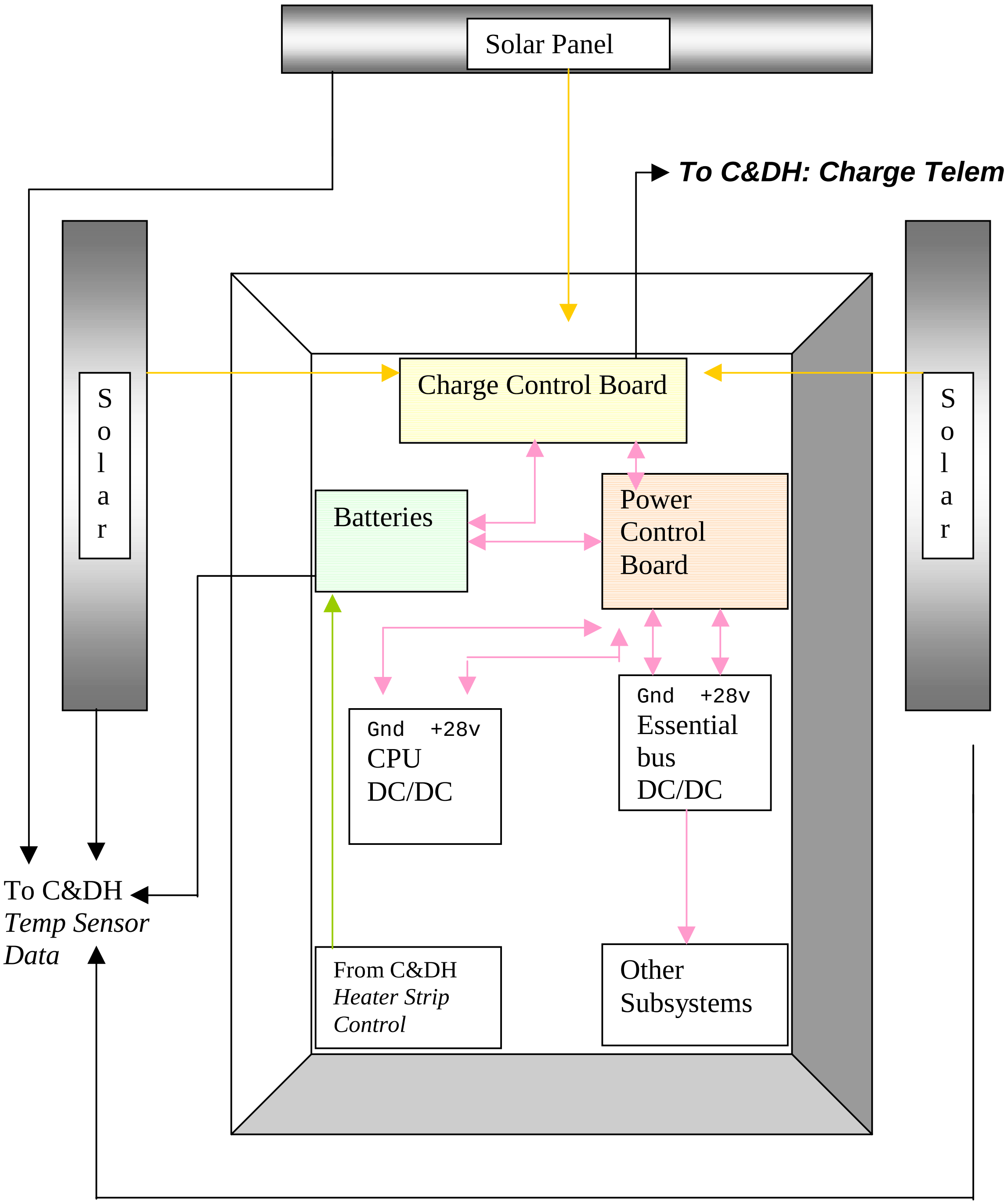


Figure 6.1. Power Block Diagram

6.1.1 Power Management

The power derived from the solar panels goes straight to a Charge Control Board (CCB) which acts like a Peak Power Tracker (PPT) for the solar panels. A PPT was chosen instead of another type of power regulation (like direct-energy-transfer) since it is a non-dissipative power subsystem. This detail is important since the spacecraft will only be using the full array power (see Power Profiles, in Section 6.4) available once a week to support communication with Earth. In addition, it will be drawing much more power at Earth at the beginning of its life (maximum power at Earth is approximately 3 kWh, see the graph in Section 6.3) assuming the arrays can produce 312 W/m^2. These figures have not accounted for the effects of the higher temperatures the solar arrays will be at, or the time spent in shadow. Accordingly, the PPT system will extract only the needed power from the array, causing less internal power dissipation.

Below can be seen the overall power distribution system including some of the cabling. Solid-state relays (SSR) were chosen because of their proven flight history, reliability and low power dissipation. The SSR turns on the non-critical bus conversion electronics, while the critical bus converter is hardwired on.

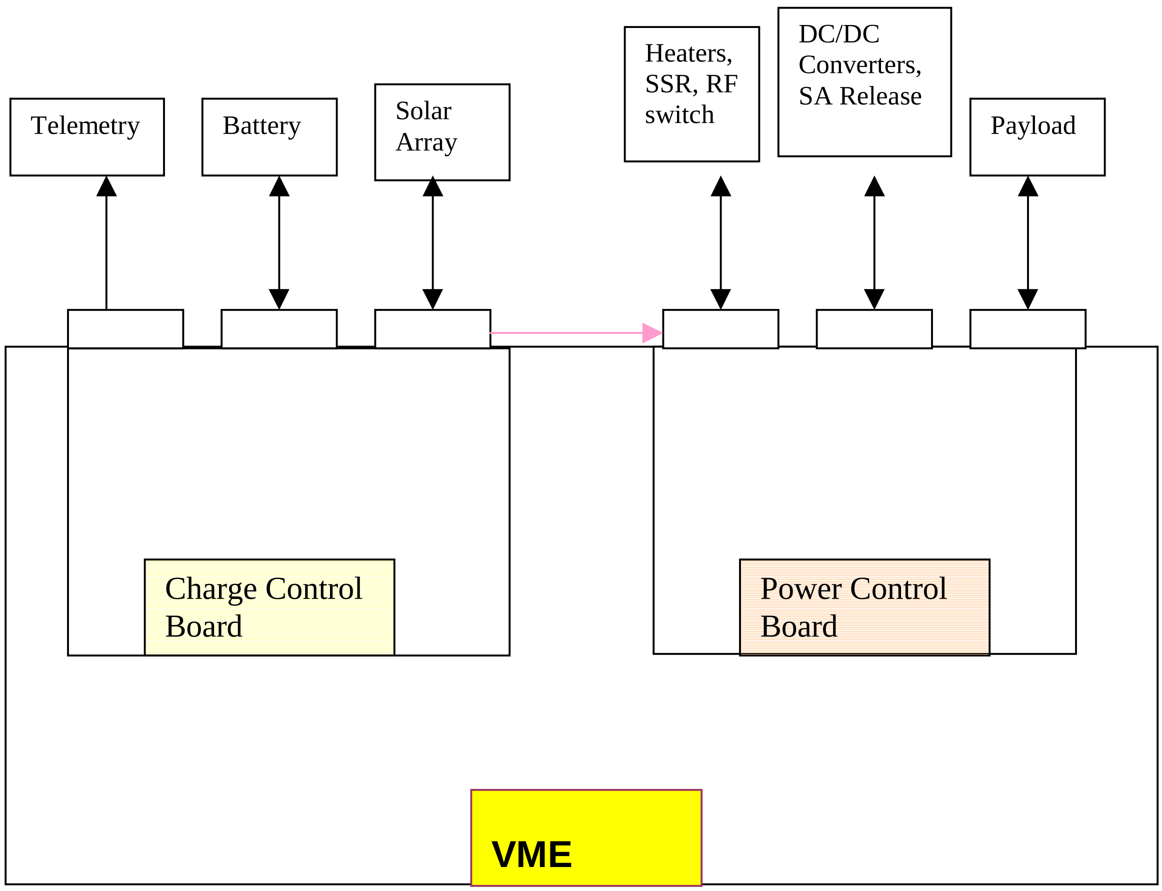


Figure 6.2. **EPS CHARGE CONTROL BOARD MECHANICAL DESIGN**

6.3 Equipment List

6.3.1 Energy Generation

The Sunstroke spacecraft will generate power from three deployable solar panels. The solar cells chosen for this mission are the TECSTAR triple junction $GaInP^2$/GaAs/Ge. They are high efficiency (24%), light weight (.845 kg/m^2 for the bare cell) and have proven flight heritage (missions include: Deep Space I; SSTI/Clark; Mightysat; SSTI/Lewis; SMEX; TRACE). The cells are MIL-Q-9858 certified and ISO 9001 certified. The cells shall be covered with Dow corning 93-5000, the industry standard. The cells shall go though tests for all the assembly and space environments expected for flight, including humidity, soldering/welding, electron, proton and ultraviolet radiation. Other characteristics are listed below.

- Fully space qualified and in flight production
- N-on-P Polarity for enhanced radiation resistance
- Design of top & bottom junctions optimized for end-of-life (EOL) performance
- 10% to 12% more EOL specific power
- Highest dual junction solar cell efficiency in the world
- Highest three junction with lowest radiation degradation in production May 2000
- Advanced product roadmap for higher efficiencies
- Greater than 3 times mechanical strength for reduced attrition in laydown
- Smooth rear surface for ease of laydown with reduced adhesive
- Recommended for ISO 9001 certification
- High radiation resistance (see Appendix A6).

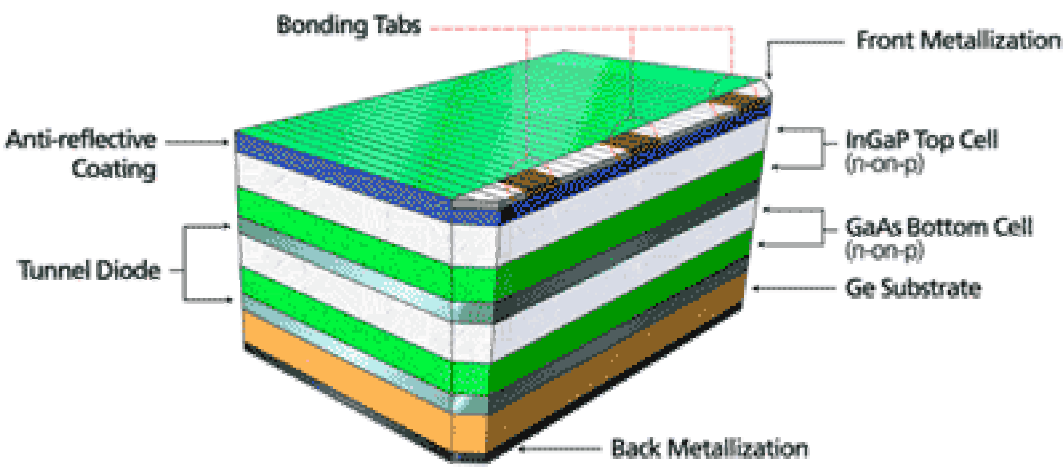


Figure 6.3. Graphical Display of Solar Cell

The design of the satellite is such that the solar arrays will be pointing to the sun during the sunlight period of the orbit because of payload requirements. Thus it was decided not to articulate the solar arrays. Even if the solar panels are a few degrees off perpendicular, there will be very little shadowing from the satellite due to solar array placement (see Section 3.0).

6.3.2 Energy Storage

The batteries are Nickel-hydrogen (NiH_2) from Eagle-Picher Technologies. NiH cells outperform NiCd cells in every category except physical volume. The amp-hr capacity of the batteries was calculated in Appendix A6.3. The specific battery chosen is the SAR-10071 (see Appendix A6) with summarized characteristics shown below.

Table 6.1 Specifics on SAR-10071

Type	Capacity (A/hr)	Mass (kg)	Diameter (cm)	Length (cm) inc. terminals	Specific Energy (Whr/Kg)	Energy density (Whr/l)
SAR-10071	18	8.9	12.8524	56.388	54.6	59.1

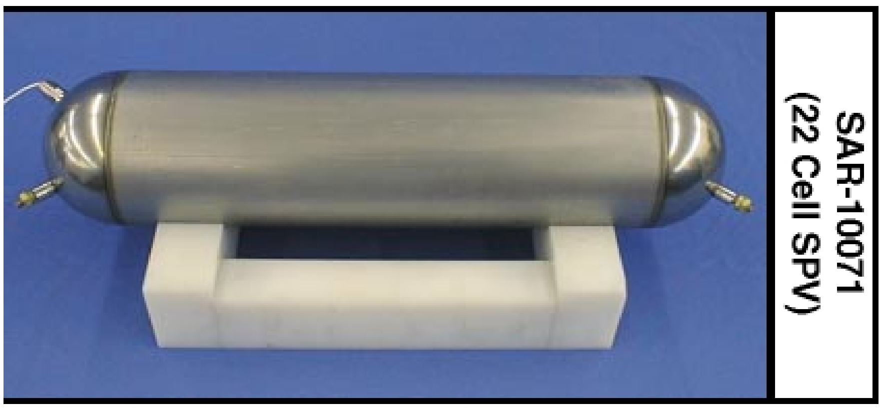


Figure 6.4 SAR10071 Battery

6.3.3 Power Management

The power conditioning boards and battery conditioning boards will be constructed by Spectrum Astro. They have a Charge Control Board (CCB) to charge the batteries. The unit only ways .84 Kg, with an efficiency of 97%, with a power draw of 4 W on average. It also has a proven heritage over four missions. The Power Control Board (PCB) is designed by Spectrum Astro, with .673kg with a power draw of 4.7W. The power conversion for the Sunstroke satellite will probably be separated into two parts. The critical bus DC-DC converter is located on the critical power board, while the non-critical bus DC-DC converter is located on the non-critical power board. The critical bus DC-DC converter is designed to provide power to critical systems on the satellite, such as: Star Trackers, course sun sensors; receiver, transmitter; charge control unit. The Non-critical bus includes: the payload, the computer, propulsion, fine sun sensors.

6.4 Power Profile

The average power needed during the day and night are shown on the table below:

Table **6.2. Power Needed**

	Daylight Power (W)	Eclipse Power (W)
TOTAL avg. power	274.2	215.7
20% Margin	54.84	43.14
TOTAL	329.04	258.84

We do not need to calculate any cosine loses since the satellite will be pointed towards the sun for the entire time the spacecraft is in sunlight around Mars. By doing some rough calculations (taken from Wertz and Larson, *Space Mission Analysis and Design, V. 3*, p.412) and assuming an orbital eclipse such that:

Table 6.3 Orbital Parameters

Parameter	Value	Unit
Period	132.7	min
Orbits/day	10.852	N/A
time in shadow	41	min

- Power to be produced DURING DAYLIGHT for this subsystem: 724.2430 (W)
- our cells produce this much: 120.0 (W/m2)
    - NOTE: this assumes 500 w/m2 on Mars, the minimum case.
- Because of
    - Manufacturing inefficiencies (85%)
    - efficiencies of power conditioning system (90%)
    - battery efficiency (95%),
- it drops to: 82.62 (W/m^2)
    - emperature inefficiencies (estimated as 90%) ,
- Power BOL : 74.0750 (W/m^2)
- Power EOL (after radiatation degredation.): 68.1304 (W/m^2).

The final values are

Table 6.4 Area, Cost, Mass of Solar Array

Parameter	Value	Unit
Area	10.3122	meters
Cost	1,458,654	$
Mass	12.6583	kg

6.4.1 Batteries

A depth of discharge of 50% was chosen since we are using NiH batteries, which will go through about 8000 charge-discharge cycles, assuming about 10.85 orbits a day and a 730 days of orbits. There will be little need to charge/discharge the batteries during the cruise phase since the system will be pulling only about 150 W (210 when transmitting),which is well under what the solar panels will pull in. (see below graph)

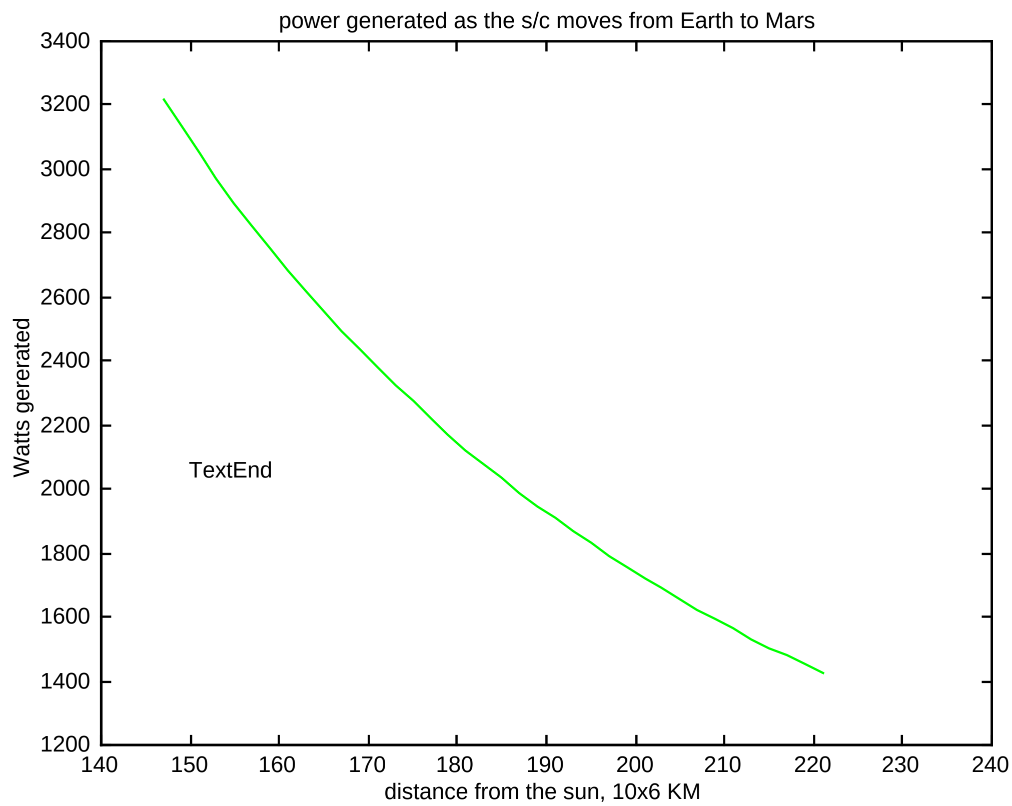


Figure 6.4. Power as Spacecraft Moves from Earth

Assuming an eclipse maximum load of 258 W, we calculate batteries as such:

Table 6.5. Battery Table

Parameter	Value	Unit
Number of batteries	1	N/A
Battery Conditioner Eff.	85%	N/A
Battery Capacity	414.824	W-hr
Battery Capacity	14.815	Amp-hr
20% Margin	82.965	W-hr

Totals	2.93	Amp-hr
	497.788	W-hr
	17.778	Amp-hr

There will be two batteries on board, each with an Amp-hr Capacity of 18. Although only one is needed for the mission, the other will provide redundancy.

6.4.2 Power Curves

The array size given above assumes the maximum power draw, at the farthest point in Mars orbit from the sun with the longest eclipse period and at EOL for the array. The power curves below show the draw for **4** different phases of the spacecraft life including. Cruise; Mars standard orbit; Mars orbit with transmissions to Earth; Safehold mode (with no science, but telemetry to Earth to resolve what is wrong and limited ADCS to control antenna pointing).
Average values for the subsystems are shown below:

Table 6.6. Average Power Values for Each Subsystem

SUBSYSTEM	Quantity	Operating Power (W)	Idle Power (W)
ADCS			
Reaction Wheel	4	7	2
star tracker	2	15	5
Fine sun sensor	1	3	3
Course sun sensor	6	0.25	1.5
IMU	1	31	31
C&DH			
Transmitter	1	63	5
Receiver	2	8	20
Thermal			
heaters	8	2	0
Payload			
solstice instrument	2	16.6	1
Power			
power control board	1	5	5
charge control board	1	4	4
CDH			
electronics	1	43	18.5
Propulsion			
thrusters	12	8.25	0
main engine	1	12	0
TOTALS	42	218.1	91

6.4.3 Power profile during Launch

The launch values were calculated (see Appendix 6), but were not graphed since there will be several one-time power needs (such as opening the solar arrays, etc). Some of those values are shown below.

Table 6.7. Launch Power Values

HOURS UNTIL SOLAR CELL DEPLOYMENT	8	HR
Battery Energy Removed in launch	48.5	Watts
Battery Depth of Discharge	46%	
Power use during cruise phase	152.5	Watts
Battery Energy Removed in Eclipse	215.7	Watts
Battery Depth of Discharge	17%	

6.4.4 Power profile during Cruise

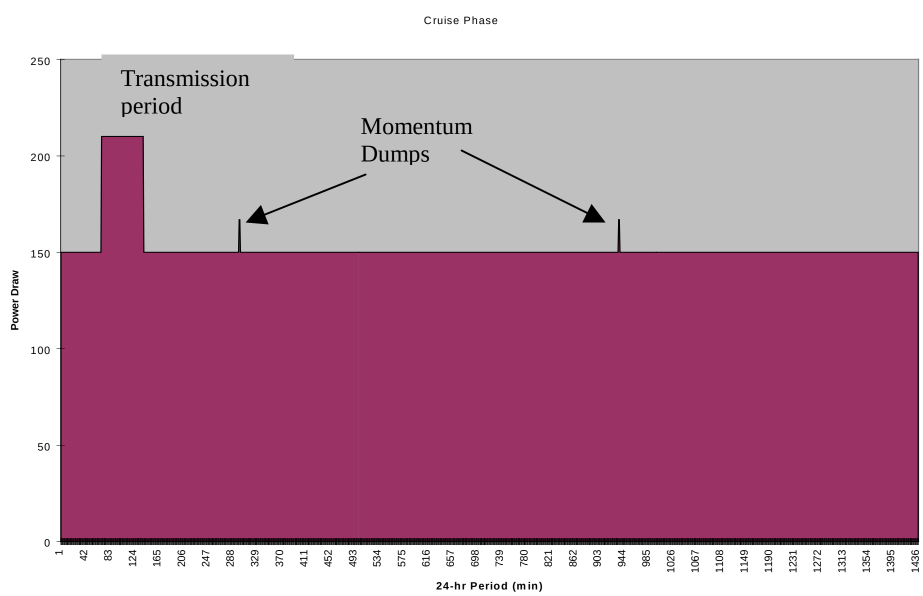


Figure 6.5. Power Profile during Cruise

There will be a 25 minute period once per day where the antenna will connect to Earth's Deep Space Network, transmit telemetry, and upload commands as needed. The firing of the propulsion engine will need a also power draw, but again these are isolated events which can easily be taken care of with the available power.

6.4.5 Power profile during Standard Martian orbit

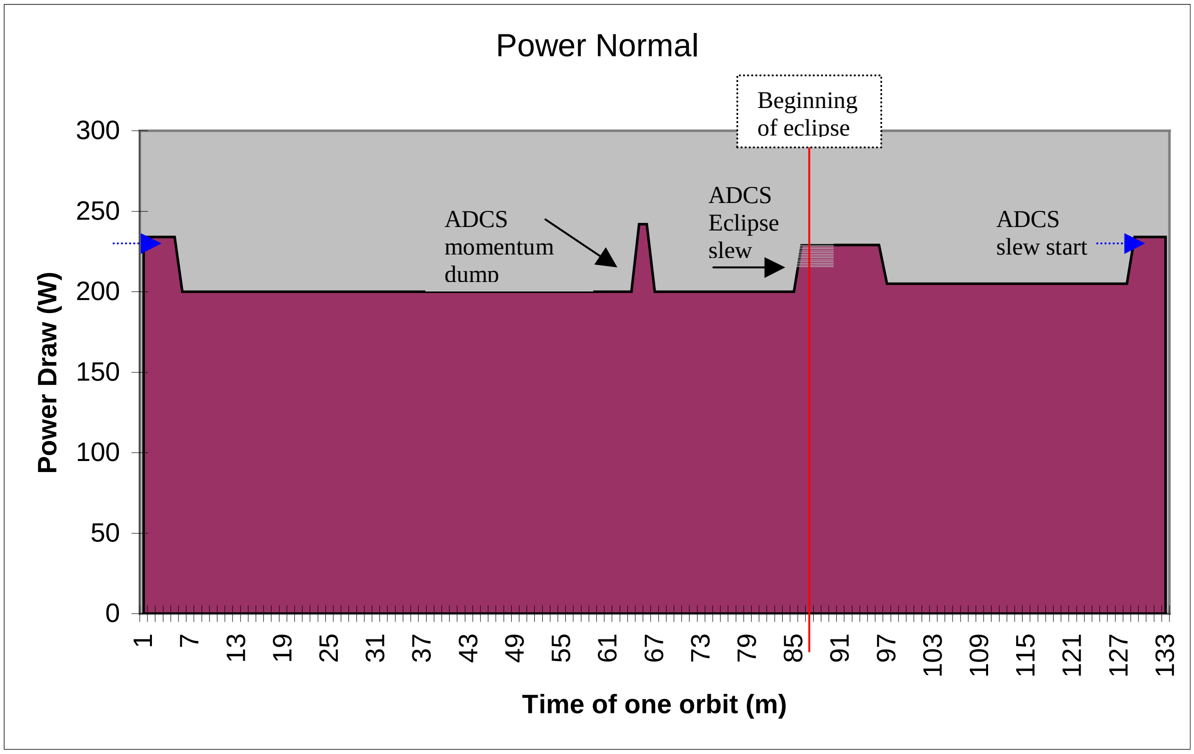


Figure 6.6. Power Profile during standard orbit

The power profile is over one orbit, where zero is the point of sun line and 92 minutes is the eclipse line. There are several peaks on the power profile. The one that appears to start at zero, actually starts at miniute127. This is the slewing of the satellite, where the reaction wheels are drawing full power to turn the spacecraft. The slew starts 5 minutes before the sun line, so when the sun line is crossed, the arrays are able to receive sunlight and power (as the spacecraft is approximately 45° to the sun). 66 minutes into the sunlight period, the thrusters fire to provide a momentum dump. This also needs the wheels to spin (down), although the power draw is less than a slew. Otherwise, the reaction wheels only draw about one watt. However, when the satellite is on the eclipse side, the reaction wheels have to compensate for the larger disturbance torque (since the spacecraft is closer to the planet at this point and the sum of forces, especially aerodynamic drag, is higher). Notice that a second slew happens 5 minutes before the eclipse line (minute 92).

6.4.6 Power profile with Transmission

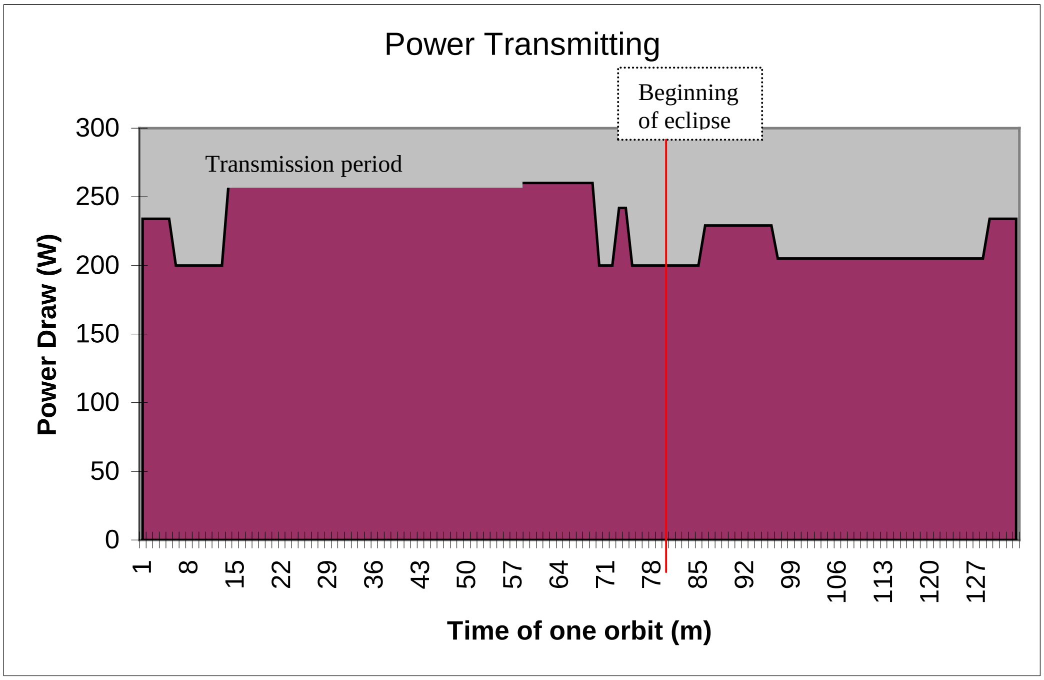


Figure 6.7. Power Profile while Transmitting

This power profile looks the same as Figure 6.6, except that there is a large draw for transmission power. Notice that the slews are still there, however, they are dominated by the transmission power draw (63 W) that takes up most of the sunlight period

6.4.7 Power profile during Safehold Model

Notice that there are no new slues since no science is being collected. During every orbit, the majority of the sunlight period is spent transmitting to Earth, hopefully solving the situation which cause the spacecraft to go into Safehold mode If the spacecraft goes into Safehold mode during the eclipse period, there will be one slew needed to turn the spacecraft back into the proper orientation.

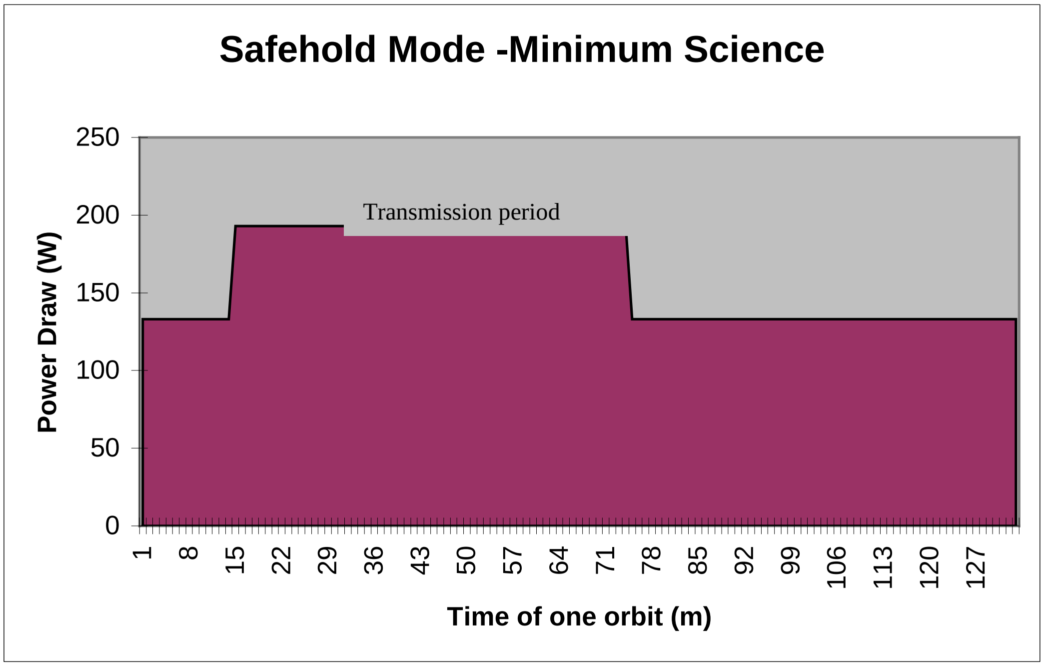


Figure 6.8. Power Profile during Safehold Mode

6.5 Mass and Power Estimates

Table 6.8. Mass & Power Estimates for EPS

		Size	Individual Mass	Total Mass	Idle Power	Operating Power
Subsystem and Component	**Quantity**	**(cm)**	**(kg)**	**(kg)**	**(W)**	**(W)**
Battery	**2**	**12.85 dia. x 56.4**	**9**	**18**	**0**	**0**
Solar Array	**3**	**(10 m^2 total)**	**N/A**	**12.65**	**0**	**0**
Cabling	**N/A**	**N/A**	**N/A**	**18**	**0**	**0**
Control Charge Board	**1**	**12.8 x 21 x 1.275**	**0.9**	**0.9**	**4**	**4**
Power Control Board	**1**	**15.75 x 23 x 1.25**	**.6**	**.6**	**2**	**5**
Totals				**50.3**	**6**	**9**

7.0 Thermal Control Subsystem

7.1 Summary of Requirements

The purpose of the thermal control subsystem is to regulate the temperatures of Sunstroke. The spacecraft thermal subsystem will remain in operation temperatures during both sunlight and eclipse phases in orbit around Mars. Similarly, adequate temperatures will be maintained during the orbit around Earth and the cruise phase of the spacecraft. A passive and active system mix will be used to control the temperatures within the specified range for each subsystem of the spacecraft.

7.2 Approach to Accommodating Requirements

7.2.1 Structural Subsystem

The bus of Sunstroke needs to be able to withstand the temperatures for the worst case hot (WCH), worst case cold (WCC), and any other temperatures experienced during the mission lifetime. From the trade study shown in Appendix A7.1, the external surface of the Sunstroke bus will be bare aluminum. Aluminum was chosen because it has one of the highest temperatures for the WCC as well as a reasonable temperature for the WCH. To ensure that the satellite does have an adequate temperature throughout the mission, a passive thermal system will be used. Multi-layered insulation (MLI) will be installed to the outside of the bus to make sure the bus does not get too hot. All of the connections and instrumentation that extrude from the bus will act as thermal radiators to cool the bus of Sunstroke. The sides of Sunstroke that face away from the sun will also act as thermal radiators to cool the bus.

7.2.2 Propulsion Subsystem

The thrusters and propellant tanks for the propellant system will be located on the side of the satellite that is not facing the sun. This is to ensure that the propellant does not get too hot. The propulsion does need to stay above zero degrees centigrade, so a combination of passive and active thermal control systems will be used. Heat pipes and heater strips will be used to keep the propellant from freezing. In addition the thrusters will also have MLI to keep the temperature fairly constant. The thermal radiators will also be used to cool the thrusters during maneuvers.

7.2.3 Attitude Determination and Control Subsystem

Overall the ADCS subsystem needs to be kept within -30°C and +70°C. Each component of the ADCS has its own temperature requirements. The inside face of the star trackers and sun sensors will be painted black, and the outside will be polished aluminum with

20% black paint. This will give the instruments good thermal coupling and will limit the heat loss during eclipse. The trackers and sensors will also be insulated with MLI to minimize heat gains or heat loses. Heat Pipes will also be used to ensure that the temperature of the reaction wheels stays within the reasonable limits.

7.2.4 Power Subsystem

The solar arrays of Sunstroke need to be able to withstand the temperatures for the WCH and WCC experienced during the mission lifetime. From the trade study shown in Appendix A7.1, the solar cells of the Sunstroke will have a solar cell-fused silica cover on the front. The back of the solar arrays will be painted with white paint, type Z93. This provides the lowest WCH and the highest WCC for Sunstroke.

Nickel hydrogen batteries will be used as an alternate power source on the Sunstroke spacecraft. Thermostatically controlled heaters will ensure that the batteries do not get too hot or too cold during the mission lifetime of the satellite around Earth, Mars, and in the cruise phase.

7.2.5 Command and Data Handling

The temperature for the main computer of Sunstroke will be controlled using heat pipes and thermostatically controlled heaters. The temperature of the electronics will be controlled using thermostatically controlled heaters.

7.2.6 Telecommunications

The telecommunications equipment needs to be kept as cool as possible. This includes the antenna used for transmitting and receiving data. The exposed side of the antenna and any of the support structure will be covered with silvered Teflon because of its extremely low absortivity. The back of the antenna will be covered with MLI. The silvered Teflon and MLI will help to reduce the temperature gradient and to limit any solar heating. The telecom equipment inside the spacecraft will have heat pipes and thermostatically controlled heaters to cool or heat the system as needed.

7.2.7 Instrumentation

The instrumentation on the Sunstroke satellite is SOLSTICE. The temperature of SOLSTICE will be controlled using thermostatically controlled heaters.

7.3 Expected Equipment Temperatures

7.3.1 Worst Case Hot (WCH)

The WCH scenario for Sunstroke is when the satellite is in an 800-km orbit around the Earth. At this point Sunstroke will be between the Sun and the Earth. This places the spacecraft at a distance of $1.464992*10^{11}$ m away from the Sun. Using the equations in Appendix A7.2 the WCH can be determined for the spacecraft body and the solar arrays. For the spacecraft if bare aluminum is used, the WCH is 83.94°C. If a solar cell-fused silica cover is used on the solar panel, with white paint, type Z93 on the back of the arrays, the WCH is 60.99°C.

7.3.2 Worst Case Cold (WCC)

The WCC for Sunstroke is when the spacecraft is in eclipse at a 400-km orbit around Mars. At this point Mars will be at apogee around the Sun. Sunstroke will be at a distance of $2.482004*10^{11}$ m from the Sun. The WCC for the spacecraft and the solar arrays can be determined using the equations in Appendix A7.2. For the equivalent surface area of a spherical spacecraft, the WCC is –66.2°C. For the solar panels, the WCC is –41.04°C.

7.3.2 Subsystem Temperature Constraints

The temperature requirements for each subsystem are listed in Table 7.1.

Table 7.1: **Temperature Requirements for Each Subsystem**

COMPONENT	TEMPERATURE RANGE MIN/MAX (C)
Bus	-66.2/83.94
Propellant	0/40
Sun Sensors Star Trackers Reaction Wheels	-15/60 -15/60 -15/60
Solar Arrays Batteries	-20/30 -5/25
Computer Electronics	0/40 0/40
Antenna Transponder	-20/30 -10/30
SOLSTICE	-45/50

7.4 System Description and Equipment List

7.4.1 Materials and Paints

Different materials and paints will be used to get specific absortivities and emissivities for Sunstroke to ensure that the spacecraft does not get too hot or too cold. The materials and paints that will be used are

- Bare aluminum for the spacecraft structure.
- Black paint for the inside face of the star trackers and sun sensors
- Polished aluminum with a little black paint for the outside face of the sensors
- Solar cell-fused silica cover for the front of the solar cells
- White paint, type Z93 for the back of the solar arrays
- Silvered Teflon for the exposed side of the antenna

7.4.2 Multi-Layer Insulation (MLI)

Multi-layer insulation will be used to minimize heat gradients and heat loses. The sections of Sunstroke that will use MLI are

- Outer surface of the satellite bus
- Thrusters
- Star trackers and sun sensors
- Back of the antenna

7.4.3 Thermal Radiators

Thermal radiators are used to radiate excess heat out into space. The subsystems that have thermal radiators are

- Outer surface of the bus, on the connections and instrument extrusions
- Thrusters

7.4.4 Heat Pipes

Heat pipes are used to transfer heat away from an area that is too hot or towards an area that is too cold. The heat pipes that will be used for most of Sunstroke will be flat or flexible heat pipes. With these types of heat pipes, it makes it easier to interface them with the other subsystems. Heat pipes will be used for

- Thrusters
- Reaction wheels
- Main computer
- Telecom equipment

7.4.5 Thermostatically Controlled Heaters/Heater Strips

Thermostatically controlled heaters will be used to add heat depending on if the subsystem needs more temperature control. These heaters will be used for

- Thrusters
- Batteries
- Computer
- Electronics
- Telecom equipment
- SOLSTICE

7.5 Mass and Power Estimates

Table 7.2 describes the mass and power estimates for the thermal subsystem.

Table 7.2: **Thermal Mass and Power Estimates**

Component	Quantity	Mass (kg)	Total Mass (kg)	Power (W)
White Paint (1MIL)	10 m^2	1	1	0
Black Paint (1MIL)	.25 m^2	0.025	0.025	0
MLI	51.75 m^2	1.5	1.5	0
Silvered Teflon	1.77 m^2	N/A	N/A	0
Thermal Radiators	1	0.85	0.85	0
Heat Pipes	6	0.18	1.08	0
Thermostatic Heaters	8	0.05	0.4	16
Temperature Sensors	60	N/A	0.5	N/A
TOTAL			5.355	16

8.0 Command and Data Handling (CDH)

8.1 Requirements

The Sunstroke C&DH system will be required to supply health monitoring and house keeping data of the spacecraft. This includes at a minimum the S/C orbit and attitude, battery temperature and state of charge. As well, the C&DH system shall provide real-time autonomous navigation. It shall command the decoding and execution of commands, protect from processor hang-ups, protect from memory anomalies, protect against single event upsets, provide S/C timekeeping, and protect from latch-up conditions. For each subsystem it manages, it shall protect from power system anomalies, maneuvers that threaten ADCS constraints, and protect from dangerous power draw of the payload and instruments. It shall provide the SOLSTICE payload with a nominal data rate of 344 bps, route commands directly from the processor, and provide memory storage for 5 days of science plus telemetry.

8.2 Data Rates

Telemetry data on Sunstroke is collected from an assortment of 128 sensors at various monitoring frequencies. Initial calculations indicate the data rate required for telemetry is 82 bps. A complete list of these sensors by subsystem is located in appendix A8 while the summarized version is shown below.

PROPULSION

COMPONENT	QTY
MAIN TANK(S)	
*PRESSURE SENSOR	1
*TEMP SENSOR	2
FEED LINES	
*PRESSURE SENSOR	2
THRUSTERS	
*TEMP SENSOR	12
*CURRENT MONITOR	12
MAIN THRUSTER	
*TEMP SENSOR	2
VALVES	
*POSITION FEEDBACK	10

STRUCTURE/THERMAL

COMPONENT	QTY
TOP PLATE	
*TEMP SENSOR	4
MID DECK	
*TEMP SENSOR	4
SIDE PANELS	
*TEMP SENSOR	4
PROPULSION MODULE	
*TEMP SENSOR	4

POWER

COMPONENT	QTY
BATTERIES	
*CHARGE LEVEL	6
*VOLTMETER	6
*CURRENT MONITOR	6
*TEMP SENSOR	6
SOLAR ARRAYS	
*CURRENT MONITOR	3
*TEMP SENSOR	3

TELECOM

COMPONENT	QTY
TRANSPONDER	
*TEMP SENSOR	2
*CURRENT MONITOR	2
AMPLIFIER	
*TEMP SENSOR	2
*CURRENT MONITOR	2
LGA	
*TEMP SENSOR	2
HGA	
*TEMP SENSOR	1

ADCS

COMPONENT	QTY
REACTION WHEEL	
*TEMP SENSOR	4
*CURRENT MONITOR	4
IMU	
*TEMP SENSOR	1
*CURRENT MONITOR	1
STAR TRACKER	
*TEMP SENSOR	2
*CURRENT MONITOR	2

CDH

COMPONENT	QTY
CDH STATUS	1
*TEMP SENSOR	2
*CURRENT MONITOR	2

PAYLOAD

COMPONENT	QTY
SOLSTICE	
*TEMP SENSOR	4
*CURRENT MONITOR	2

Figure 8.1 Telemetry Sources

The SOLSTICE payload contributes a 344 bps data rate if active. The telemetry has a maximum data rate of 82.0 bps. The Sunstroke communications system allows for a downlink rate of 21.3 Kbits per second at maximum distance. It should be noted that the data must have a Reed-Solomon coding effort of 2.5 and a compression of 2.0. This gives a total of 17.1 Mbits of data per downlink. Thus the amount of time required to send all data collected over a 24 hour period is 801 seconds. Because of the small transmit time required, the Sunstroke program will only transmit data every 7 days to free up time needed on the Deep Space network and reduce overall mission operating cost. The transmit time required at maximum distance every 7 days is approximately 93.5 minutes. With a possible transmit time of XXX per Mars orbit, a total of XXX orbits will be required to send data. The total time on the DSN dish will be XXX.

8.2 System Block Diagram

The Sunstroke C&DH subsystem will use VME style bus architecture. It shall be composed of eight 6U VME cards connected to a common backplane.

Slot	Component
1	Attitude Control Interface (ACI)
2	Power Control Board (PCB)
3	Charge Control Board (CCB)
4	Spacecraft Processor Module (SPM)
5	Payload Module Memory Board (PMMB)
6	SPARE
7	SPARE

Figure 8.2 VME Board List

The ACI board shall provide the interface for controlling and monitoring the 4 reaction wheels, IMU, 2 star tracers, fine sun sensor, 3 coarse sun sensors, 2 axis gimbal on the HGA, and the thrusters. The board shall be a modified Spectrum Astro PACI board. The modification includes acceptance of an additional star tracker as well as control of the 2-axis gimbal. Other specifications of the board include 96 analog inputs for sensors and 7 digital inputs and outputs for digital sensors. The functional diagram is shown below.

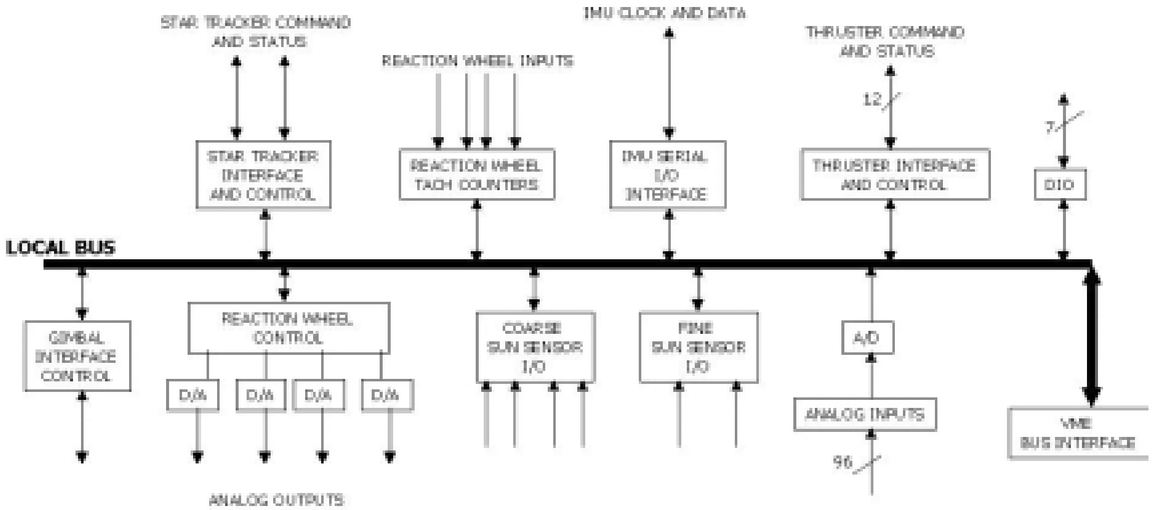


Figure 8.3 Attitude Control Interface Card Functional Diagram

The Charge Control Board (CCB) shall provide an interface to the batteries, solar arrays, and ground testing umbilical. Its primary operation will be to maintain bus voltage and the battery state of charge. The current best estimate for power and mass is based on the Spectrum Astro CCB used on the Deep Space 1, HESSI, and SWIFT missions.

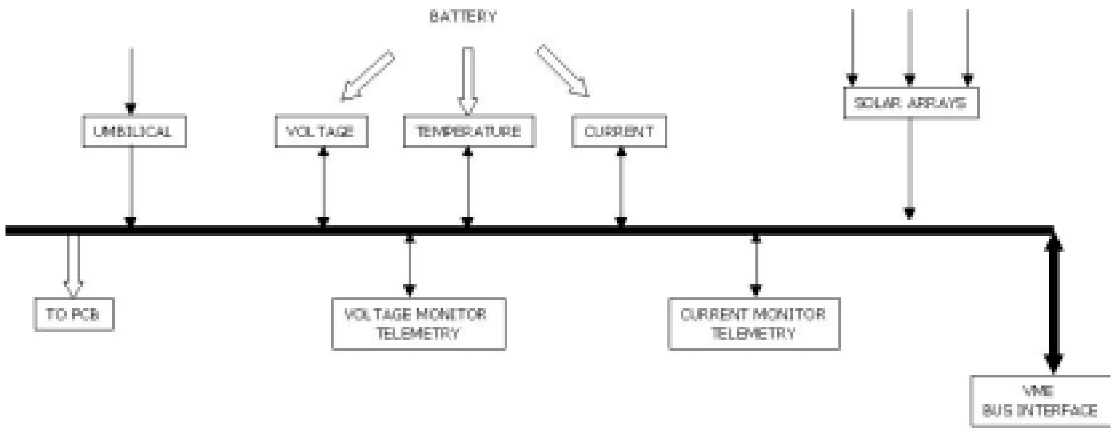


Figure 8.4 Charge Control Board Functional Diagram

The Power Control Board (PCB) shall work in conjunction with the CCB. Its main purpose is to distribute the power needed to the instruments on board the S/C as well as provide all the current monitors for housekeeping and payload.

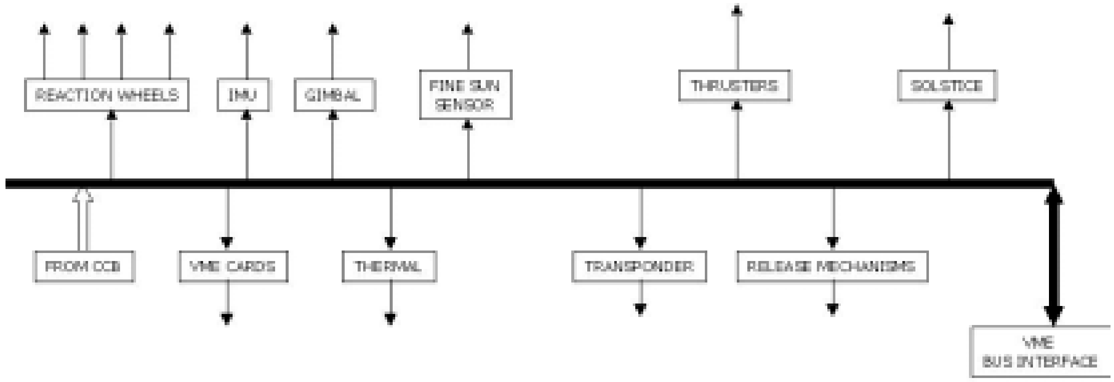


Figure 8.5 Power Control Board Functional Diagram

The Spacecraft Processor Module (SPM) shall use a Harris Electronics rad hardened processor with a throughput of 31 MIPS at 20 MHz. It will have 512kbyte of boot EEPROM 2DMA, 4 timers, Async serial port, and a 40 Mbps Serial port. The SPM shall be responsible for collecting and sorting all telemetry and science data to send to the transponder via a RS 422 connection for downlink. It is noted that the transponder has the Reed-Solomon encoding and decoding capabilities.

The Payload Module Memory Board (PMB) shall provide a direct interface for the SOLSTICE payload consisting of a 344 bps nominal data rate pathway and command handling. As well it shall provide 128 Megabytes of DRAM for science and telemetry storage.

Finally there will be space for spare boards to allow easy upgrade to the system or to accommodate another payload. It also creates a flexible design for future missions.

The overall system interface diagram is shown in section 1.2 of this document.

8.4 Data Storage Capability

The C&DH system uses solid state memory due to its small size and lack of moving parts. Magnetic disk based storage units and tape recorders are large, contain moving parts, and consume larger amounts of power than solid state. Additionally the science and telemetry requirements for this mission are extremely small with initial calculations showing 6.5 Mb of required storage capacity for every 24 hours. Thus the memory storage board shall be combined with the payload interface board. To meet the 5-day science data storage requirement with appropriate factors of safety, 128 Mbytes of DRAM shall be used. The memory used is EDAC compliant and shall withstand at least 30kRad radiation dosage.

8.5 Mass and Power Estimates

The standard VME card measures approximately 6.3in X 9.2in X 0.5in with a weight of .1kg with no components on it. Using this information, a general packaging scheme for the C&DH system has been developed as shown below.

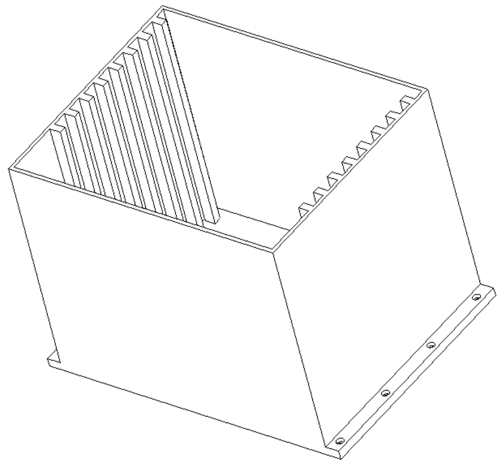


Figure 8.6 CDH Structural Interface

It should be noted that to maximize packaging, the DC/DC converters will be mounted to the inside wall where there are no VME card slots. Basing information gathered from Spectrum Astro and Harris Electronics, the following estimates for power and mass were calculated.

COMPONENT	CBE MASS [KG]	CBE MAX POWER [W]	CBE NOMINAL POWER [W]	CBE STDBY POWER [W]
ACI	0.5	6	6	5
PCB	0.6	4	4	2
CCB	0.8	4	4	2
SPM	1	10	10	1
MMB	0.5	5	5	1.5
SPARE	0	0	0	0
SPARE	0	0	0	0
VME BACKPLANE	0.5	4	4	2
ENCLOSURE	3.6	0	0	0
TOTAL	**7.5**	**33**	**33**	**13.5**

Figure 8.7 CDH Mass and Power Estimates

9.0 Telecommunication System

9.1 Summary of Requirements

The telecommunication system provides a way of communication between the spacecraft and the ground station. This is provided in all of the following six stages of the mission:

- Pre-Launch
- Initial acquisition
- Trajectory to Mars
- Orbit insertion at Mars
- Science phase in orbit at Mars
- Maximum range

In addition to these the emergency mode provides a way of communication also if the high gain antenna can't be pointed correctly, e.g. if the attitude control of the spacecraft is lost. Table 9.1 and Table 9.2 show details of the different modes and the different mission stages.

Table 9.1 Communication Modes

	Communication Mode	**Direction**	**Data rate [bps]**	**Power Margin [dB]**
1	Command	Up	5	10
2	Engineering and Payload Command	Up	60	10
3	Health and Status	Down	60	10
4	Engineering and Science Telemetry	Down	21,300 (max range) 42,700 (mid range) 100,000 (close range)	5
5	Emergency	Up	5	10
6	Emergency	Down	65	10

Modes 1 and 3 will be used at initial acquisition. To provide a link even if the spacecraft attitude is unknown the hemispheric antennas are used for communication (data rates as for emergency mode).

Table 9.2 Communication Stages

Stage	**Range**	**Uplink Mode**	**Downlink Mode**	Comment
Pre-Launch	-	-	-	Conductor bound communication provided by launch vehicle
Initial acquisition	800 km	1	3	Omni-directional antenna
Trajectory to Mars	800 km -	2	4	High gain antenna
Orbit insertion at Mars		2	4	High gain antenna
Science phase in orbit at Mars	<3.78e11	2	4	High gain antenna
Maximum range	3.78e11	2	4	21,300 bps
Emergency	<= max	5	6	RX: Omni-directional antenna TX: Low gain antenna

The transmitter can send signals coherent to the received signal for range and range-rate measurements. This is important for navigation.

9.2 Summary of Link Calculations

9.2.1 Communication Frequencies

The following frequency ranges have been allocated by the International Telecommunication Union (ITU) for use in deep space research:

- S-band uplink 2110-2120 MHz
- S-band downlink 2290-2300 MHz
- X-band uplink 7145-7190 MHz
- X-band downlink 8400-8440 MHz

The JPL Frequency Manager assigned channel 21 to the Sunstroke project. This channel represents an uplink frequency of 2115.699846 MHz (S-band) or **7170.403550 MHz** (X-band) and a downlink frequency of 2297.592593 MHz (S-band) or **8424.506174 MHz** (X-band). We will use the X-band frequencies for uplink and downlink. The advantage of X band frequencies over S band frequencies is the higher gain of a given antenna size (higher frequency -> higher gain due to smaller beamwidth). A change from the initial frequencies proposed in the RFP was necessary because those frequencies were not supported by the DSN for coherent range and range-rate measurements. We get a frequency ratio of

$$\frac{f_d}{f_u} = 1.1749 = \frac{309}{263}$$

9.2.1 Link Budget

The detailed link budgets are in the Appendix A9.

Table 9.3 Summary of link modes (normal mode)

Distance range [m]	Data rate [bps]	Uplink	Downlink	Budget
0 .. $1*10^{11}$	100,000	34m HEF DSN	34m HEF DSN	See Appendix A9
$1*10^{11}$ .. $1.55*10^{11}$	42,700	34m HEF DSN	34m HEF DSN	See Appendix A9
$1.55*10^{11}$ .. $3.1*10^{11}$	42,700	34m HEF DSN	70m DSN	See Appendix A9
$3.1*10^{11}$ ..	21,300	34m HEF DSN	70m DSN	See Appendix A9

For up- and downlink the parabolic antenna is used on the spacecraft.

Table 9.4 Summary of link modes (emergency mode)

Distance range [m]	Uplink		Downlink		Budget
	Groundstation	Spacecraft	Groundstation	Spacecraft	
0 .. $2.1*10^{10}$	34m HEF DSN	Hemispheric	34m HEF DSN	Hemispheric	See Appendix A9
$2.1*10^{10}$ .. $3.5*10^{10}$	34m HEF DSN	Hemispheric	70m DSN	Hemispheric	See Appendix A9
$3.5*10^{10}$ ..	34m HEF DSN	Hemispheric	70m DSN	Parabolic	See Appendix A9

At larger distances (from $3.5*10^{10}$ meter to maximum distance) emergency downlink is only possible with the high gain parabolic antenna. At ranges of less than $3.5*10^{10}$ meter up- and downlink will use the hemispheric antennas. If the spacecraft will get out of control at large distance, there will be no downlink available (at the given maximum bit error rate), as the parabolic antenna has to be pointed correctly.
Needed downlink times are given in the C&DH section.

9.3 System Block Diagram and Description

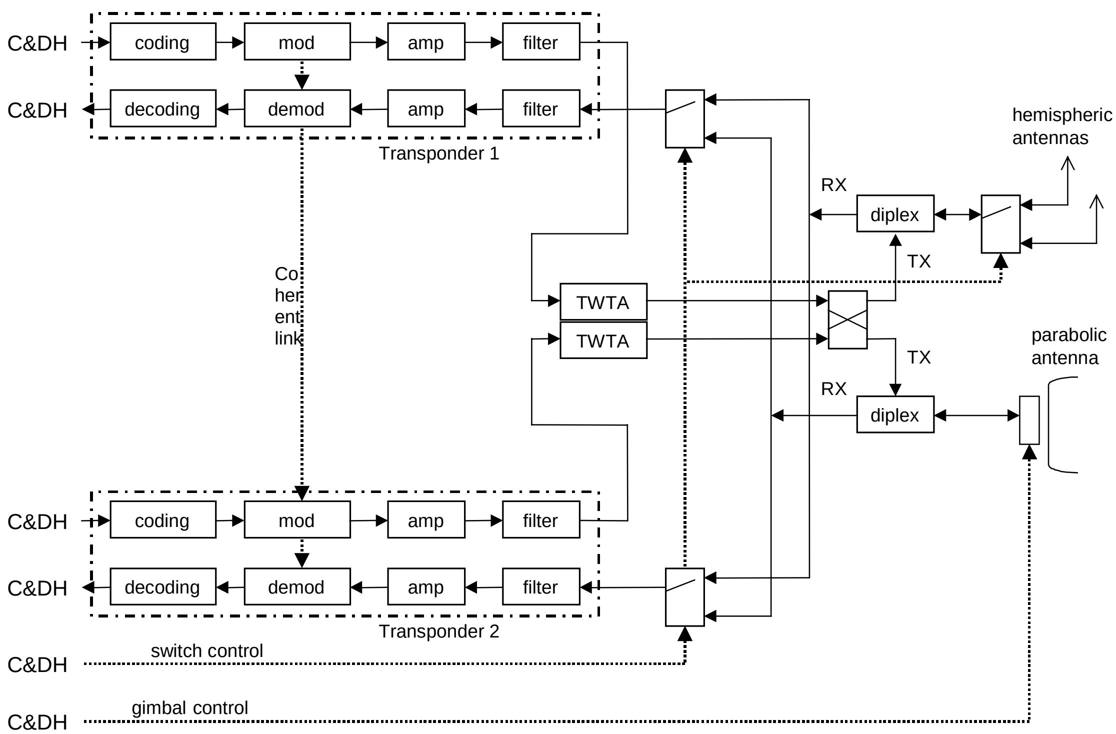


Figure 9.1 Telecomm Block Diagram

The telecommunication subsystem consists of a high gain parabolic antenna on a gimbal, two hemispheric low gain antennas, two transceivers and two travelling wave tube amplifiers (TWTA). The system is fully redundant and can overcome at least a single defect part.

9.4 Equipment List

Transceiver (2x):

Motorola Small Deep Space Transponder, X Band (7.1704036 GHz uplink, 8.424506174 GHz downlink). Includes Coding (Reed Solomon) and a CDU (Command Detector Unit). Class B.
Heritage: Mars Surveyor 2001, Space Based Infra-Red Telescope Facility (SIRTF), Mars Micromission and Deep Impact.

TWTA (2x):

Travelling wave tube amplifier, 30 W RF power, 50 W power consumption.
Heritage: Nahuel 1 + 2, Amsat

Switching and diplexing subcircuit (1x)

Containing single fail proof switches, 2x single pole double throw (1P2T), 1x double pole double throw (2P2T). Diplexer for parabolic antenna and diplexer for hemispheric antennas.

Low gain hemispheric antennas (2x)

High gain parabolic antenna (1x):

1.5m diameter. Mounted on mast and gimbal. 39.9 dB gain (downlink).

Cabling, waveguide and mechanical fixtures.

9.5 Antenna Characteristics

The spacecraft will carry three antennas. Two hemispheric antennas will cover a 360-degree field of view (with a few dead spots such as the solar panels). The hemispheric antennas will be used for the initial acquisition in earth orbit. After that these antennas will only be used for emergency transmissions. Emergency uplink will always go over the hemispheric antennas, whereas the emergency downlink is only possible within a certain range (see link budget section).

The 1.5-meter parabolic high gain antenna has a beam-with of 1.66 degree. This antenna will be used for all non-emergency communication, except for the initial acquisition in earth orbit. It also will be used for emergency downlink within the range where it is not possible to use the hemispheric antennas.

9.6 Mass and Power Estimates

Table 9.5 Mass and power estimates

Component	Quantity	Mass [kg]	Power [W]	Dimension [cm]
Antenna Parabola	1	8	0	150 dia. x 70
Antenna Hemis.	2	0.25	0	8 dia. x 4
TX Amplifier TWTA	2	2.3	50	30 x 10 x 8
Transponder	2	3.1	RX: 8 RX & TX: 13	12 x 18 x 14
Switches & Diplexer	1	0.5		10 x 10 x 2
Wiring	1	3	0	
Mechanical and others	1	3	-	
TOTAL		25.8	RX:8 TX:63	

Sources Appendix A9.3

9.7 Cost

The total material cost of the telecommunication system comes to $4,209,000. Details are shown in the appendix.

Alternatives to reduce cost are:

- Only selective redundancy (especially transponder and TWTA)

- Use telecommunication link (Mars <-> Earth) provided by Mars 2001 and Mars Micromission. In this case, Sunstroke would only need an UHF link in Mars's orbit. For the Earth Mars trajectory would only a low data rate, low cost S band link be required. In this case, the Sunstroke project would depend on a success of the Mars 2001 and Mars Micromission.

10.0 Cost Proposal

10.1 Brief Summary of Costs by Work Breakdown Structure (WBS) element

Cost is a major driver for this mission. Because of the interplanetary nature of this mission, and the new political mandate to "not fail", it may prove difficult to stay within the cost cap. Below is the cost proposal for the flight spacecraft. It uses the Work Breakdown Structure (WBS) (see Appendix A10).

100 Management

The cost for management is tallied as the management for the project, as well as the secretaries, accountants, janitorial staff, and general labor pool. These costs are estimated at .75 million dollars per year. These costs also include Performance Assurance and Safety. In Section 11.4, some cost cutting techniques are elaborated; the table there has been reproduced here for visualization

Table10.1. Cost Cutting Techniques

Method	Mechanism	Comments
Trading on Requirements	Eliminates non-critical requirements; permits low-cost technology	Science maybe de-scoped. Working with the science team is critical
Concurrent Engineering	Increases feedback between engineering and management	Can achieve optimal design
Design to Cost	Makes costs paramount	Rarely used
Schedule Compression	Reduces the overhead of the labor pool; forced program drives down cost	Can results in poor initial design; work vs. schedule must be reduced
Reducing the Cost of failure	Allows both ambitious goals and calculated risk in order to make major progress	Fear of failure can drive cost-spiral
Using Microprocessors	Minimizes weight; allows on-orbit programming	Single-event upsets; software development may spiral out of control
Large Margins	Reduces testing; cost of engineering, manufacturing	Allows the ability to "buy" the way out of a problem
Using non-space equipment	Takes advantage of existing designs	Not space qualified
Autonomous systems	reduces operation costs	Can increase non-recurring costs
Standard components and interface	Reduces cost and risk by reusing hardware	Standardization is standard in most manufacturing; however, spacecraft are individual creatures

200 System Engineering

The cost for system engineering are several and varied. Most of the equipment will be contracted out instead of being built in-house.

300 Spacecraft System

The costs for the sunstroke spacecraft have been tallied as:

Table 10.2 Spacecraft System Costs

ITEM	Fy1 (million$)	Fy2 (million$)	Fy3 (million$)	Total (Real Yr.) (million$)
	Inflation mod: 0%	Inflation mod: 3.9%	Inflation mod: 4.1%	
Phase A/B	6.5	1.039	0.260	7.799
Phase C/D	5	48.833	11.451	65.284
ELV and services	0	2.078	3.123	5.201
Tracking Support	0	0.000	4.164	4.164
Total Cost	11.5	51.950	18.998	**82.448**
				Minus Phase A/B
			Total:	**74.649**

Phase A/B costs have been subtracted from the total monies spent, as stated in the RFP.

To help estimate the costs, a cost model was employed. This was used with a grassroots technique (a method where many individual estimates of parts plus labor is done), to compare the two accounting system together. The grassroots system is cumbersome for this early design trades, and some guestimates were made. The results of the cost model are in each subsection,

The basic equation of this cost model is

$$\$ = A\,(x)^B$$

Where $ is the cost of designing, developing, fabricating, testing, and spares cost. A is a calibrating constant. X is the value of a performance parameter. And B is an exponent. Table 10.3 below shows the cost of each subsystem, plus modifier. The Inheritance Modifier is for where the equipment comes from. The modifier is 1 if all new, .7 if moderate, .4 if “off shelf”. The Quality Modifier is for quality class (Class A is 1, Class B is .8, Class C is .6).

Next to the subsystem cost (in Table 10.3), is the “grassroots cost” – the cost of the components, not including the engineering. As you can see, it falls well below the costs estimated by the cost model. The true figure lies somewhere in between, and both numbers were used as a guide to estimate total costs. All monies above a beyond the grassroots costs will be considered overhead and will be spent accordingly.

Table 10.3 Cost Estimator Table

Subsystem	unmodified output	inheritance modifier	quality modifier	subsystem cost	grassroots cost
-	($ in millions)	-	-	($ in millions)	($ in millions)
solar panels	4.672	0.4	0.8	1.495	1.458
Power conditioning	12.383	0.4	0.8	3.963	1.268
ADCS	141.670	0.4	0.8	45.334	3.500
Propulsion, Liquid	22.669	0.7	1	15.868	11.108
mechanical devices	7.926	0.4	0.8	2.536	2.530
C&DH	27.150	0.7	0.8	15.204	8.514
Telecom	30.205	0.4	0.8	9.666	4.500
Integrating structure	17.800	0.4	0.8	5.696	5.690
			subtotal	99.762	38.568
			time (3yr)	1.000	1.000
			new subtotal	99.762	38.568
			plus15% margin	114.726	44.353
			reserves	11.473	4.435
			TOTAL	126.199	48.788

Clearly, the subsystem cost and the grassroots cost are widely divergent. All costs above a beyond the grassroots cost will be applied as overhead.

How each unmodified output was generated is shown below:

310 Structure Cost

Using the cost model, we note that the craft is a planetary orbiter, with an equation of

11.1(4)^.5, or 17 million.

The mechanical devices have their own separate equation, which is

4.1(# of separations)^.6 = 7.92 million

(there are 3 separations: the solar panels, the antenna, and the launch vehicle).

320 Thermal

There is no cost model for this subsection, so we estimate roughly on the equipment needed. However, the integrating structure has the thermal cost built in. In any case, thermal costs are typically around 5% of the total spacecraft cost.

330 Electrical Power Subsystem Cost

Using the cost model, the solar panels are panels with have an equation

1.1^.6.7(Peak kWh)^.4

(a 1.1 modifier since we are using GaAR cells).

The peak kWh is about .32, which works out to be 4.67 million.

The grassroots cost of solar panels from TecStar is $1,458,000

The power condition system has an equation

1.1*19.9*(kWh)^0.5,
(modifier of 1.1 since we are using NiH batteries)
which works out to about 12 million.
The grassroots cost is about 1.2 million

340 Attitude Determination & Control Subsystem Cost
Using the cost model, we have a complex system with 3-axis control and high accuracy with star trackers.

44.8*(Degrees of accuracy)^(-0.5).

The accuracy for this mission is .05, so the cost is 200 million.
However, the cost of the ACS components can also be calculated in the grassroots approach as shown below.

Table 10.4 ADCS Costs

Item	**Number**	**Cost (in K$)**	**Total Cost (in million$)**
Reaction Wheel	4	400	1.600
Star tracker	2	700	1.400
Course sun sensor	4	8	.032
Fine sun sensor	1	20	.020
IMU	1	500	.500
		TOTAL:	3.500

350 Propulsion Subsystem
Using the cost model, we have liquid system where the equation is

0.0000004*(Isp)^2.9*(fuel Mass*2.2*Isp/1000)^0.4,

which works out to 14.12 million.

360 Telecommunications Subsystem Cost
The basic algorithm is

2.3(complexity)^.8.

The complexity depends on the antenna type (fixed parabolic) and use of multiple bands. Since we are using two bands (S and X), we have a complexity index of 25, resulting in 30 million.

The cost estimate of the telecommunication system using initial vendor quotes is shown in Table 10.5

Table 10.5 Telecom Costs

Part	Quantity	Price each	Price total
Transponder	2	900,000 $	1,800,000 $
Parabolic antenna	1	1,300,000 $	1,300,000 $
Hemispheric antenna	2	12,000 $	24,000 $
Switch and diplexer unit	1	60,000 $	60,000 $
TWTA	2	500,000 $	1,000,000 $
Cabling, waveguide		15,000 $	15,000 $
Mechanical		10,000 $	10,000 $
Total cost			**4,209,000 $**

Alternatives to reduce cost are:

- Only selective redundancy (especially transponder and TWTA)
- Use telecommunication link (Mars <-> Earth) provided by Mars 2001 and Mars Micromission. In this case, Sunstroke would only need an UHF link in Mars's orbit. For the Earth Mars trajectory would only a low data rate, low cost S band link be required. In this case, the Sunstroke project would depend on a success of the Mars 2001 and Mars Micromission.

370 Command & Data Handling Cost
Using the cost model

2.3(Complexity)^.7,

The complexity is architecture (central computer), 1 instrument controlled, formatted data, data storage on solid state, and we use redundant state of the art microprocessors, which comes out to 34, or 30 million.

400 Flight and Ground software

Because of the unique character of spaceflight, most software must be written especially for the mission at hand. However, whenever possible, certain key thoughts should be used to keep costs (and risks) at a minimum. They are[1]

8.0 Provide large margins
9.0 Use commercial tools whenever possible
10.0 Accommodate singularities and change
11.0 Provide large margins
12.0 Reuse existing software
13.0 Use good software engineering practices

We note that traditionally 50%+ of software costs go into software maintenance and upgrades. This will be especially true on this interplanetary mission where new software code will be uploaded to the spacecraft.

[1] Taken from Larson & Wertz, *Reducing Space Mission Cost*, p.95

500 Performance Assurance/Safety

These costs have been rolled into the Management costs, are so are counted as 0.

600 Assembly Test and Pre-Launch Operations

Test dictated by the launch vehicle provider.

700 Calibration

Both of these costs have been rolled into System engineering for accounting simplicity, so the effective cost are zero. The payload will have to be calibrated.

800 Launch Vehicle

The launch vehicle costs are being provided for by NASA. There will be some costs incurred here for the transportation of the spacecraft out to the launch site and on-pad changes, but they are estimated at .5 million cost total.

11.0 Management Plan

11.1 Management Philosophy

The management philosophy is tied in with the management plan. The plan has three main objectives:

- Deliver a product that meets the requirements of the specification
- Deliver such product in the delivery schedule
- Meet the cost objectives

To do this a philosophy of management was adopted with both innovative business and management strategies.

Table 11.1 Management Strategies

Strategy	Comments
Keep it simple	Late in the program, costs will mount up, so save pennies where you can. In the end, only so much money can be saved by pinching on certain costs.
Schedule Aggressively	However, have plenty of contingency time. Offer a fast schedule to the team with a possible reward – and a slightly slower schedule to the client.
Responsibility	Using small team architecture, each engineer has plenty of responsibility and will generally rise to the challenge, Make sure everyone has real authority in purchasing, design, etc.
Communication	With only eight engineers, communication is most important. As the project manager was also an engineer, constant communication was there.
Documentation	Once the basic design was agreed on, some documentation was asked for. However, it was left to a minimum (for example, meeting notes are written up and posted on the FTP site as well as e-mailed to group)
Frequent Reviews	BI-weekly meetings and constant e-mail helped the flow and allowed all engineers to participate in decision making and the general design processes. An FTP site for placing documents relevant to all members has been set up.

Subsystem engineers will work directly with the contractor and act as liaison between the Project management and the manager.

11.2 SCHEDULE

General Schedule:

Phase A - Preliminary analysis

- Subsystem trade studies
- Analysis of perform RQMT
- Identification of advanced technology/long lead items
- Risk assessment
- Cost estimation.

Phase B - Definition

- Revalidation of mission requirements
- Final risk assessment
- More specific trade studies
- Begin initial prototyping
- CDR review

Phase C/D - Design and Development

- Negotiate schedule with contractors
- Spaceflight systems designed
- Integration/test/launch
- Ground system development
- Validate flight software
- Test readiness review
- Flight readiness review

Phase E - Operations

- Launch
- Validate subsystems
- Detect and correct anomalies
- Calibrate instruments

Table 11.2 Spacecraft Bus Chronology

PROGRAM MANAGEMENT SPACECRAFT BUS CHRONOLOGY

Events	Date	In Relation to Phase A Start	In Relation to Launch
Phase A - Preliminary Analysis			
Receipt of The Solar Irradiance Monitoring Mission at Mars PDR	15-Feb-00	A-1wk	E-107wk
Phase A Study Start, Sunstroke Aerospace formed	20-Feb-00	***A (Phase A Start)***	E-106wk
Survey Architecture options (launch vehicle, orbit, spacecraft design)	1-Mar-00	A+1wk	E-105wk
Select final requirements, begin prelim. Design	14-Mar-00	A+3wk	E-103wk
Complete Prelim design and cost	28-Mar-00	A+5wk	E-101wk
PHASE B - Definition			
Final Risk Assessment	1-May-00	A+10wk	E-97wk
Specific trade studies	2-May-00	A+10wk	E-97wk
CDR Review	4-May-00	A+11wk	E-96wk
Begin initial prototyping	30-May-00	A+15wk	E-92wk
Begin ordering long lead parts	30-May-00	A+15wk	E-92wk
Spaceflight systems final design	15-Aug-00	A+23wk	E-82wk
Complete detailed design and cost	15-Sep-00	A+27wk	E-78wk
Negotiate Schedule with contractors	30-Sep-00	A+29wk	E-76wk
Phase C/D -Design and Development			
Complete procurement of major subsystems	28-Apr-02	A+61wk	E-46wk
Complete fabrication	28-Apr-02	A+61wk	E-46wk
Complete integration and test	28-Jun-02	A+65wk	E-41wk
Test Readiness Review	28-Jul-02	A+69wk	E-37wk
Ground System Development	28-Sep-02	A+77wk	E-29wk
Validate flight Software	28-Dec-02	A+99wk	E-9wk
Flight Readiness Review	28-Dec-02	A+99wk	E-9wk
Phase E - Operations			
Launch	1-Mar-03	A+106wk	***E (Phase E Start)***
Validate Subsystems	8-Mar-03	A+107wk	E+1wk
Detect and Correct anomalies	10-May-03	A+108wk+	E+2wk
Calibrate Instruments	28-Sep-03	A+136wk	E+30wk

11.3 Method of Contracting

In general, a system of fixed price contract with incentive fees will be used. Manufacturing of subsystem components contracted out to best bidder with 90% paid at delivery, 10% paid after successful deployment in space. This is an excellent incentive system.

11.4 Techniques Employed to Hold cost and Schedule

A Chart of cost reduction techniques is shown below.

Table 11.3 Cost Reduction Table

Method	Mechanism	Comments
Trading on Requirements	Eliminates non-critical requirements; permits low-cost technology	Science may be de-scoped. Working with the science team is critical
Concurrent Engineering	Increases feedback between engineering and management	Can achieve optimal design
Design to Cost	Makes costs paramount	Rarely used
Schedule Compression	Reduces the overhead of the labor pool; forced program drives down cost	Can results in poor initial design; work vs. schedule must be reduced
Reducing the Cost of failure	Allows both ambitious goals and calculated risk to make major progress	Fear of failure can drive cost-spiral
Using Microprocessors	Minimizes weight; allows on-orbit programming	Single-event upsets; software development may spiral out of control
Large Margins	Reduces testing; cost of engineering, manufacturing	Allows the ability to "buy" the way out of a problem
Using non-space equipment	Takes advantage of existing designs	Not space qualified
Standard components and interface	Reduces cost and risk by reusing hardware	Standardization is standard in most manufacturing;

11.5 Brief Experience Description of Team Members

Organizational Structure

The management organizational structure of the investigation team can be seen below.

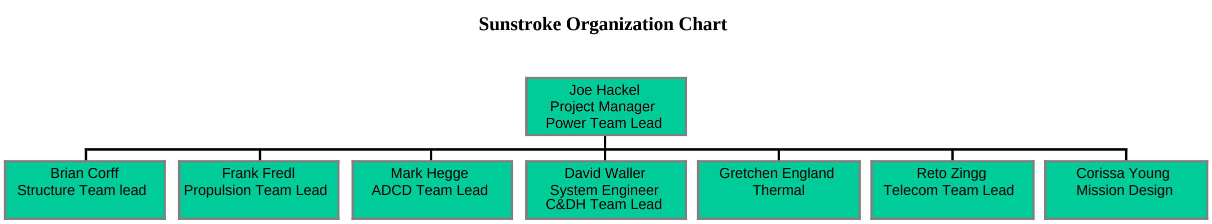


Figure 11.1 Sunstroke Organization Chart

A more complete description of the Team Leads can be found in Appendix A11

Project Manager: Joe Hackel
Experience; 2 years as high school teacher and 2 years as trainer in Singapore give him the international experience to manage a group of engineers. If he has the patience to deal with high school students and to train people in computers whose first language may not be English, he has the ability to lead a team of engineers. He is devoting long hours to ensure that his group of engineers remains focused. However, he is using a hands-off approach. He knows that he team is motivated, he just has to make sure not to demotivate them. He also relies on the System engineer to help nudge the team members along.

Power Team Lead: Joe Hackel
In addition, his work on the Citizen-Explorer satellite gives him the qualification to be the power team lead. Although he is not well versed in power elements, he is a quick study and has spent some time in a trade study of solar cells/power. He is spending long periods of time researching different contractors and vendors to find the best services for the lowest cost. He finds it easy to talk to various members of companies to ferret out information at this early design stage that they maybe otherwise reluctant to reveal.

System Engineer/C&DH Team Lead: David Waller
Experience includes 3 years of industry work as a spacecraft design engineer. In that time he has worked on interplanetary, DOD, and NASA scientific satellite systems. While primarily focused on mechanical design and integration, additional responsibilities also included interface documentation, testing procedures, and software writing. Because of familiarity with most of the S/C systems, the CDH role is a natural progression. This background will help speed the learning curve to produce a good CDH design.

ADCS Team Lead: Mark Hegge
Mark Hegge's role in the development of the Sunstroke proposal is primarily as the Attitude Control System (ACS) specialist. He is responsible for determining all ACS requirements for the mission and designing the most reliable and cost effective ACS that meets those requirements. Mark works closely with each of the other spacecraft subsystems and draws from his deep resources at Ball Aerospace. Mark has more than 5 years of design engineering experience including 3 years of space flight hardware design.

Thermal Team Lead: Gretchen England
Gretchen England's experience in thermal control is very limited. The only aspects of thermal systems that Gretchen has been involved in includes classes taken in thermodynamics and materials. The reason Gretchen chose to work on the thermal control section is to do some sort of design she has never done before. In spacecraft design, she already had experience with mission design, structures, and propulsion.

Mission Design Team Lead: Corissa Young
Corissa Young has a multitude of experience in the area of mission design. She has assisted in the development of the orbital parameters for a spacecraft currently being built

by the Colorado Space Grant College. Currently, she is performing the Orbital Debris Assessment of this small satellite for NASA. Ms. Young has also worked with Satellite Tool Kit (STK) software to aid in the mission design of other Earth orbiting satellites.

Structure Team Lead: Brian Corff
Brian Corff has a B.S. degree in Metallurgical and Materials Engineering from the Colorado School of Mines and is completing his Masters in Mechanical Engineering at the University of Colorado. He in experienced in machine and mechanism design through various class work and projects and through hands-on experience at the Idaho National Engineering Laboratory and TDA Research. He also has a strong background in computer-aided design and drafting. His experience with materials science and engineering allows him to correctly analyze and understand tradeoff decisions. Previous design projects include Formula race cars and materials development for nuclear waste handling and spacecraft applications.

Propulsion Team Lead: Frank Friedl
As propulsion subsystem designer, I was required to analyze performance, weight requirements, and helped aid in the mission design. My background includes courses on propulsion and energy systems. Additionally, I have performed combustion research with the Coal Utilization Group at the University of Wyoming. I have also been involved in many design projects and am currently working on the design of a hypersonic strike fighter. I therefore have extended team experience and work very professionally in groups.

Telecommunication Team Lead: Reto Zingg
An education in electrical engineering and specialization on high frequency electronics, antennas and wave propagation give him the ability to keep the oversight over the telecommunication system without losing contact to the details. Experience also in software development for digital wireless communication broadens his expertise in this field.

12.0 Outreach Plan

12.1 K-12 Outreach Approach

The goal of the Sunstroke spacecraft team is not only to build, launch, and successfully operate a satellite, but also to promote the excitement of space exploration in elementary school, middle school, and high school classrooms. Space is a subject area that captivates people of all ages, especially children. It is the perfect medium for fostering interest and advancement of skills in math, science, and technology in K-12 students. These areas of study are often viewed as difficult and useless in everyday life, which does not encourage many students to pursue higher educational levels of math, science, and technology. Math and science teachers around the globe are always answering the following question: "When am I ever going to use this in the real world?" with "this" usually referring to geometry, trigonometry, biology, chemistry, or any other concept that students struggle with. The Sunstroke spacecraft team hopes to use space exploration as the answer to the question uttered by frustrated, discourage students of all ages all over the world!

Ultra Violet (UV) radiation exposure has become an increasingly important public concern over the past decade due to increasing skin cancer rates and decreasing ozone coverage over the South and North Poles. The only way to combat the hole in the ozone is through education of the general public and more importantly school-aged children. One of the most effective ways to educate children and adults is through hands-on activities. Creating an interactive learning environment can enhance retention of material as well as the level of understanding of the participants. The best part of this interactive, hands-on environment is that is makes learning fun! The Sunstroke spacecraft team is using the above issues (UV radiation) and ideas (hands-on learning) to create a dynamite program for K-12 students across the United States and world.

There are essentially four parts to the K-12 outreach program:

- First is a series of classroom activities that involve studying UV radiation, satellite operations and construction, and space travel.
- Next, teacher training workshops will be held to help teachers understand all of the concepts associated with the project.
- The third phase of the project involves Sunstroke Corporation employees visiting classrooms and giving tours of the Sunstroke spacecraft facilities.
- Finally, all of the participating students will have the option to be part of either a paper symposium (middle and high school students) or a poster symposium (elementary school students) to be held each year in different locations throughout the United States.

12.1.1 Classroom Activities

UV Radiation Study:
As was mentioned previously, UV radiation has become a public concern issue during the last decade. The satellite being built by the Sunstroke team will be taking measurements of the amount of UV radiation reaching Mars. Students will be taking measurements of the amount of UV radiation reaching their school using a simple UV meter. A company entitled SunSor builds simple meters that give a unitless measurement of UV radiation, which are very easy to use and only cost $40 each. The UV meter is shown in Figure 12.1 below.

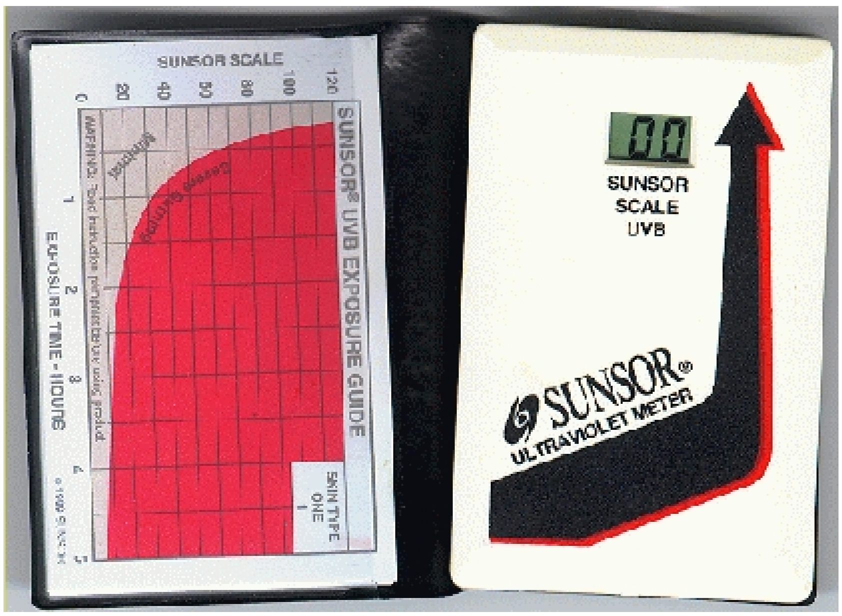


Figure 12.1**: UV meter schematic**

Students will be collecting data from Earth and will be receiving data from Mars via the Sunstroke spacecraft website. Thus, students will be actively collecting and comparing UV radiation data from multiple sources. This data collection can initiate other studies in the following areas

- Composition of Earth and Mars atmosphere
- Differences in the impacts of UV radiation on Earth and Mars
- Variation of UV radiation on Earth and Mars
- Distance between Earth and Mars
- Composition of space
- Factors affecting UV radiation (i.e. Sun, time of year, time of day, etc.)

Spacecraft Study:
There will also be classroom activities focusing on different components of a spacecraft. Many simple activities have already been developed to demonstrate different spacecraft concepts. Students will be able to follow the process used to build and operate a satellite via the Sunstroke spacecraft website, which will include:

- Pictures of construction and testing of the Sunstroke spacecraft
- Simple explanations of spacecraft concepts
- Schedule of events
- Pages to print out and color
- Multi-media games and activities

Orbital Mechanics Study:
The final portion of the classroom activities will focus on orbital mechanics. This will allow students to practice both math and visualization skills. Orbital mechanics can be a very complicated subject and will require lots of explanation and pictures on the webpage. Live rocket and shuttle launches will be both advertised and displayed live on our website. Satellite Tool Kit (STK) software will be used to illustrate orbital mechanics on the website as well.

12.1.2 Teacher Workshops:

Periodically the Sunstroke spacecraft team will hold teacher-training workshops at the Sunstroke spacecraft facilities. These workshops will be designed to provide teachers with enough information to initiate the Sunstroke spacecraft project in their classrooms! The topic areas that the Sunstroke team will cover during the two-day workshops are:

- Overall program organization
- Detailed information on the satellite that we are building
- Background scientific information on Ozone, Ultra-Violet Radiation, and Atmospheres (both Earth and Mars)
- How to use all of the different instrumentation involved in this project including:
  - -UV meters
  - -Webpage
- Tour of the Sunstroke spacecraft facilities
- Exposure to different curriculum developed for the project
- Mapping of the Sunstroke Spacecraft Project to the state standards for education

12.1.3 Guest Lectures and Tours

Each member of the Sunstroke team will be required to participate in one outreach activity every year that the satellite project is in operation. These activities will include visiting local schools, giving tours of the Sunstroke spacecraft facilities, speaking at community organization meetings, and helping with the Sunstroke Symposiums. An education team will be in charge of coordinating all outreach activities with schools, community organizations, and employees of Sunstroke.

12.1.4 Symposiums

High school and middle school students will have the opportunity to share their research projects and results at an annual paper symposium. The subject area will be limited to any type of space research project. Students will have to submit a paper and present to a panel of judges, which will be comprised of Sunstroke spacecraft team members. This will be a multi-day event including keynote speakers, tours of local engineering companies, and group competitions. A symposium will be held every year that the Sunstroke satellite is in operation (approximately 3 years), and each year the location will be changed to allow students from across the United States to attend.

Elementary students will have a poster symposium that is completely separate from the high/middle school symposium. This will also be a multi-day event that includes guest speakers and other activities. As with the high/middle school symposium, this symposium will also be sponsored every year that the Sunstroke satellite is in operation.

12.2 Public Outreach Approach

The best and most effective way to reach the biggest sector of the public is through the media. Therefore, the Sunstroke spacecraft team plans to comprise several press releases detailing the satellite and K-12 outreach programs. Both programs will also be actively promoted in the political arenas at the state and national level, gaining support and media coverage. These activities combined with the K-12 outreach program will reach a multitude of people in countries across the world, which will increase the amount of participation in all outreach activities.