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1.0 PROGRAM MANAGEMENT

1.1 Background

The Solar TErrestrial RElations Observatory (STEREO) Program is part of NASA's Sun-Earth Connection Program. The program aims to improve our understanding of the origins of solar variability and its effect on Earth climate and weather. STEREO is the third of five Solar-Terrestrial Probes called for under NASA's Space Science Enterprise Strategic Plan. STEREO's objective is to:

> *"understand the origin and consequences of Coronal Mass Ejections (CMEs). CMEs are the most energetic eruptions on the Sun. They are responsible for essentially all of the largest solar energetic particle events and are the primary cause of major geomagnetic storms"*[1]

In order to address this objective and answer questions concerning the origin and propagation of CMEs, the Science Definition Team (SDT) has proposed the launch of two instrumented spacecraft, both in heliocentric orbit. One spacecraft would lag the Earth in its orbit the other would lead. These vantage points would allow three-dimensional imaging of solar activity, CME generation and propagation.

The Johns Hopkins University Applied Physics Laboratory (hereafter referred to as APL) has been funded in Pre-Phase A to complete a conceptual design of the spacecraft and mission that would meet the objective outlined above. This report is was initially intended to coincide with the completion of Pre-Phase A and entry into Phase A. However, due to an extension of Pre-Phase A, this report will provide a summary of the conceptual design effort completed to date. Areas that have not been addressed adequately are carried under the risk section and will be focused upon during the remainder of Pre-Phase A.

1.2 Roles and Responsibilities

Goddard Space Flight Center (GSFC) is the NASA Center responsible for implementation of the STEREO mission. It is intended that APL and GSFC will work as partners in implementing this mission. This partnering relationship is expected to extend to all levels of the STEREO program. In order to foster communication, it is expected that GSFC will be apprised of APL's status both formally and informally, on a regular basis. This includes regular attendance of APL team meetings as well as the more formal reviews. It is also expected that APL Mission Operations personnel will routinely meet with GSFC Science Operations personnel so as to develop a ground system that supports the mission requirements. Table 1-1 shows the partitioning of roles and responsibilities between APL and Goddard Space Flight Center.

1.3 Report Overview

This report is broken into six sections plus appendices. The goal is to provide a top-down overview of APL's effort including system and subsystem engineering, mission design, integration and test and mission operations. The report addresses requirements, implementation and identifies areas that need particular attention at the system and subsystem level. The appendices are used to provide additional documentation as well as governing documents that are germane to the program.

This document is intended to provide a snapshot in time and is not a final report. The design will continue to be iterated and will not be finalized until the Critical Design Review.

1.4 Cost and Schedule

Cost and schedule information is provided under a separate cover.

[1] The Sun and Heliosphere in Three Dimensions, Report of the NASA Science Definition Team for the STEREO Mission, 1 December 1997.

Table 1-1 Partnering Approach

(GSFC Draft STEREO Mission Requirements Document)

Function	APL	GSFC
Program Mission Manager		Lead
Project Mission Manager	Lead	
Project Scientist	Assist	Lead
System Engineer	Lead	Assist
Spacecraft Engineering	Lead	
Subsystem Engineering	Spacecraft Lead	Instrument Lead
Integration Engineering	System Lead	
Ground System Engineering	Lead	Assist
Science Operations	Assist	Lead
Mission Operations	Lead	Assist
Launch Vehicle Acquisition	Assist	Lead

2.0 MISSION DESIGN

2.1 Introduction

The goal of the Pre-Phase A mission design analysis effort was to identify the science and system design parameters that drive orbit selection and to develop a preliminary mission design. Previous work on the mission design had identified the launch energy envelope required to achieve the desired range of the spacecraft's relative drift rate to Earth. The proposed method for achieving the desired orbit was a direct insertion into heliocentric orbit [Reference 1]. One of the results of this work was the selection by the Science Definition Team of a two spacecraft formation. One spacecraft is placed into a heliocentric orbit that leads the Earth, while the second spacecraft follows behind the Earth in its orbit. Selection of the direct transfer mode identified a nominal launch energy, C3 = 1.0 km^2/sec^2. The scope of the current study is to identify the additional factors that impact the mission design. Factors considered for this study include single versus dual launch, launch window constraints, launch parameters and drift orbit selection.

2.2 Science Definition

The definition of the spacecraft orbit needed to fulfill the science objectives for the mission is derived from the recommendation of the Science Definition Team for one spacecraft to lead the Earth, with a second to follow the Earth with the following characteristics:

> "STEREO #1, leading Earth, will dwell near 20° between 200 and 400 days into the mission, and near 45° between 600 and 800 days. STEREO #2, lagging Earth, will dwell near 30° and 60°, respectively."

2.3 Solar Drift Orbit Mechanics

The heliocentric orbits selected by Solar TErrestrial RElations Observatory (STEREO) mission are well represented by the classical Keplerian orbital elements of Semi-major axis α, Eccentricity e, Inclination i, Right Ascension of the Ascending Node Ω, Argument of Perihelion ω, and True Anomaly ν, in the heliocentric reference frame. The science definition is primarily concerned with the semi-major axis because it directly determines the mean drift rate relative to Earth, and is the primary factor in determining the dwell time history.

A convenient mapping of the heliocentric orbit into the departure conditions from the Earth is provided by the Zero Sphere of Influence Patched Conic Model. In this model the departure condition is simply defined by the V_∞ vector as illustrated in Figure 2-1. V_∞ is the vector difference between the velocity of the spacecraft's heliocentric orbit and the velocity of the Earth. The magnitude of V_∞ is referred to as the hyperbolic excess speed. The equation $C3 \equiv |V_\infty|^2$ relates the constant C3, to the hyperbolic excess speed. The escape angle α is defined as the angle between the Earth's velocity direction and V_∞ as shown in the figure. Although the figure shows the vectors drawn in the Earth's orbit plane; selected pairs of V_∞ and α actually represent a locus of solutions which describe a cone of half-angle a around the Earth's velocity vector.

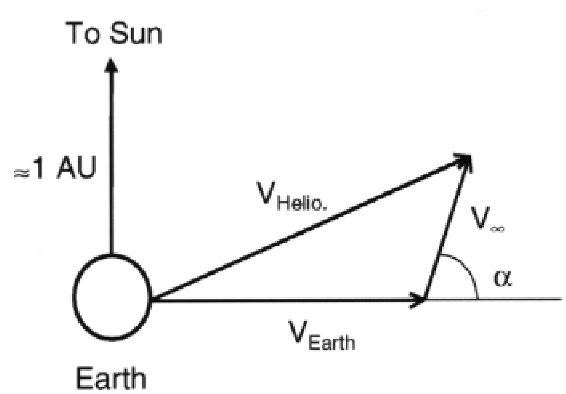


Figure 2-1 Earth Escape Parameters

Neglecting the small effect of the Earth's orbital eccentricity (0.017), it can be shown that the spacecraft's heliocentric semi-major axis, and therefore mean drift rate, is only a function of the spacecraft's heliocentric velocity. From this we are able to parameterize all heliocentric drift orbits by C3 and α. For the energies of interest to STEREO, (e.g., $C3 \approx 1.0\ km^2/sec^2$) the trajectory design space can be described by a contour plot showing the mean drift rate relative to Earth, η as a function of C3 and α as shown in Figure 2-2. A positive drift rate defines a leading orbit, negative a lagging orbit. In the scenario where the spacecraft is inserted directly into a heliocentric orbit by the launch vehicle, two important features can be identified from this mapping. First, for a selected mean drift rate, increasing the C3 value lessens the sensitivity of the drift rate to C3 variations. This is shown by the fact that contours of constant drift rate become nearly vertical as C3 increases. C3 variations are the major error source from the launch vehicle.

A second, somewhat contrary feature is that for a given mean drift rate, selecting the C3 value near the minimum C3 lessens the sensitivity of the drift rate to launch time. This is shown by the fact that contours of constant drift rate become nearly horizontal near the minimum C3. This assertion is made because we can assume that the rate of change of α is proportional to the Earth's rotation rate for most launch scenarios. The exception is a launch vehicle that can fly a variable azimuth trajectory as a function of launch time.

The mapping of C3 and α into the mean drift rate shown in the figure is idealized because we've used the Zero Sphere of Influence Patched Conic Model. This approach was selected to permit a closed form solution. The mapping also exists in more complex models that more fully

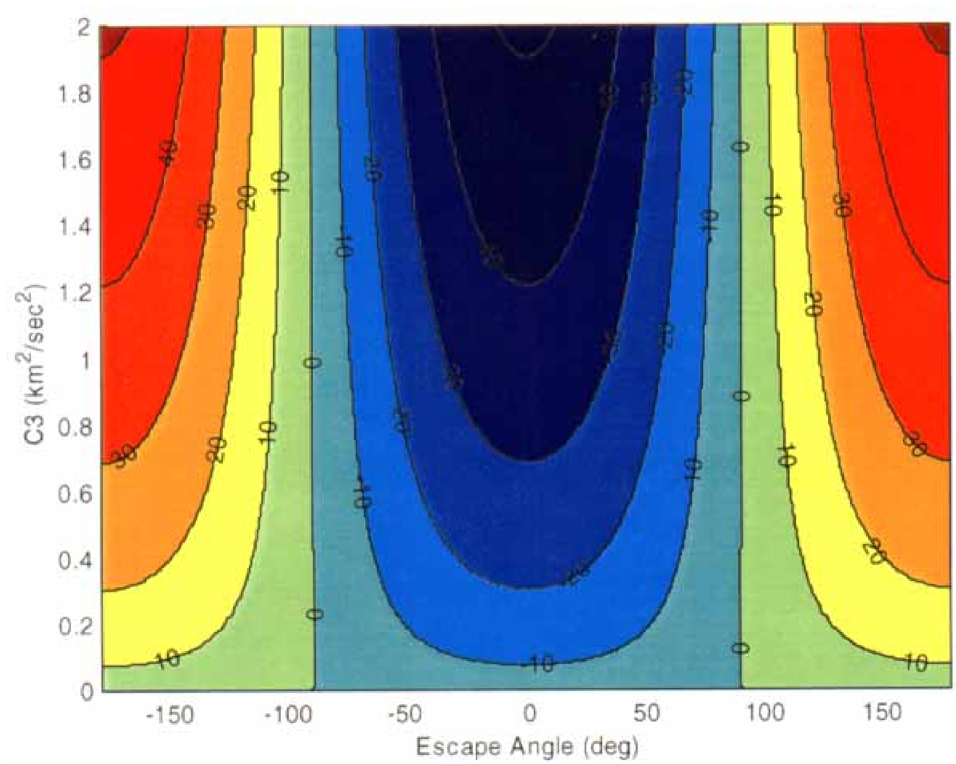


Figure 2-2 STEREO Mission Design Space

account for other perturbations. In these models the mapping retains it basic topology, but becomes increasingly distorted for low energy trajectories with escape angles approaching ±90°. The design point for most STEREO trajectories considered for this study, as well as the nominal design to be presented are far from this region.

All the trajectories for this study were computed in a complete solar system model that includes point mass gravitational effects of the Sun, Earth, Venus, Mars, Jupiter, and Saturn for the heliocentric phase. The heliocentric phase is defined as spacecraft to Earth distances > 900,000 km. For smaller distances, the gravitational effects of the Earth, Sun and Moon were used to model the spacecraft motion. The Earth was modeled with a 4×4 gravity field. The Sun and Moon are modeled as point masses. Perturbations due to solar radiation pressure were used in modeling both mission phases.

2.4 Families of Solar Drift Orbits

The primary interaction of the science definition and the orbital mechanics is the mean drift rate, which directly determines the semi-major axis. Figure 2-2 shows that there are actually families of drift orbits distinguishable by sets of the ordered pair (C3, α). One important family of trajectories are those that represent the minimum C3 (α = 0°, 180°) for a given drift rate. Ideally, these solutions correspond to maximum payload mass. Another trajectory family are those orbits with an $i = 0$. These planar solutions have their V_∞ in the ecliptic plane. For this report, the nominal mission trajectories were not restricted to any particular family of solutions, but utilized a full three-dimensional parameterization to best meet the science definition and optimize the system design.

Figure 2-3 and 2-4 show an ecliptic plane projection of a leading and lagging trajectory with a mean drift rate of +30 and −30°/year, respectively. A number of important parameters that impact the system design are derived from each trajectory. Figure 2-5 shows the spacecraft-Sun distance as a function of time (power, thermal). The spacecraft-Earth distance is shown in Figure 2-6 (telecommunications). Permutations of Sun, Probe, Earth angles are given in Figures 2-7 through 2-9 (telecommunications).

A principal concern of this study is the Sun-Probe-Earth (SPE) angle, which defines the antenna gimbal limits for the High Gain Antenna system for the spacecraft's nominal Sun pointing attitude. For the purposes of this study, the gimbal angle is equal to the SPE angle, where 0° corresponds to conjunction of the Sun and Earth as seen from the spacecraft. Figure 2-9 shows that for the leading trajectory the SPE angle is greater than 90° for about the first 200 days of the mission. The maximum value is approximately 165° after approximately 75 days. This is a general characteristic of leading trajectories. Both the maximum value and duration above 90° have been identified as major design drivers for the telecommunications system.

2.5 Transfer Trajectory and Launch Mode

Three types of transfer trajectories have been identified for possible use by STEREO to achieve the desired solar drift orbit. The choice of transfer trajectory is coupled to the possibility of launching the spacecraft independently (single launch) or together (dual launch). The three types of transfer trajectories are lunar flyby, libration point phasing, or direct insertion. The key ingredient to the lunar flyby trajectory is the use of a single or multiple flybys of the Moon in order to achieve the desired V_∞. In general, this type of trajectory requires the lowest C3, and therefore yields the highest payload mass of the three options. Spacecraft propulsion is required to accurately achieve the desired lunar flyby conditions. One or two months in a phasing

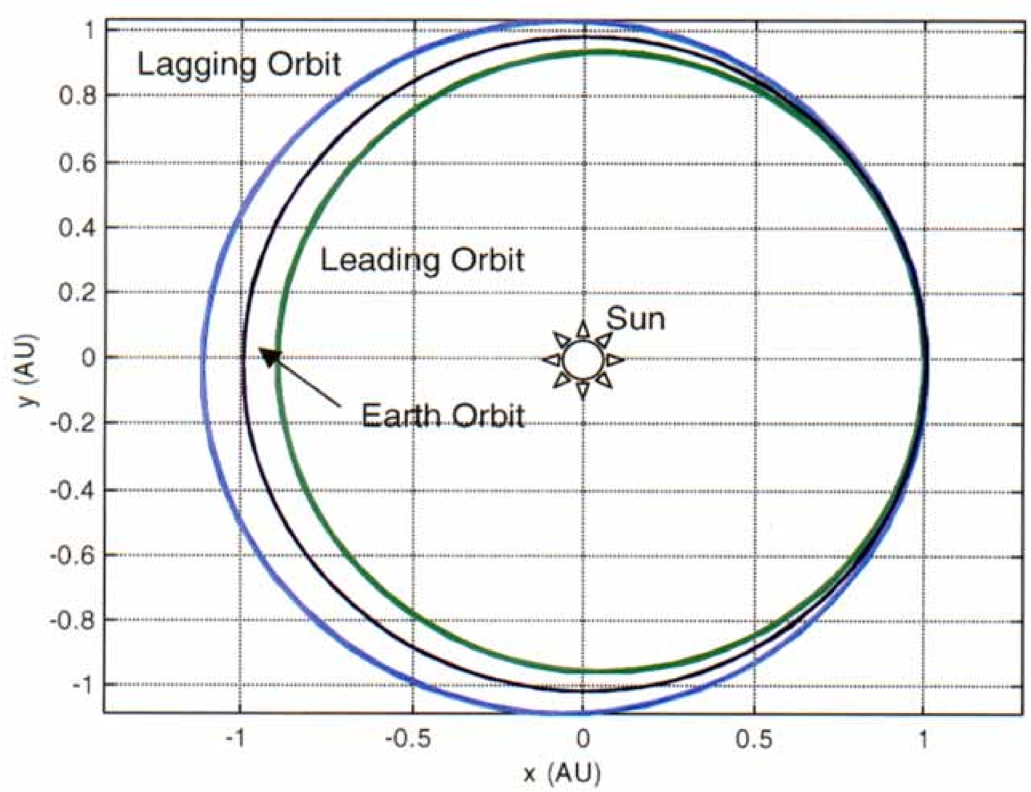


Figure 2-3 Sample Orbit, Heliocentric View

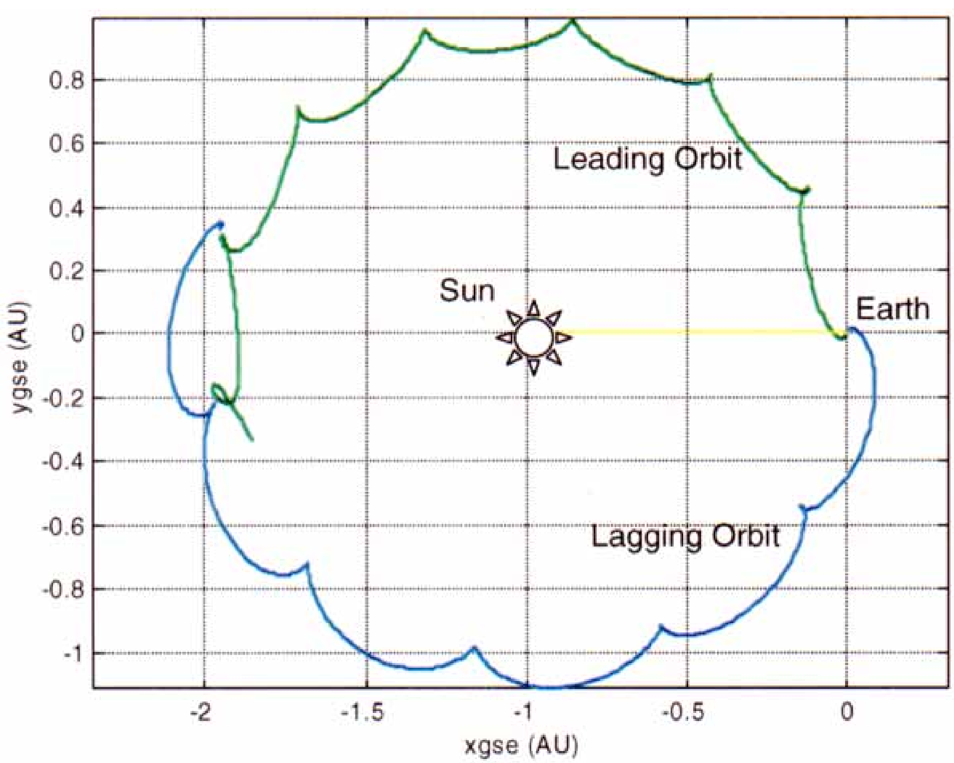


Figure 2-4 Sample Orbit, Geocentric Solar Ecliptic View

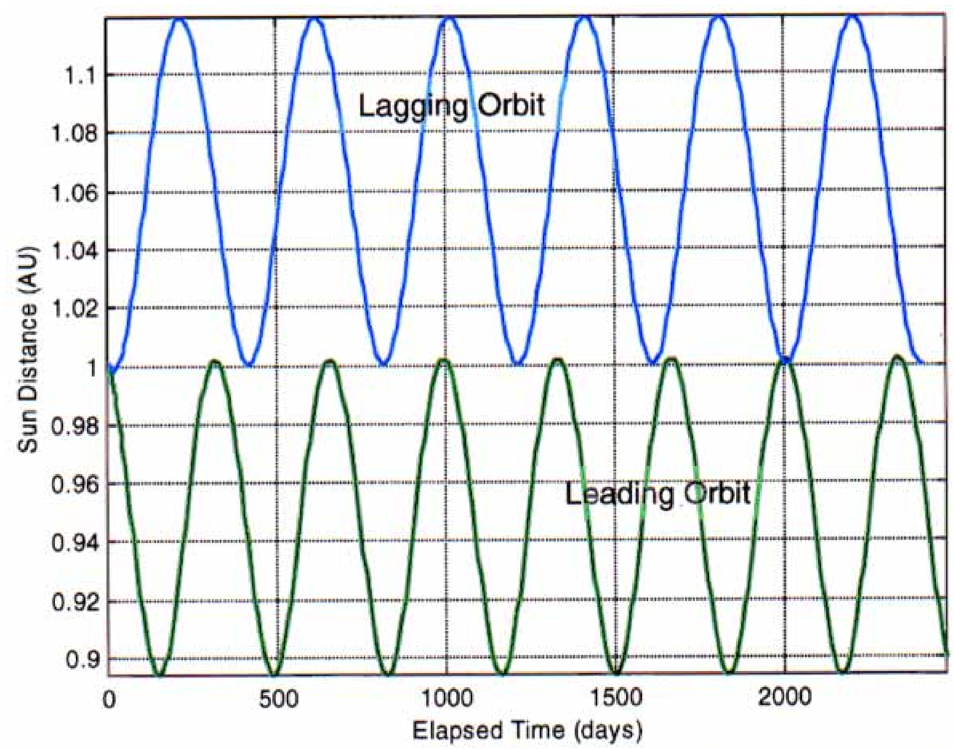


Figure 2-5 Sample Orbit, Sun Distance

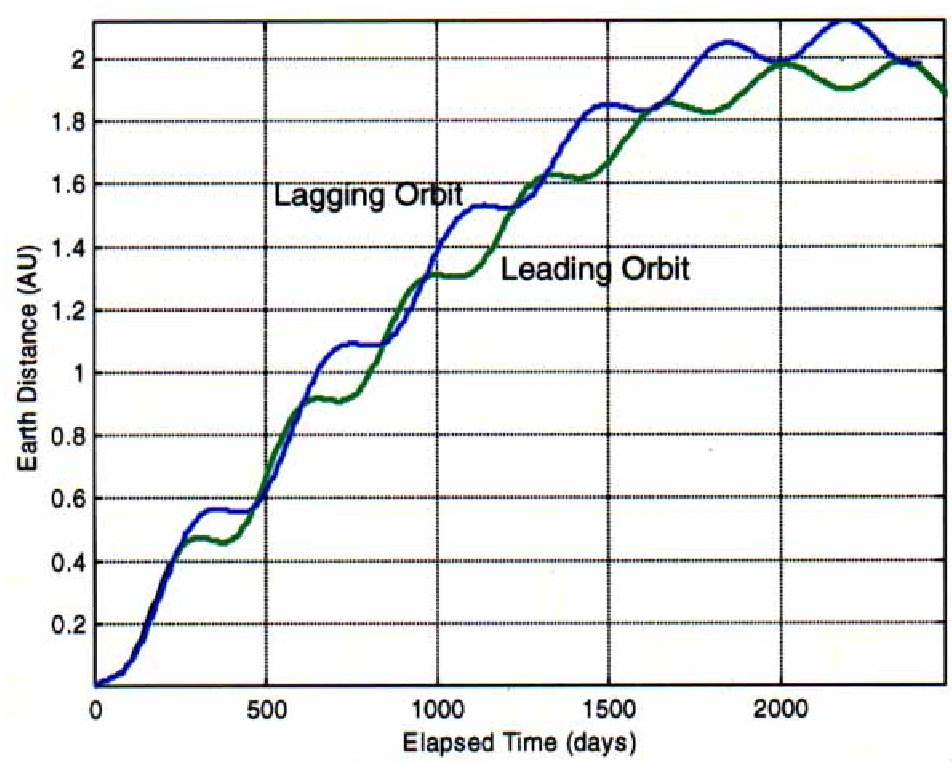


Figure 2-6 Sample Orbit, Earth Distance

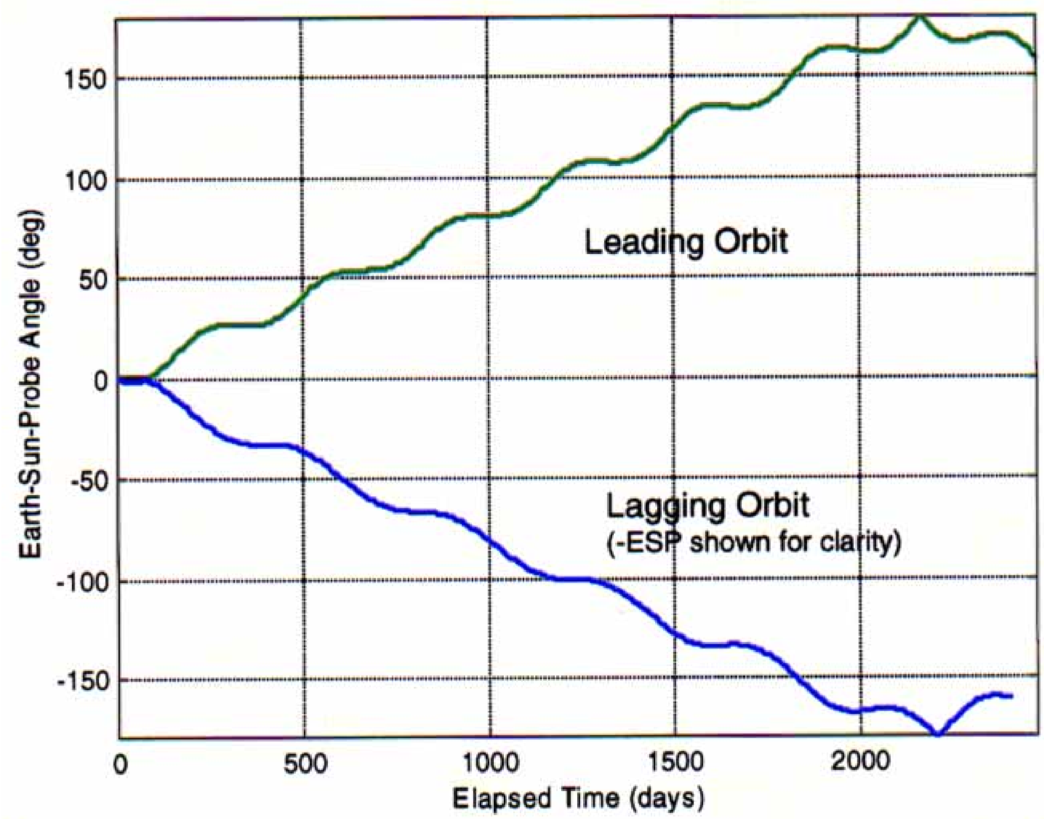


Figure 2-7 Sample Orbit, ESP Angle

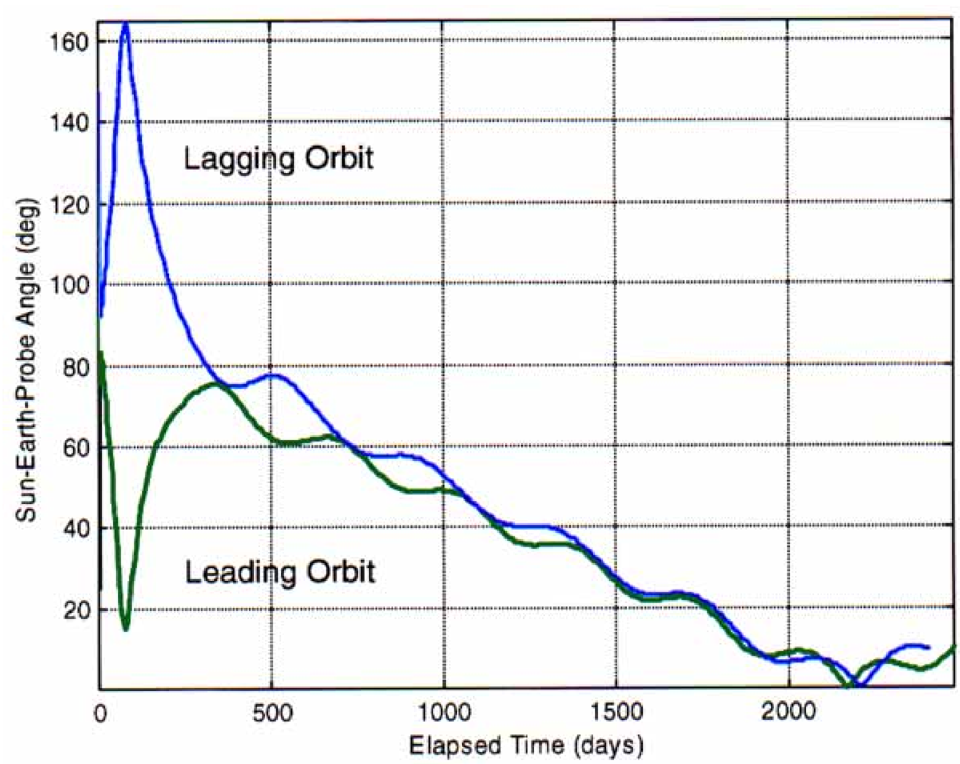


Figure 2-8 Sample Orbit, SEP Angle

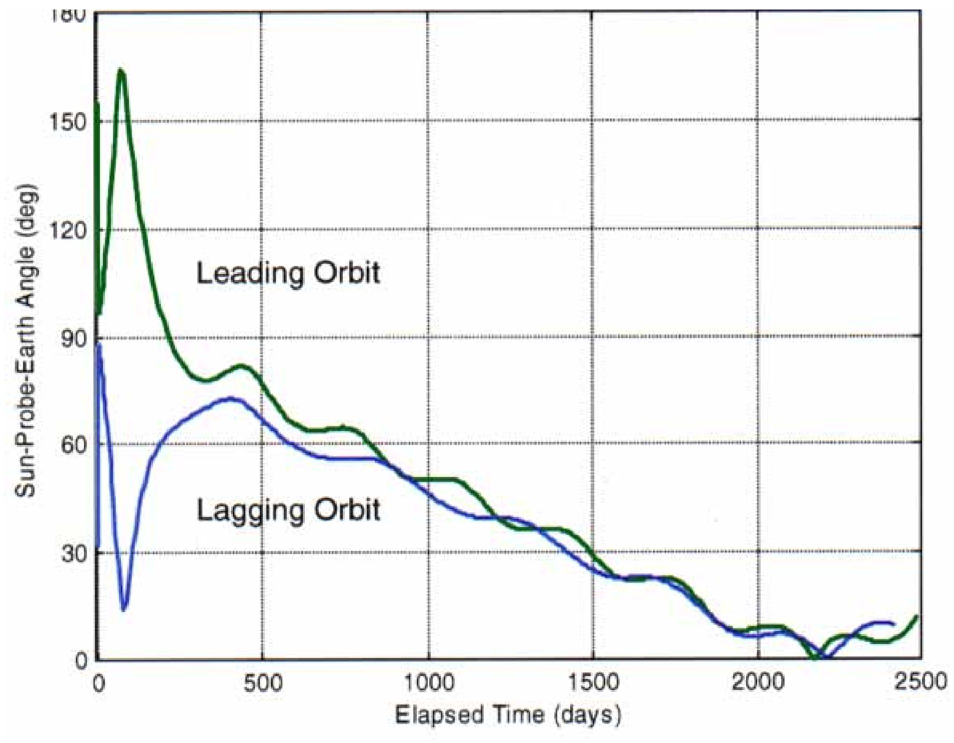


Figure 2-9 Sample orbit, SPE Angle

orbit prior to the first lunar flyby is desirable in order to decrease spacecraft fuel requirements and provide a reasonable number of launch opportunities per month.

The libration point transfers feature an excursion to either one or both of the interior or exterior libration points. The libration points are designated L_1 and L_2 for the interior and exterior points, respectively. They are located on the Sun-Earth line, approximately 1.5 million kilometers from Earth. Libration point transfers have an intermediate C3 requirement when compared to lunar flyby or direct insertion trajectories. The excursion to the libration point will typically consume more than seven months prior to Earth escape. Libration point transfers are well suited for a dual launch scenario where flexibility in selecting the final heliocentric orbit of each spacecraft independently is highly desirable. Spacecraft propulsion is required to control the orbit prior to escape from the Earth. Launch opportunities are generally available most days of the year, with a few days excluded each month due to undesirable lunar perturbations. Desirable lunar perturbations (i.e., lunar flybys) can be combined with libration point phasing to offer a high mass, high flexibility trajectory. The International Sun Earth Explorer (ISEE-3)/ International Cometary Explorer (ICE), Wind, Solar and Heliospheric Observatory (SOHO), and ACE missions have aptly demonstrated the utility of lunar flyby and libration points orbits. The Microwave Anisotrophy Probe (MAP) and Genesis missions have also selected these transfer types as their baseline mission design.

Direct insertion into heliocentric orbit offers the simplest approach to the STEREO mission design. This approach is well suited for the single launch mode. When the launch vehicle has sufficient lift mass, a dual launch can be accomplished. For a typical expendable launch vehicle (ELV) such as the Taurus, Athena-II, or Delta-II there is a significant impact to the flexibility of heliocentric orbit selection if a dual launch were used. A dual launch on the Space Shuttle offers more flexibility in orbit selection because of the extended mission duration (days vs. 1.5 hours). The preferred scenario for an ELV is the single launch mode. Up to one complete revolution in a low Earth parking orbit is required to fully exploit the trajectory design space shown in Figure 2-2. Although, if required by launch vehicle mass limits a direct ascent by the launch vehicle is a feasible means for achieving heliocentric orbit. No spacecraft propulsion is required for direct insertion transfers. This allows for a less complex spacecraft design and may negate the mass advantage of the lunar flyby and libration point trajectories. By definition, the heliocentric orbit is established once the spacecraft leaves the low Earth parking orbit less than two hours after launch. Launch opportunities are generally available most days of the year. A direct insertion approach was selected as the transfer trajectory for this study.

2.6 Launch Window

Two types of launch windows are defined for STEREO. The first, called the Launch Opportunity Window is defined as the days on which the spacecraft can be launched. The second, is the Daily Launch Window which is defined as the time of day you can launch. In order to discuss either type, we must first make the connection between the Earth escape parameters, represented by V_∞ and the launch vehicle and parking orbit parameters. A convenient parameterization of V_∞ is shown in Figure 2-10. The angles β and Δ are measured from the Earth's velocity direction. The angle β is a right-handed rotation around the Ecliptic pole and describes the offset in right ascension of V_∞ from the reference direction. For lagging trajectories the reference direction is along the Earth's velocity vector, for leading trajectories the reference direction is opposite the Earth's velocity direction. The angle Δ is the declination of V_∞ to the Ecliptic plane. The escape angle α, or it's supplement, is the hypotenuse of a spherical right triangle with sides β and Δ. The minimum energy trajectory for a selected mean drift rate corresponds to $\beta = \Delta = 0$. This particular parameterization allows for a straightforward mapping of V_∞ into the heliocentric orbital elements.

A second parameterization of V_∞ is the right ascension and declination of the vector relative the Earth's equator. These parameters, along

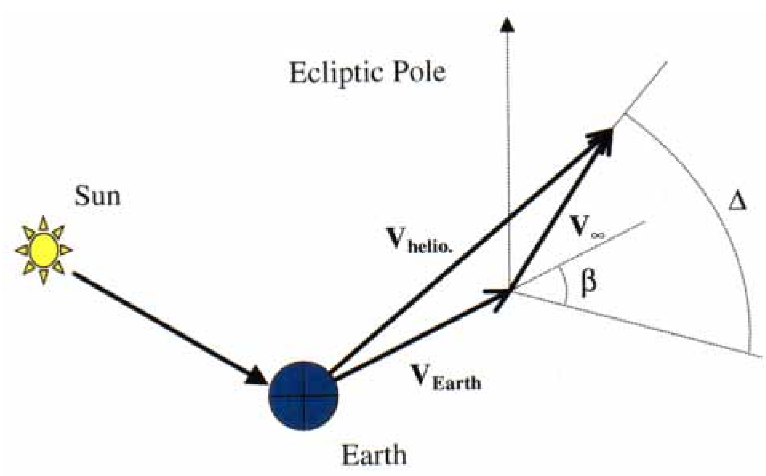


Figure 2-10 Design Parameterization

with the vector magnitude provide the geometric constraints needed to determine the launch time and parking orbit coast times for any given launch date. In general there are two launch opportunities per day, each with an associated parking orbit coast time. It is anticipated that only one of the two daily opportunities will be used. Since both opportunities result in the same heliocentric orbit, other factors such as station visibility, eclipse conditions, and launch time will be used as selection criteria between the two.

A significant constraint to any Earth escape mission is that the geocentric declination δ, of V_∞ can not exceed the parking orbit inclination for a planar injection. The launch site location and the launch azimuth determine the range of available parking orbit inclinations. Two factors impact the selection of launch azimuth. First, range safety limits the available azimuth range. Second, the maximum payload mass is achieved using a launch azimuth of 90° to take maximum advantage of the Earth's rotation.

2.6.1 Launch Opportunity Window

The minimization of the SPE angle becomes a factor in determining the Launch Opportunity Window (LOW) because of constraints imposed on the selection of Δ. The selection of a nonzero value of Δ results in an inclined heliocentric orbit. The spacecraft motion out of the ecliptic plane in an inclined orbit reduces the maximum SPE angle as shown in Figure 2-11. The value of the maximum SPE angle is related to all three of the design parameters (V_∞, β, Δ), but is most strongly dependent on Δ. In general, increasing Δ reduces the maximum value of the SPE angle for the leading spacecraft. Figure 2-12 shows the relationship between the time of year and maximum Δ. The figure shows the right ascension and declination of the Earth's velocity vector, which is our reference direction over the course of a year. The limiting case occurs at the equinoxes. Recall that the Earth's velocity is 90° out of phase with the Sun, so that the maximum declination occurs at the equinox rather than the solstice, while the zero crossings occur at the solstices. Also, recall that the maximum geocentric declination of V_∞ is a function of the parking orbit inclination, which in turn is determined by the launch azimuth and launch site latitude.

Two cases of interest to STEREO are an ELV or Shuttle launch from the Eastern Range. Assuming the ELV flies a maximum payload mass trajectory by launching due East (Launch Azimuth, AZ = 90°) the corresponding parking orbit inclination is 28.5°. Therefore, the maximum geocentric declination for V_∞ is also 28.5°. If the launch takes place near the equinox, the maximum Δ of 52°

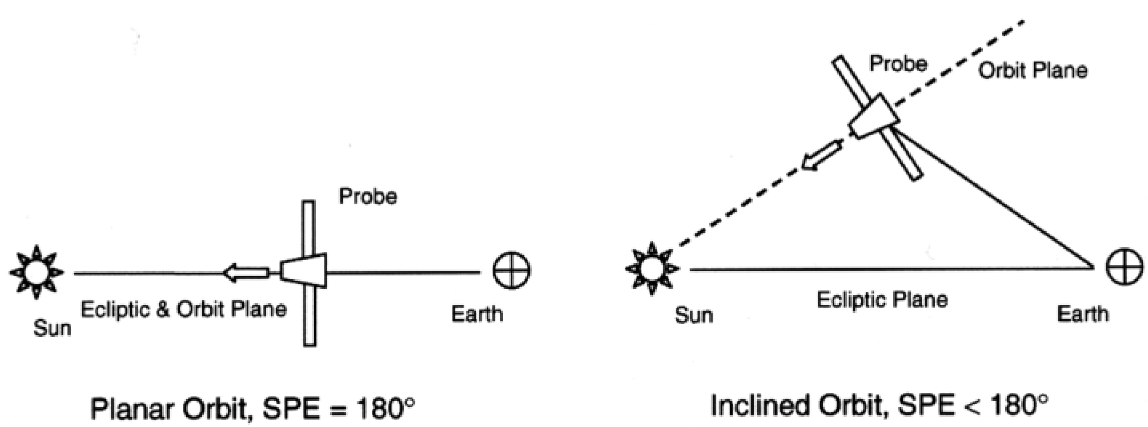


Figure 2-11 SPE Angle Minimization for Inclined Orbits

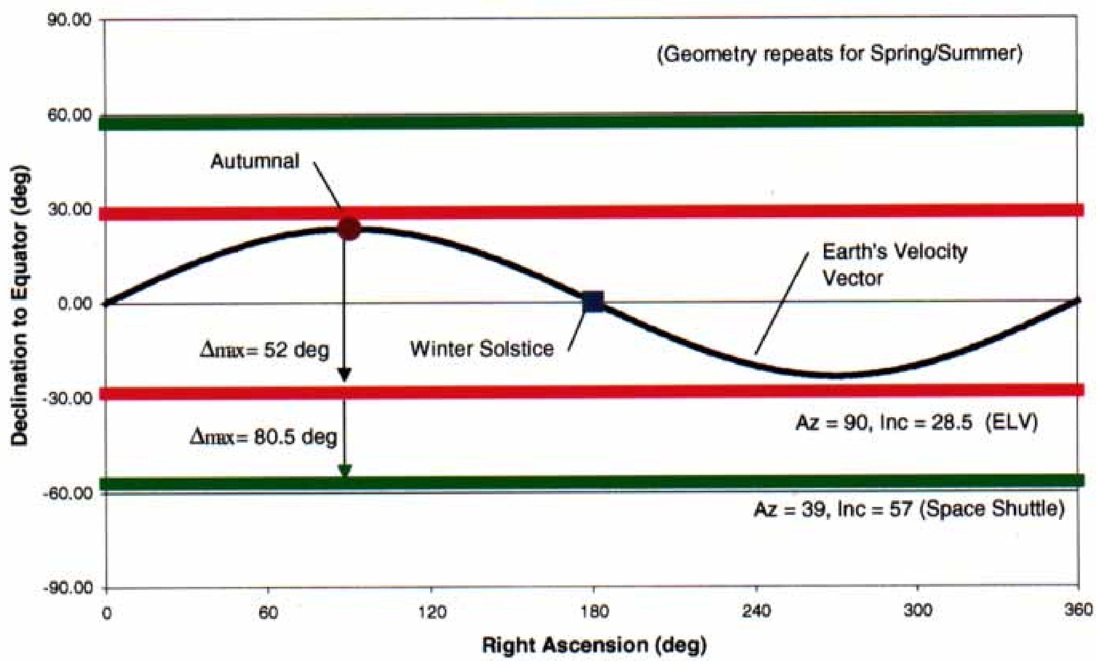


Figure 2-12 Launch Opportunity Window Trades

may be achieved. For a launch near the solstice, the maximum Δ is smaller. Since the higher Δ values are desired to reduce the maximum SPE angle for the leading spacecraft, launch near the equinox is preferred. If we consider a launch using the Space Shuttle, and assume no payload mass penalty for a high inclination parking orbit, the maximum value of Δ increases to 80.5°. Analysis performed for this study shows a 10° to 20° reduction in the maximum SPE angle for a Space Shuttle launch using a parking orbit inclination of 57°. A more detailed analysis of the impact of launch date on the SPE angle will be performed during Phase A.

2.6.2 Daily Launch Window

The primary drivers for determining the daily launch window are the early mission geometry with respect to the Sun and Earth, parking orbit coast time, and the sensitivity to launch time. As mentioned previously, there are typically two launch opportunities per day that achieves the same V_∞, and therefore the same heliocentric trajectory. The two launch opportunities have different launch times and parking orbit coast times. In some cases, the desired escape direction may require a full revolution in the spacecraft parking orbit. Long parking orbit coast times impact both the launch vehicle and spacecraft. The launch vehicle may require additional battery lifetime or expendables to coast for an entire orbit. The parking orbit coast time is a significant factor in the battery sizing for STEREO. Spacecraft power and thermal requirements may also impose additional launch vehicle requirements for attitude control during the coast phase. The parking orbit coast time also determines the post-heliocentric orbit insertion spacecraft-to-station geometry. The combination of launch and parking orbit coast times determines the spacecraft-Sun geometry, which determines the nature of spacecraft eclipse events.

The length of the daily launch window may be determined be examining the sensitivity of the heliocentric trajectory to the launch time.

Assuming the launch vehicle performs the same ascent trajectory over the daily launch window the primary effect on the trajectory is dispersion of the mean drift rate of the heliocentric orbit. A numerical estimate of this sensitivity can be obtained using Figure 2-2. For trajectories of interest to STEREO, ($C3 \cong 1$ km^2/sec^2, $\alpha \cong 50°$) the sensitivity in drift rate is approximately 1° per year for an 8-minute launch window, assuming the change in α is proportional to the Earth's rotation rate. This result has been verified for the nominal trajectory design presented below. The final determination of the daily launch window will be based on the launch vehicle error analysis performed for the STEREO configuration and the desired tolerance on the mean drift rate.

2.7 Nominal Trajectory Design

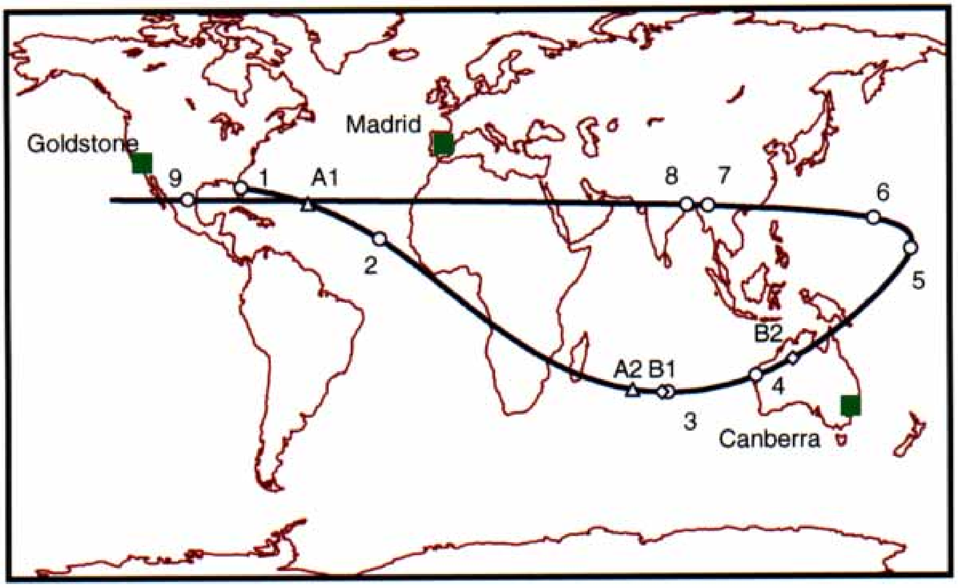
The science definition and mission design drivers that are cited above have all been considered in developing a nominal trajectory design for the STEREO mission. The mission design is based on two single spacecraft launches, sixty days apart aboard an Athena-II ELV from the Eastern Range. The spacecraft will be placed directly into heliocentric orbit from a low Earth parking orbit. A preliminary parking orbit definition was provided by Lockheed-Martin for the Athena-II based on a launch azimuth of 93°. The targeted mean drift rates are 20° and 28° per year for the leading and lagging spacecraft, respectively. No spacecraft propulsion is required to achieve or maintain the mission orbit. A nominal C3 of 1.0 km^2/sec^2 was selected for both spacecraft. This launch energy is greater than the minimum required for the mean drift rates in order to reduce the sensitivity to launch vehicle energy dispersions. The trajectory design parameters are given in Table 2-1. The leading spacecraft, which has a lower mean drift rate, is launched first in order to minimize the impact on the orbital formation of any delays in the launching of the second spacecraft. The launch date for the leading spacecraft is also closer to the equinox. For a given values of η and C3, α is fixed. For the leading trajectory, the values of β and Δ were selected to minimize the SPE angle during early mission and place the location of the maximum SPE angle as close to Earth as possible. Their values are subject to the constraints that α, β, and Δ form a spherical right triangle and geocentric declination of V_∞, $\delta \leq$ parking orbit inclination. The design parameters for the lagging trajectory were selected so that the launch phases for each spacecraft are identical. Figures 2-13 through 2-21 present the nominal mission design.

2.8 References

Reference 1: *The Sun and Heliosphere in Three Dimensions: Report of the NASA Science Definition Team for the STEREO Mission*, December 1997.

Table 2-1 - STEREO Mission Design Parameters

Parameter	Leading (STEREO-1)	Lagging (STEREO-2)
Launch Date	October 1, 2002	December 1, 2002
η (deg/yr)	20	−28
C3 (km2/sec2)	1.0	1.0
α (deg)	60	45
β (deg)	−41	30
Δ (deg)	49	35
δ (deg)	28	28



No.	Time	Event	No.	Time	Lighting Event
1	L + 0 min	Launch		*STEREO-1*	
2	L + 16 min	Park Orbit Insert.	A1	L + 10 min	Enter Eclipse
3	L + 49 min	Helio. Orbit Insert.	A2	L + 45 min	Exit Eclipse
4	L + 56 min	Canberra AOS			
5	L + 2.0 hr	Goldstone AOS		*STEREO-2*	
6	L + 4.3 hr	Goldstone LOS	B1	L+ 48 min	Enter Eclipse
7	L + 9.2 hr	Canberra LOS	B2	L + 60 min	Exit Eclipse
8	L + 9.9 hr	Madrid AOS			
9	L + 22.9 hr	Madrid LOS			

Figure 2-13 Early Mission Groundtrack

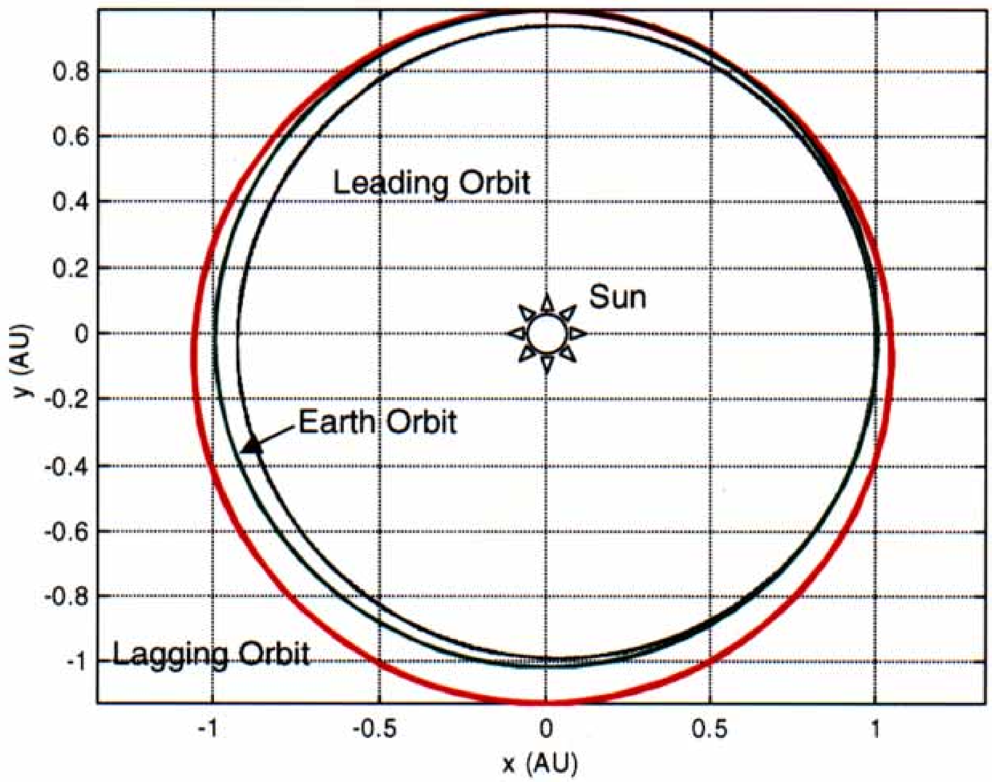


Figure 2-14 Nominal Heliocentric View

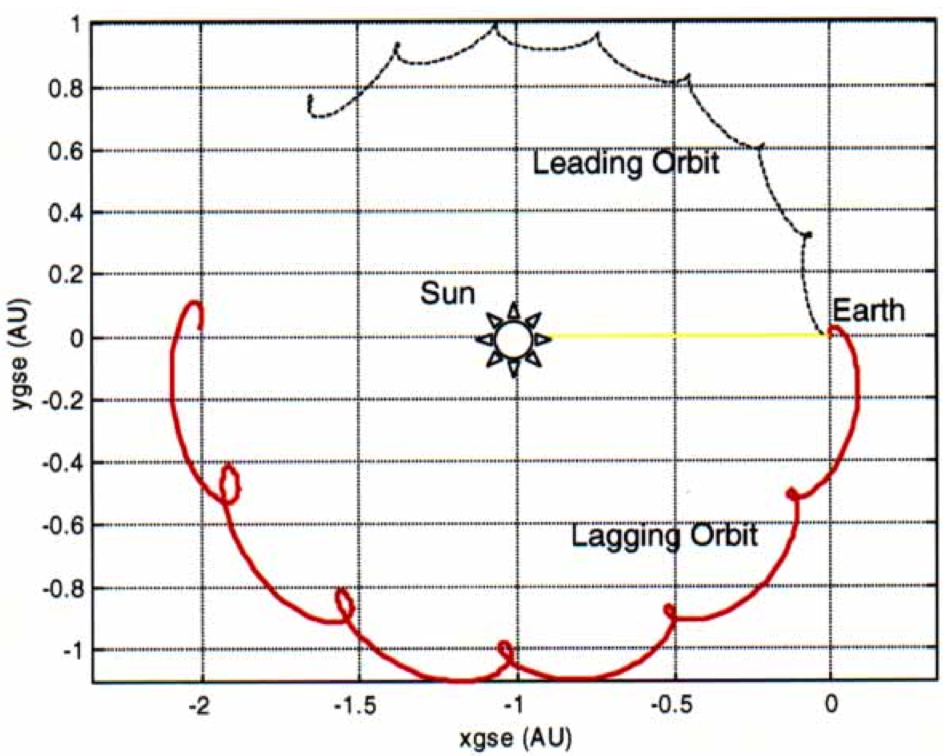


Figure 2-15 Nominal Geocentric Solar Ecliptic

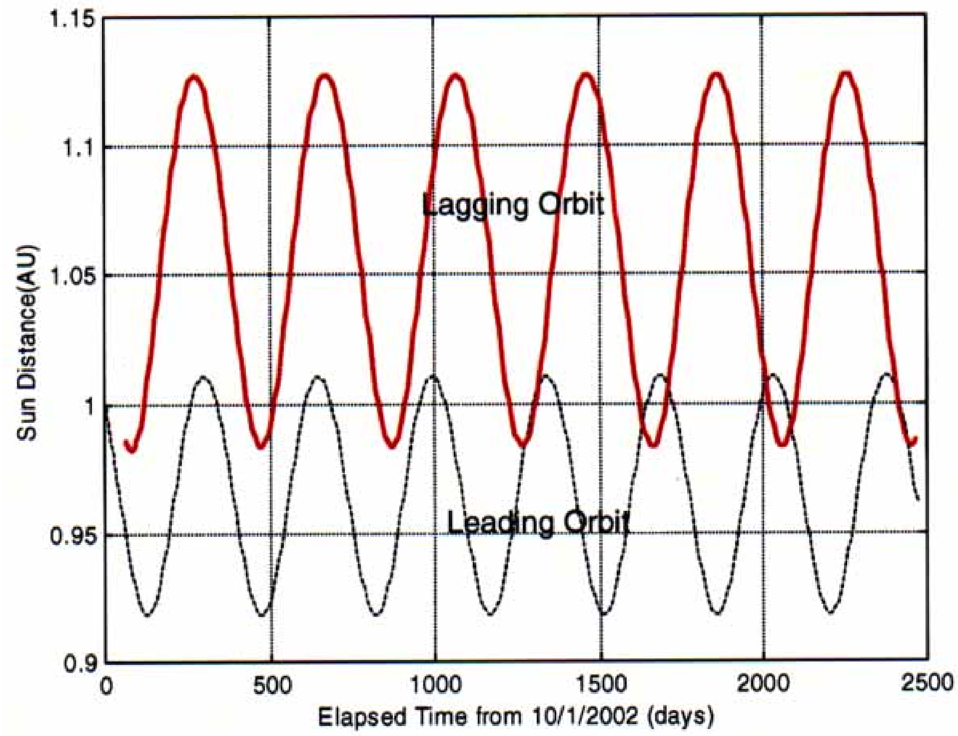


Figure 2-16 Sun Range

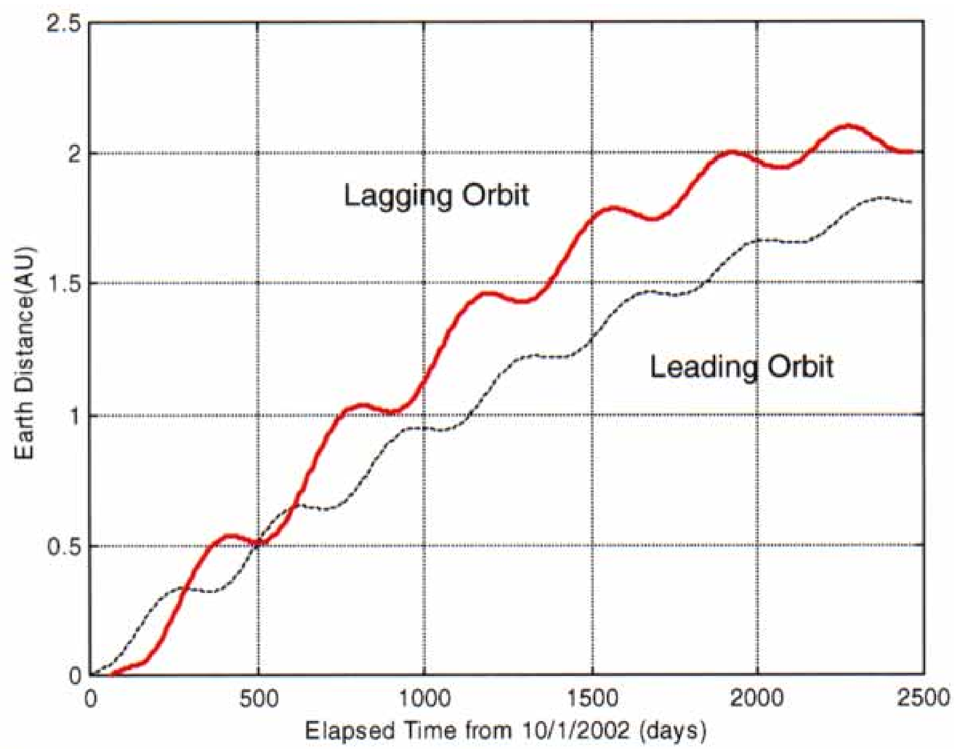


Figure 2-17 Earth Range

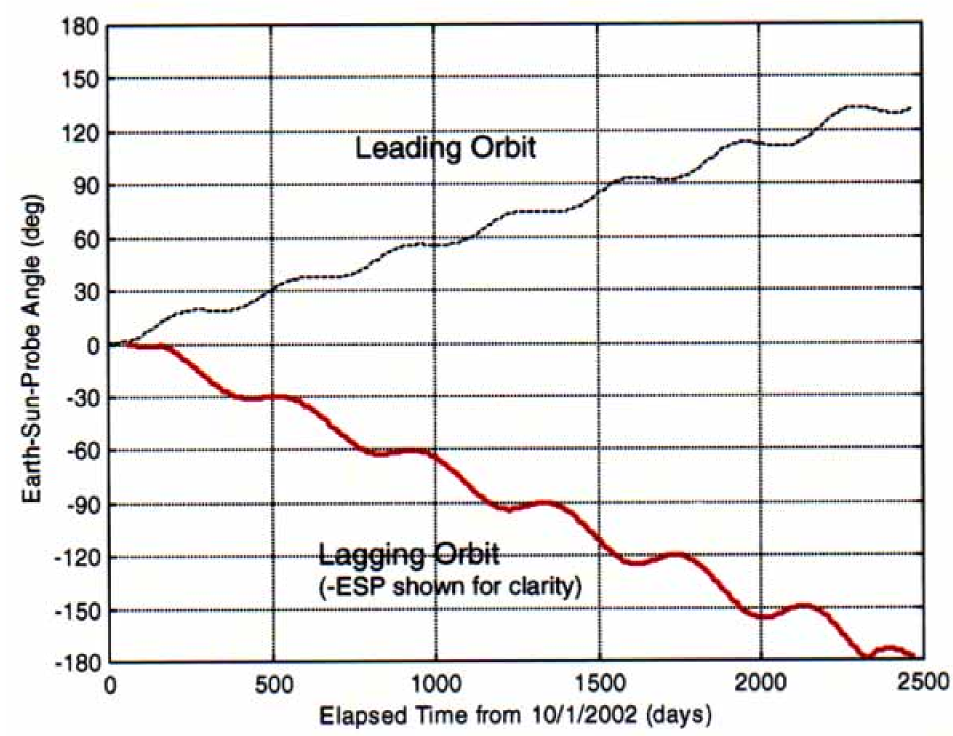


Figure 2-18 Earth-Sun-Probe Angle

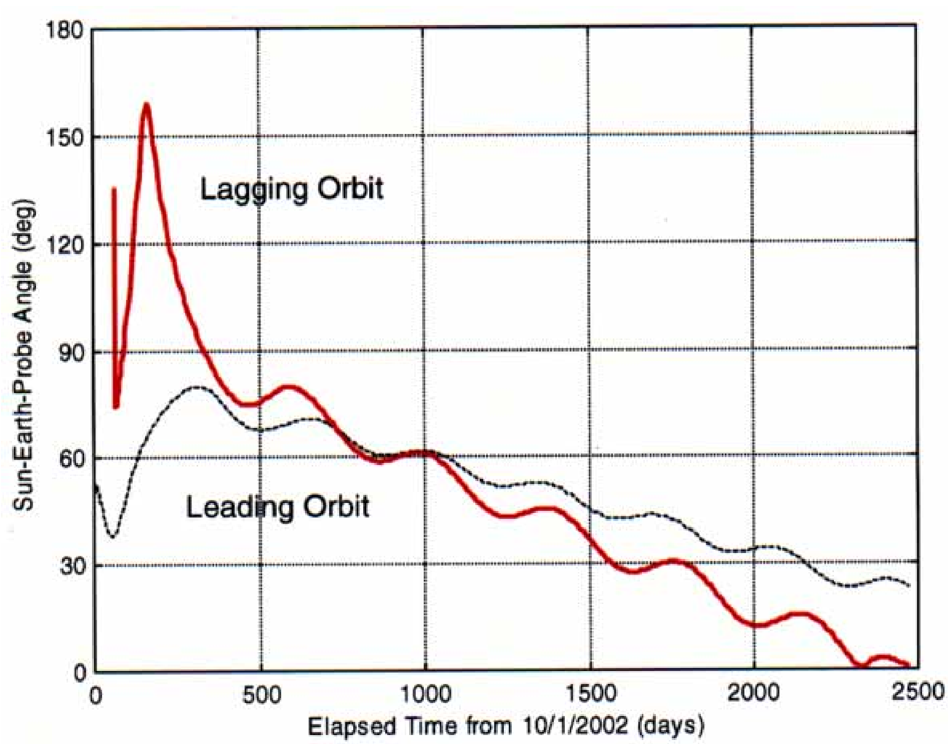


Figure 2-19 Sun-Earth-Probe Angle

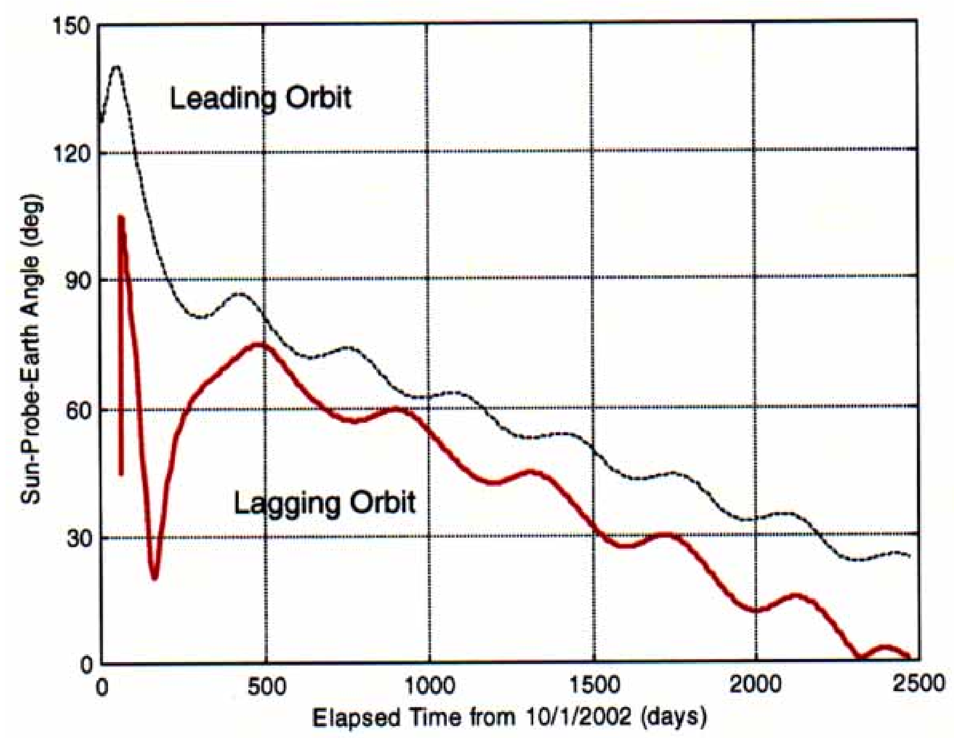


Figure 2-20 Sun-Probe-Earth Angle

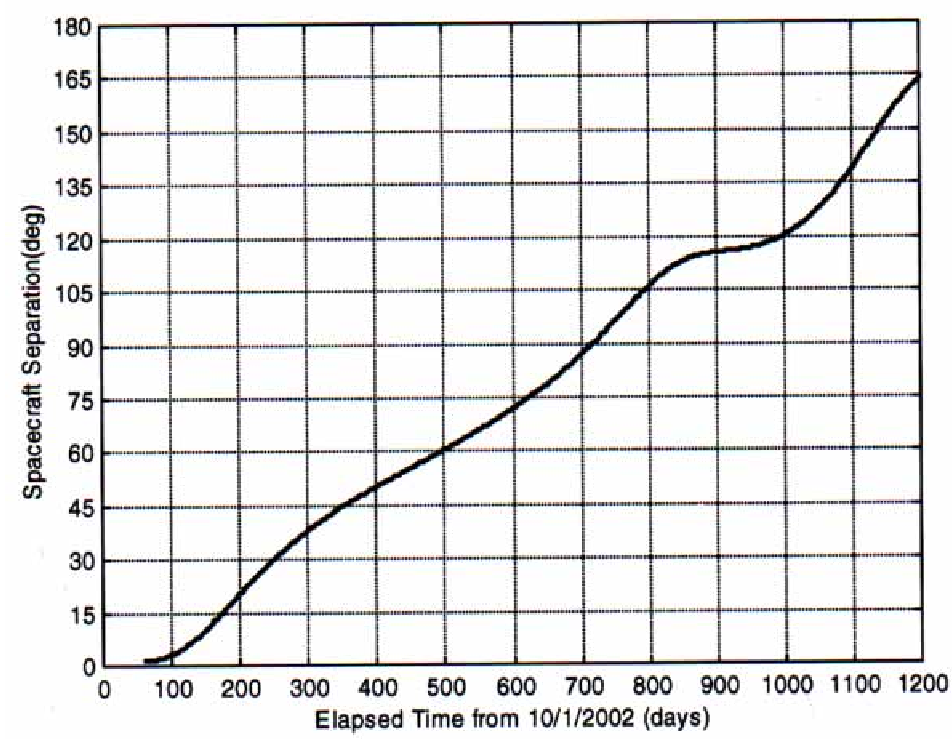


Figure 2-21 Angular Separation

3.0 SYSTEM ENGINEERING

3.1 System Description

The implementation described in this report is derived from requirements resulting from the mission described in the Solar TErrestrial RElations Observatory (STEREO) Science Definition Team (SDT) Report and the follow-on GSFC Report. Additionally, there has been close interaction between the APL spacecraft development team and scientists and instrument developers from GSFC and APL. The top-level requirements that have the greatest impact on the system are defined in Table 3-1.

The STEREO spacecraft is implemented as a three axis stabilized platform that takes advantage of the Thermosphere, Ionosphere, Mesosphere, Energetics, and Dynamics (TIMED) spacecraft architecture as well as specific TIMED designs. In order to reduce cost risk, the STEREO program plans to make use of a single string derivative of the TIMED spacecraft Command and Data Handling (C&DH) and Guidance and Control (G&C) processing architectures. Figure 3-1 shows a system level block diagram where designs that have significant (> 90%) legacy to past APL designs are shown in blue. Table 3-1 shows individual components, their legacy and the scope of the changes that are needed to meet STEREO requirements.

STEREO Instruments. The instruments for the STEREO spacecraft have not been selected. The Announcement of Opportunity for the instruments is expected to be released in April/ May of 1999 with full instrument team participation starting in October. This lack of instrument definition makes the conceptual

Table 3-1 Top-Level System Driving Requirements

Requirement	Parameter
Mission Life	2 years prime, 5 year extended (expendables to 5 years)
Science Data Volume	5 Gbit/day
Broadcast Mode	500 bps (when not in a DSN pass)
Science Power	60 Watts (20% Margin at system level)
Science Mass	66 kg (20% Margin at system level)
Navigation Knowledge	7,500 km
Radiation (total dose)	10 Krad
Required Orbits	See Mission Design Section 2.0
Time Maintenance	0.5 seconds between two spacecraft
Maximum Mass	350 kg w/ 20% margin
Spacecraft Differences	None[1]
Mission Ops Concept	De-Coupled Science and Spacecraft Operations
Cleanliness	Class 100,000 until instrument I&T then Class 10,000
Non-Bus Point Information Required	SCIP Loss-Of-Sun Error Signal
Pointing Knowledge (roll)	± 20 arcsec (3σ)
Pointing Knowledge (pitch/yaw)	± 0.1 arcsec (3σ)
Pointing Control (roll)	± 0.1° (3σ)
Pointing Control (pitch/yaw)	± 20 arcsec (3σ)
Jitter (roll)	30 arcsec (RMS)
Jitter (pitch/yaw)	1.5 arcsec (0.1 to TBD Hz)

[1]There will be minor differences between the instrument packages, however, the spacecraft and instruments will be form, fit and functionally identical.

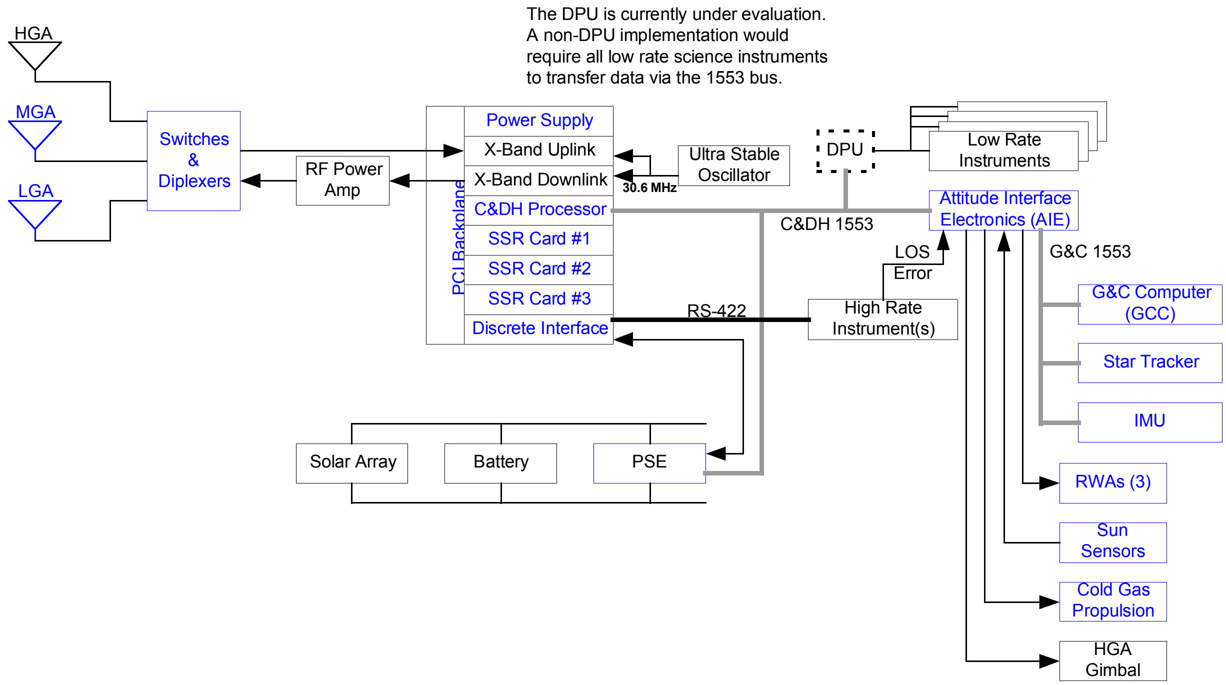


Figure 3-1 System Level Block Diagram

Table 3-2 STEREO Component Legacy

Spacecraft Component	Legacy	Scope of Changes
G&C Computer	TIMED	None
Star Camera	TIMED	None
IMU	NEAR	None
RWAs	NEAR	None
Sun Sensors	COTS	None
Attitude Interface Electronics	TIMED	None
Propulsion	OrbComm	Minor
SSR Cards	TIMED	Minor
C&DH Processor Card	TIMED	None
Uplink Card	TIMED	Major
Downlink Card	TIMED	Major
Discrete Interface Card	TIMED	Medium
Power Supply Card	TIMED	Minor
Ultra Stable Oscillator	Planet-B	Minor
Low Gain Antenna	NEAR	None
Medium Gain Antenna	NEAR	Medium
High Gain Antenna	COTS	Medium
HGA Gimbal	COTS	None
TWTA	Hughes	None
Battery	SWAS	None
PSE	TIMED	Minor

design of a spacecraft difficult. To fill this void APL has defaulted to the instruments discussed in the Science Definition Team Report for the STEREO Mission. This report discusses six instrument packages that makes up a complement necessary to meet mission objectives. These instruments are:

- Solar Corona Imaging Package (SCIP), two instruments in one package
- Heliospheric Imager (HI)
- Solar Wind Plasma Analyzer (SWPA)
- Energetic Particle Detector (EPD)
- Radio Burst Tracker (RBT) and
- Magnetometer (MAG)

These instruments are defined in terms of capability with heritage to previous instruments providing rough mass and power requirements. This provides enough information to package the spacecraft, including fields of view and develop mass and power budgets.

Though a conceptual design has been completed for this mission, the lack of instrument definition and interface control poses a significant risk to the design. In order to accommodate expected changes in the instrument baseline, we are carrying 35% margin in power and 18% margin in mass. The mass margin is considered smaller than is comfortable (for this level of design), however, many of the bus component masses used are actuals; therefore there is little margin risk due to the bus.

STEREO Subsystems. The STEREO spacecraft is broken into eight subsystems; Command and Data Handling, Software, Guidance and Control, Power, Telecommunications, Mechanical, Thermal and Propulsion. These subsystems are discussed in detail later in the report.

Data Routing. Figure 3.1 shows a physical block diagram of the system. The spacecraft operates with its x-axis (instruments) pointed at the Sun and the High Gain Antenna (HGA) pointed at the Earth. The position of the HGA changes at a frequency of several times per day. The instruments operate at a 100% duty cycle, generating data per their stored command sequences. The instrument suite generates approximately 5 Gbit of data per day with over 90% of the data coming from the Solar Corona Imaging Package (SCIP) instrument. In order to handle the high data rate from the SCIP an RS-422 interface will be added to the Discrete Interface Card in the Integrated Electronics Module (IEM). The other instruments send their data to the C&DH subsystem by way of the C&DH 1553 bus interface. This bus is shared with data from the C&DH subsystem.

Should there be a desire not to burden the smaller instruments with 1553 hardware, a Data Processing Unit (DPU) is costed as an option as described in Appendix C. The DPU would take data from all of the smaller instruments and format their data into a 1553 interface. This relieves the cost, complexity and mass burden from the smaller instrument providers.

Instrument data is generated 24 hours per day, all of which is stored on the Solid State Recorder (SSR), even during ground contacts. The real time instrument data that is recorded is assumed to have the same priority as previously recorded data. Therefore, real time data is not preferentially treated for downlink.

Once a contact with the Deep Space Network (DSN) has been initiated, stored data (science and spacecraft bus) from the recorder is formed into Consultative Committee for Space Data Systems (CCSDS) compatible transfer frames in the framing portion of the Downlink Card. From there the frames are moved to the Radio Frequency (RF) section of the board and on to the Travelling Wave Tube Assembly (TWTA). The C&DH processor is responsible for controlling the flow of data between the SSR and the Downlink Card, as well as between the SSR and the instruments.

Antenna selection for STEREO depends on the spacecraft's distance from the Earth with a goal of maximizing bit rate and thus minimizing DSN contact times. Nominally, the spacecraft relies on the Low Gain Antenna (LGA) for early operations, moves to the Medium Gain Antenna (MGA) and eventually to the HGA as the mission proceeds. However, the lagging spacecraft can operate from the HGA almost immediately after launch. This is not the case with the leading spacecraft. Due to a Sun-Earth-Probe angle of greater than 160°, the leading spacecraft cannot use its HGA until almost 200 days into the mission. This problem arises because the maximum swing angle of the HGA antenna is 115°. This leaves some gaps in data taking capability (Section 4.5). Resolution of this problem will take place at the system level and will be resolved during the next phase of APL's effort.

The same antenna that is selected to downlink data also receives spacecraft commands. The uplink RF is routed to the Uplink Card where the commands are decoded and routed. Commands for the spacecraft bus are routed either to the C&DH processor or, if marked critical, to the power system electronics (PSE) for immediate execution. Instrument commands are routed to the C&DH processor and "bent-piped" to the specific instrument.

Attitude Determination and Control. The STEREO spacecraft is a three-axis stabilized platform that relies on a star camera and Digital Solar Attitude Detector (DSAD) for coarse pointing and on an Inertial Measurement Unit (IMU) for rate information. In order meet the tight pitch and yaw pointing requirements (20 arcsec, 3σ, control) for the mission, the SCIP instrument will provide an error signal to the Attitude Interface Electronics (AIE). This error signal will provide the pitch and yaw knowledge required to meet the pointing requirements discussed in Section 4.3.

Attitude control of the spacecraft is accomplished by use of three Reaction Wheel Assemblies (RWAs) mounted along each of the spacecraft's principal axes. The spacecraft is configured as a zero-momentum system. A momentum-biased approach is a possible work around should an RWA be lost. The primary force generating adverse torque is solar pressure acting at the center of pressure (Cp) of the spacecraft. This force is proportional to the offset between the spacecraft Cp and Cg. This offset changes with the position of the HGA.

Once the system has reached a predetermined momentum, it is dumped by using the RWAs to generate torque against propulsion system firings. The RWAs can store enough momentum so that momentum dumping occurs on intervals of four days or longer. Momentum dumping occurs autonomously, in a time window that is set aside for spacecraft maintenance each day. Instruments are provided ample warning of when a propulsion event will occur so that they may safe themselves as required.

Processing for the attitude control system occurs in the Guidance and Control Computer (GCC). The primary tasks for this computer are processing all of the sensor data, run the control loops that manage the actuators, autonomously control system momentum and HGA steering and handle the safing function for the spacecraft.

The AIE acts primarily as an interface box for non-1553 instruments. Conceptually, it also acts as a back-up processor when the spacecraft goes into Earth-Acquisition Mode. The implementation of the safing architecture will be studied in the next phase of APL's effort.

Power. The spacecraft's power system consists of two solar array wings, a 21 AH Super NiCd battery and PSE. The spacecraft is designed to operate at 100% duty cycle without dipping into the battery, except for propulsion events, which nominally, occur on intervals of four days or greater. The battery is also used to support the spacecraft prior to solar array deployment and sun acquisition.

The PSE contains a Peak Power Tracker that controls the power generation of the system. It also contains the relays for power distribution and pyrotechnic events.

STEREO Configuration. The STEREO Spacecraft configuration is shown Figure 4-15. During all modes, the spacecraft is kept with its x-axis oriented toward the Sun. This orientation allows for the instruments to be properly oriented with respect to the Sun and the antenna suite to be properly oriented with respect to the Earth.

The STEREO bus has only two deployment mechanisms, these are for the two solar array wings. Each wing of the solar array consists of a single panel. This minimizes the complexity of the arrays.

The spacecraft is configured in such a way as to minimize the offset between the spacecraft's center of gravity (Cg) and center of pressure (Cp). The difference between these two affects the need for spacecraft momentum dumps and thus fuel load. The difference between Cg and Cp changes over the course of the mission due to changes in the HGA pointing angle. During Phase A, APL will assess the use of trimmable flaps to minimize the affects of this Cg-Cp offset.

The biggest driver of the spacecraft configuration was placing the largest HGA possible into a position where it has the most travel, without obscuring any instrument field of views. The results were a 1.1 meter dish with a travel from 5° to -115° where is zero degrees is bore-sighted with the x-axis. At the -115° point the antenna is slightly obscured by structure and loses approximately 3 dB of gain. This -115° is insufficient to solve the leading spacecraft's Sun-Probe-Earth (SPE) angle problem.

3.2 Spacecraft Fault Protection Architecture

The STEREO spacecraft has only one Operational Mode (Figure 3-2, Table 3-3). The lines of sight of the instruments are all pre-defined; this allows the whole bus to be kept pointing at the Sun. The requirement is to point within ±20 arcsec in pitch and yaw and ±0.1° in roll. The attitude control subsystem is provided with a "Loss of Sun" error signal that is generated by the Solar Coronagraph Instrument. This error signal is used in the attitude control loop to maintain pointing.

The spacecraft has two addition modes (Figure 3-2, Table 3-3), both of which are classified as safe modes. They are designated Safe-Hold Mode and Earth-Acquisition Mode. The spacecraft enters Safe-Hold Mode when a serious fault such as an unexpected battery discharge, computer reset or G&C health check violation occurs. When entering safe hold mode, the spacecraft suspends all time tagged commands, shuts instruments off (except for survival loads), positions its antennas towards Earth and lowers the telecommunications rate to a predefined emergency rate. The spacecraft can only revert back to Operational Mode by ground command.

The third spacecraft mode is an Earth-Acquisition Mode. Unlike Safe-Hold Mode, no roll axis knowledge or navigation data is assumed. The spacecraft enters this mode either directly from Operational Mode or from Safe-Hold Mode. This

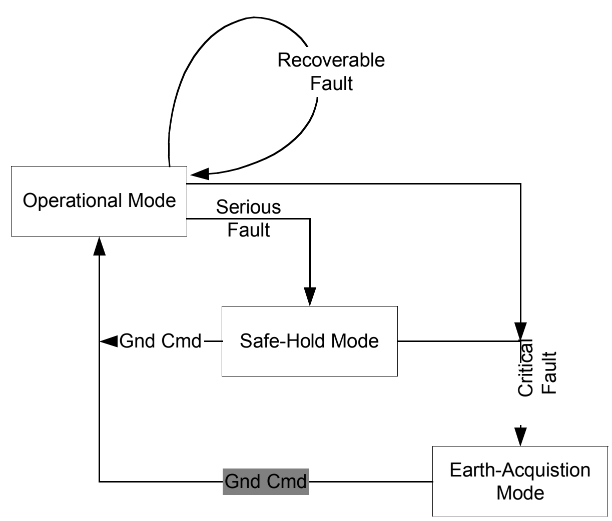


Figure 3-2 Mode Transition Diagram

Table 3-3 Spacecraft Mode Change Requirements and Mode Configurations

Spacecraft Mode Configuration	Mode Change Requirements
Operational Mode: • Time Tagged Cmds Enabled • All Instruments On • Sun w/all antennas pointed at Earth. • Telecom over HGA	Recoverable Fault: • Instrument Fault • Configuration Error
Safe-Hold Mode: • Suspend Time Tagged Commands • Instruments Off • Resest Spacecraft State • Sun point, antennas at Earth • Emergency Rate Telecom over MGA	Enters Due to Serious Fault: • C&DH or G&C Reset • Unexpected Batter Discharge • G&C Component Failure • G&C Health Check Violation • (Sun-Keep-In, Thruster Use, Orbit Span) • TBD
Earth-Acquisition Mode: • Suspend Time Tagged Cmds • Instruments Off • Reset Spacecraft State • Sun Point and rotate 1°/minute	Enters Due to Critical Fault: • Expiration of Cmd Loss Timer • Low Bus Voltage • Loss of UT • Multiple G&C Health Check Violations • TBD

mode is entered when a critical fault is detected in the system. Examples of such faults are loss of Universal Time (UT), multiple G&C health check violations (e.g., loss of attitude knowledge/ control) or a low bus voltage. When entering this mode, the spacecraft stabilizes (if necessary) with solar arrays pointing at the Sun and the spacecraft rolling about the x-axis at a rate of 1°/minute. This roll rate sweeps the MGA over the Earth at regular intervals. This allows the DSN to make contact with the spacecraft without the spacecraft having knowledge of the Earth's position. However, it assumes that the fault that caused the spacecraft to enter this mode did not affect the spacecraft's ability to control its attitude. During the next phase, we will look at the feasibility of eliminating the Earth-Acquisition Mode by making use of the broader beam low gain antennas (LGAs) for emergency contacts. We'll also look at a variety of implementation schemes including safing processors, bootable code segments for safing and hardware and software allocation of safing requirements.

3.3 Top Level Spacecraft Descope Plan

The STEREO spacecraft are single string spacecraft that do not easily lend themselves to descoping. No single piece of equipment can be removed from the spacecraft bus without causing the functionality of the spacecraft to be reduced. Therefore, the primary areas for descoping should be in the flight software, ground system hardware and software, program procedures (i.e., configuration control, integration and test) and Mission Operations preparation. Furthermore, it is important to note that the performance of any particular descope action falls off significantly as the program proceeds into its later stages. Table 3-4 summarizes the descope options and the benefits/penalties associated with them. Details on each option follow.

Table 3-4 Descope Option Summary

Descope Option	Benefit Due to Exercising	Penalty Due to Exercising
Eliminate Variable Length Packets	Cost of Software/Hardware Development and Test	Non-Optimal use of Downlink Bandwidth
Use Momentum Bias System	Cost of wheel(s)	Degraded pointing/jitter control
Remove RTX-2010	Cost of AIE Software Development and Test	Removal of B/U safing processor. Additional HW Costs
Build only one GSE set	Cost of GSE hardware and testing	Limited I&T options. Schedule delay.
Autonomous Contacts	Cost of Software and Test.	Additional MOC Staffing
Spacecraft Simulator	Cost of integrating and testing simulator.	Inability to fully test new software loads. Difficulty is ringing out spacecraft anomalies.
Eliminate Spacecraft Emulators (assuming exercise of option)	Cost of Emulators Hardware, Software and Test	I&T schedule risk associated with not testing instrument interfaces prior to I&T.

Flight Segment Descope Options. The primary area for descoping the spacecraft bus lies in the system software. Current plans are to make use of as much of the TIMED software and interfaces to Gournd Support Equipment (GSE) and Mission Operations as possible. This implies that descopes in software may have a ripple effect through the program because much of the system architecture has already been developed and vetted.

Since TIMED is our baseline, areas of software for descoping should come from those areas that are being changed from the TIMED baseline. Those areas include the use of variable length packets in transfer frames and much of the G&C software.

Spacecraft designers agree that the use of variable length packets allow for more efficient use of telemetry bandwidth and easier formation of the downlink data packets. However, hardware and software changes are necessary to enable them. A cost analysis will be made during Phase A to ascertain the cost associated with the use of variable length telemetry packets. Once these costs are understood, the benefit of using this as a descope option can be calculated.

Another possible descope option is to remove the RTX-2010 processor from the AIE. The AIE serves two primary functions. First, it provides an interface for all non-1553 attitude hardware. Second, it acts as a “processor of last resort,” should there be a failure in the G&C processor. If this option were to be exercised, the AIE would serve purely as an interface box and all safing software would migrate to a separate bootable section within the G&C computer. This results in software savings due to the difficulty is writing and compiling code for the RTX-2010 processor resident in the AIE. This reduces the functionality of the AIE and makes it purely an interface box. Software cost savings will have to be weighed against any additional hardware costs resulting from removing the processor. Additionally, this change affects the safing architecture of the spacecraft and must be assessed at the system level. This architecture change will be addressed in detail during the next phase of the program.

If the sponsor is willing to trade on pointing requirements, another descope option would be to move from a three RWA, zero momentum system to a single or dual wheel momentum-biased system. The momentum-biased system has the benefit of only operating on one or two wheel(s). This system may result in slightly degraded system pointing. Jitter control would also be effected. Furthermore, additional propulsion gas is necessary to control momentum precession and in the single wheel case, nutation damping as well. This option saves the cost of either one or two wheels, however, software costs would remain the same. As with all options, further analysis would be necessary to ascertain their complete impact.

Ground Segment Descope Options. Descoping efforts with regard to the ground segment fall into three categories: GSE, Mission Operations hardware/software and procedures. All three of these categories have certain areas that can be descoped, but at what savings to the program?

The integration and test philosophy for STEREO requires two sets of subsystem level GSE, one for each spacecraft. For example, two sets of power and RF GSE are required to operate both spacecraft simultaneously through the Integration and Test (I&T) process. This allows for both spacecraft to be processed concurrently, thus meeting our goal of a 10-month I&T schedule. The two sets of GSE also provide the flexibility of having each spacecraft located in different locations. This is useful should one spacecraft experience a problem. The second spacecraft would not be held up because the single set of GSE was tied up with the problematic spacecraft.

A possible descope option is to build only one set of GSE. This saves the cost of the additional set, but lengthens the I&T process and increased schedule risk. Of particular concern is thermal vacuum testing, which in order to meet the projected schedule, must be done in parallel.

It is unclear what the program savings would be if one set of GSE were eliminated. Though the I&T schedule would lengthen somewhat, it is assumed that the I&T team members would be more efficient at integrating the second spacecraft. Procedures and work arounds would already be in place for the second spacecraft. Under the dual GSE I&T plan, there was little time between subsystem integration on to the spacecraft, thus any procedural problems would most likely affect both spacecraft.

Another area for descoping in the ground segment is the elimination of spacecraft emulators (currently priced as an option) should they be required and/or the spacecraft simulator. Spacecraft emulators are delivered to the instrument providers and allow them to work with an emulation of the spacecraft and their electrical interface well before instrument I&T. Eliminating these emulators would save money, but would increase the risk of a troublesome instrument integration period. It would also prevent the instruments from fully testing their instruments prior to integration.

The spacecraft simulator is a composition of engineering units and GSE that is used to emulate the spacecraft and the environment it operates in. It is used during mission operations to test new software uploads and debug spacecraft problems. If this simulator were not built, the program would save the cost of simulator integration and test as well as any special software. However, without the availability of such a simulator, software uploads would become riskier due to the inability to test them in the spacecraft environment. Spacecraft debugging would also become more difficult because there would be limited ability to repeat the bugs on the ground.

The final area for descoping within the ground segment would be the elimination of the ability to do autonomous contacts. Current plans are to staff the MOC only during business hours rather than in support of every contact, which can occur anywhere in a 24-hour period. This relieves staffing pressures by not forcing personnel to work odd shifts. To allow business hour staffing, it is planned to handle all non-business hour contacts autonomously. The software and procedures for this would have to be developed during Phase C/D.

By eliminating autonomous contacts, the program saves the funds associated with developing and testing this capability. However, Phase E costs will rise because of the need to provide staffing at anytime during a 24-hour period. Additionally, these non-business hour operations typically lead to a high staff turn over rate. This makes mission operations more difficult because of the discontinuity in spacecraft expertise and the additional training requirements.

3.4 Risk Identifcation

This section discusses risks as they pertain to the spacecraft and mission operations development effort. The risks identified in Table 3-5 have been classified into four categories; cost, schedule, technical and operations risks. As with all programs the first three categories are interrelated. Risks are also given a subjective rating that assesses the difficulty of mitigating the risk and its effect on the program. A risk is rated high if the risk is difficult to mitigate and it greatly effects program cost, schedule and/or technical performance. These risks will receive particular attention during the next phase.

Single String Spacecraft Risk. Due to cost constraints, the STEREO spacecraft is a single string spacecraft based on a derivative of the TIMED design. In order to meet mission requirements, all hardware must work as specified. As discussed in Section 3.5, this does not preclude a degraded mission. As can be seen from Table 3-6, there are some hardware failures that will cause the loss of the spacecraft. This risk addresses the spacecraft's ability to recover from all software failures and a subset of hardware failures in such a way that ample time is given to construct work-arounds.

This risk is addressed by constructing a safing architecture that centers around a safe mode that relies on a small, well-tested section of software that can place the spacecraft in a slow rotation about the x-axis, with the x-axis pointed at the Sun. This rotation allows the MGA to sweep the Earth at a know frequency. The spacecraft must also be able to receive the commands in this mode. Where this software resides and what hardware complement is required to implement it, will be part of our next phase activities. We will also be looking at a safe mode that relies only on a set of LGAs. This allows for the safe-mode hardware complement to be reduced, by not requiring roll control or knowledge.

Lack of Instrument Definition. See Section 3.1.

Data Rates and High DSN Requirement. This risk addresses the requirement for eight hours of DSN time per spacecraft at the end of the mission. DSN requirements start off at two hours early in the mission and then escalate to four and on to eight hours. These requirements are based on a 5 Gbit science data volume, a 200 kbps maximum bit rate at 1 AU, a 40 watt (RF)

Table 3-5 Risk Summary

Area	Risk	Category	Level
System	Single String Spacecraft and Safing	Technical	Medium
System	Lack of Instrument Definition	Technical	Medium
System/Telecommunications	Data Rate, High DSN Requirement	Technical	Medium
Telecommunications	Leading Spacecraft Data Drop Out (SPE Angle)	Technical	High
Guidance and Control	Jitter Control	Technical	Medium
Integration and Test	Two Spacecraft I&T	Schedule	Low
Mission Operations	Autonomous Contacts	Operations	Low

Table 3-6 Failed Component Contingencies

Component	Failure Mode	Affect on Mission	Possible Work Around
Solar Array	String Failure	Loss of Power	Power sharing
Battery	Loss of cell	Minor Peak load affected	n/a
SSR	Single Card	Reduced Data volume	Use other cards
C&DH Processor	Card failure	Loss of Mission	None
X-Band Cards	Card failure	Loss of Mission	None
IEM Power Supplies	Card failure	Loss of Mission	None
Discrete Interface	Card failure	Loss of Mission	None
USO	Unit Failure	Degraded Navigation Data	None
AIE	Unit Failure	Loss of Mission	None
G&C Computer	Unit Failure	Degraded pointing	Use AIE as backup w/ new software
Star Tracker	Unit Failure	Loss of accurate roll knowledge; difficult reacquisition.	Low grade at best
IMU	Singe Gyro	Cannot meet pointing or jitter requirements	Use angle data from DSADS, ST and instruments.
Sun Sensor	Single Sensor	Slower Safe Mode Acq.	None
RWA	Singe RWA	Degraded Pointing Possible. Shorter Mission	Use other wheels and propulsion
Propulsion	Subsystem Failure	Loss of Mission	None
HGA Gimbal	Unit Failure	Reduction of Data Volume	Bus maneuver to point antenna
TWTA	Unit Failure	Loss of Mission	None
G&C Software	Critical Failure	None	Enter Safe mode (AIE software)

Travelling Wave Tube Assembly (TWTA), a 1.1 meter HGA and use of DSN's 34 meter Beam Wave Guide (BWG) antenna. Note that with regard to DSN charges and additional 45 minutes to one hour per pass is required for DSN setup and calibration time.

In order to reach the desired two hours per day per spacecraft of DSN time, the maximum bit rate at 1 AU must be raised to 800 Kbps. This can be accomplished by either increasing the spacecraft's antenna size and/or its radiated power and/or make use of DSNs 34 meter HEF antenna or 70 meter dish.

This trade is a complex one that involves the telecommunications, power, mechanical disciplines and mission operations disciplines. Due to the ramifications of this trade; it will be worked on early in the next phase of our effort.

Jitter Control. The jitter requirements for the STEREO mission are to keep line-of-sight jitter to levels under 30 arcsec RMS in roll and 1.5 arcsec between 0.1 Hz and TBD Hz in pitch and yaw. This requirement is driven by the SCIP instrument. These are challenging requirements. In order to meet these requirements a high control bandwidth is desirable, but not necessarily feasible. Limiting factors are wheel torque and linear range of fine pointing control. Other issues include the excitation of structures such as booms and solar arrays. Excitation sources are the RWAs, HGA Gimbal, instrument mechanisms and the propulsion system.

In order to assess the spacecraft's ability to meet the jitter requirements, a high fidelity simulation will be built that models the spacecraft's structure (including booms and solar arrays), RWAs and propulsion system. This model will be build early in the program. Should the model show that jitter exceeds requirements several mitigation are available including, control shaping, addition of a fourth RWA, structural damping and isolation of the RWAs. All of these have cost implications.

Integration and Test. In order to meet the current STEREO schedule it will be necessary to integrate and test two spacecraft in 10 months. This is an aggressive schedule. It almost demands that the two spacecraft be integrated concurrently. Concurrent integration, however, would have significant cost implications.

As a method of meeting this schedule, the two spacecraft will be integrated and functionally tested as if they were a single redundant spacecraft with a side A and a side B. This means that a subsystem is integrated onto the first spacecraft, functionally tested and then the same subsystem is integrated onto the second spacecraft. This occurs on intervals of about a week. The same integration method would apply to instruments also.

The methodology has several benefits:

- Allows the I&T team to remain in the same testing configuration for both spacecraft.
- Allows the I&T team to apply lessons learned from one spacecraft to the second.
- Permits the use of a single I&T team.

Once the spacecraft are integrated, they will be tested for performance and function concurrently using a single I&T team and scripting GSE. Dual GSE for power, RF and instruments will be required. Once functional and performance testing is complete, the spacecraft will be environmentally tested linearly except for thermal-vacuum which will occur concurrently. After thermal-vacuum testing, the spacecraft are shipped for launch.

This test philosophy represents our current thinking. However, detailed planning still needs to be done. Additionally, contingencies that a address a problem with a particular subsystem or instrument that effect one or both spacecraft need to be addressed. This analysis will occur during Phase A.

Autonomous Contacts. With the intention of reducing Mission Operation costs, a nine-by-five, week day schedule has been baselined for the Mission Operations Team. This provides the benefit, of smaller staffing levels and stability. In order to meet this desire and the requirement of supporting contacts that occur once per day, it will be necessary to automate some of these contacts. Automated contacts will support all non-business hour tracks.

While the use of automated contacts bestows significant Phase E savings, there are risks associated with it. Most of the risks are generated from the industry's lack of experience with automated contacts. Automation of functions such as command verification and acceptance, spacecraft recorder management, DSN interfaces, data archiving are some the areas that need to be addressed.

One should note that by the time STEREO launches the industry will have considerable experience with automated servicing of spacecraft. Programs such as TIMED, Wide-Field Infrared Explorer (WIRE) and The Compton Gamma Ray Observatory (GRO) either have baselined or are using automated operations as an adjunct. This greatly reduces the risks to the STEREO program because we can make use of their lessons learned. This is particularly true for TIMED which is another GSFC/APL program.

3.5 Inherent Spacecraft Redundancy

Clearly, the STEREO spacecraft is not a one for one redundant design, however, the spacecraft does have some inherent redundancy in several key areas that would allow for a degraded mission should a particular component fail. For example, failure of one or two SSR cards would not cause loss of the mission, though science volume may be reduced significantly. Table 3-6 shows the implications of the failure of a major component and what inherent redundancy exists to replace that component's functionality.

3.6 Selected Spacecraft Redundancy

Due to cost constraints the STEREO spacecraft is a single-string design. This poses several risks with regard to unit failures and safing architecture. Section 3.5 addresses some inherent redundancy that exists in the STEREO spacecraft and lists possible work around that could be put into place should there be a component or unit failure. Another possibility is to add some selected redundancy to the spacecraft in areas of higher risk. Cost must also be considered in selecting these redundant items. Table 3-7 lists areas where selected redundancy can be applied, the cost of said redundancy and the risk reduction effected by adding this redundancy.

As one can see there is a direct relationship between additional program cost and the effect on overall risk. Considering the cost constraints on the program it is difficult to argue for significant levels of redundancy that one might get from adding an additional IEM or processor. Costs for this sort of redundancy come from the cost of the unit itself, the additional software and testing.

The items that seem attractive from a cost-benefit perspective are adding an additional RWA, TWTA and/or fine sun sensor. The additional RWA gives each spacecraft 4:3 redundancy, but

Table 3-7 Risk vs. Reward for Selected Redundancy

Item	Redundancy Added	Cost per S/C	Effect on Overall Risk (high is better)	Other Benefits
RWAs	Add 1 RWA, 4:3	~$170k	Low	Enable Wheel Speed Control
GCC	2:1	Medium (SW costs)	Medium	
AIE	2:1	Medium – High (difficult to implement)	Low-Medium	
Sun Detectors	Add fine DSAD	Low	Medium	Enables pointing with loss of LOS signal
Star Tracker	2:1	~$500K	Low	Better bus pointing knowledge
IMU	2:1	$621K for two IMUs	Low–Medium	
Single IEM Card	Selected	Medium (Chassis and backplane redesign	Low–Medium	
IEM	2:1	Very High	High	Almost full redundancy
TWTA	2:1	Low	Low–Medium	
Battery	2:1	Low	Low	

Table 3-8 Sparing Philosophy

Sub-System	Flight Item	Total Required (both spacecraft)	Spares Purchased/ Fabricated	Notes
G&C	RWAs	6	0	
	Star Camera	2	0	
	IMU	2	0	
	Sun Sensors	2 sets	0	
Prop	Propulsion Hardware	2 sets	0	
EPDS	Solar Panels	4	0	
	Battery	2	1	
RF	TWTA	2	0	
	Ultra Stable Oscillator	2	0	
	RF Switches/Diplexers	2 sets	0	
	Antennas	2 sets	0	
	HGA Gimbal	2 sets	0	
In House Fab.	Discrete Components	2 sets	1 set	Spares will not be kitted
	Unpopulated Boards	2 sets	2 sets	Little or no cost
	Structure	2	0	

it also enables wheel speed control which could significantly mitigate the jitter problem. Adding a TWTA provides redundancy for an item that, historically has some reliability problems. Adding a fine sun sensor provides some redundancy for the bus's dependency on the SCIP for pointing information. A fine sun sensor would enable fine pointing control should the SCIP fail or the Loss of Sun (LOS) signal drop out. Of course, any of these options affects a dwindling mass margin.

3.7 Sparing Philosophy

The sparing philosophy chosen for the STEREO mission is consistent with those used on other APL spacecraft. The philosophy is slightly modified because of the tight cost constraints and multiple spacecraft build. Table 3-8 outlines our sparing philosophy for procured and in-house fabricated components.

As with recent APL programs, there will be no sparing of procured items such as the IMU, Star Tracker or HGA Gimbal. These components are spared at the piece part level at the vendor. It is assumed that, should a problem arise with one of these components, it could be sent back to the vendor where repairs would be made in relatively short order. The one exception is the spacecraft battery, where a single spare in being purchased for both spacecraft. This is due to the long lead time associated with the battery and the inability to make repairs on it.

Fabricated items are being spared at a 1:2 (spare to required) ratio at the discrete component and board levels. Spare flight components are being purchased to preclude any problems that may occur during the board fabrication process. These items will remain un-kitted, in bonded stores. Spare flight boards are being fabricated because of the low cost of fabricating additional boards while the flight ones are being manufactured. These boards will remain unpopulated and in bonded stores.

Historically, this philosophy has been shown to present little risk to the program. In the past, procured flight items have rarely caused schedule delays that could have been solved by having an available spare. The purchase of spare discrete components and the fabrication of additional boards alleviate risks associated with internal fabrication of components.

It is important to note that should a problem arise, it may be possible to make use of the second STEREO spacecraft being fabricated. Current plans call for each spacecraft to be launched, two months apart, on separate Athena II expendable launch vehicle (ELV)s. The plan is for the one spacecraft to lag the other (in schedule) and come together at the end of integration and test. This timing may allow components on one spacecraft to act as temporary spares for the other spacecraft during subsystem testing. It may also be possible to switch the order of the spacecraft should the need arise.

3.8 Spacecraft Mass and Power List

This section describes the STEREO mass and power component level allocations. The program goal was to enter Phase A with a 20% margin in both mass and power. All margins are kept at the system level.

The mass margin for the spacecraft is currently 18%. However, many of the weights used for components are actuals and therefore have little error associated with them. The largest mass margin risk lies with the instruments, which have yet to be selected.

The current margin on the power system is 35%. This number is sensitive to the aphelion of the lagging spacecraft because the solar panels are as large as they can get within the Athena II fairing (without additional hinged panels). Currently, the mission design calls for a lagging spacecraft aphelion of 1.125 AU. Like the mass margin, the largest risk to the power margin is the undefined instrument suite. Power and mass figures are given in Table 3-9.

3.9 Technology Insertion Areas

During Phase A, the STEREO program will be studying several areas for technology insertion. They include non-propulsive momentum dumping, non-coherent transceiver navigation, advanced battery chemistry and advanced recorder management. The only technology insertion candidate that is part of the baseline is the non-coherent navigation. This is because of the relatively low risk and cost savings that it provides to the mission. All of the candidates are discussed below. They all offer benefits to STEREO and other spacecraft. The risk vs. reward will be studied during Phase A.

Trimmable Flaps for Momentum Dumping. Each of the STEREO spacecraft is capable of countering adverse torque imparted to the system caused by solar pressure acting at the Cp. This torque is proportional to the offset of the spacecraft's Cp from its Cg. If this Cg–Cp offset were constant throughout the mission, it would be possible to correct the Cp by adding some structure, which would move the Cp to coincide with the Cg. However, because the HGA changes it position throughout the mission, the Cp is constantly moving.

Any adverse torque to the spacecraft is countered by using the three RWAs. This causes the RWA's rotational speed to increase, eventually, to the point where the wheels need to be de-saturated. This is done using the spacecraft's cold gas propulsion system. The spacecraft carries enough propulsion for the five year mission. The requirement is for the RWAs to store enough momentum for a four day period.

A previously untried method of minimizing momentum build up is to shift the Cp to coincide

Table 3-9 Mass and Power Budgets

All Masses are in kg			Peak Power	
All Power is in Watts	**Mass**	**Totals**	**Normal Ops**	**Totals**
POWER SUBSYSTEM		58.20		19.3
Ga-As Solar Array (2 wings, 36 sq.ft.)	16.40		0.0	
Super Nickel-Cadmium Battery (21 amp-hr)	23.80		0.0	
Power Switching Unit (PSE)	13.50		19.3	
Peak Power Tracker (PPT)	4.00		As part of PSE	
Power Shunt/Fuse Box	0.50		0.0	
ATTITUDE CONTROL SUBSYSTEM		46.80		80.5
NEAR Inertial Measurement Unit	5.50		25.0	
NEAR Reaction Wheel (3 reqd) and Electronics	12.90		9.0	
TIMED Star Tracker	6.40		12.5	
TIMED Attitude Flight Computer (AFC)	2.40		20.0	
TIMED Attitude Interface Unit (AIU)	6.60		7.0	
Cold Gas Prop System (4 thrusters)	11.00		6.0	
Adcole Sun Sensor (5 heads reqd) and Electronics	2.00		1.0	
RF SUBSYSTEM		21.00		80.0
High Gain X-Band Dish Antenna (1.1 m dia)	6.50		0.0	
Antenna Gimbal Drive and Electronics (90°)	4.50		0.0	
X-Band Amplifier (TWTA w/power supply)	3.60		80.0	
RF Coax Switch (3 reqd) Assembly and Flex Cables	3.70		0.0	
RF Diplexer	0.20		0.0	
Mid Gain X-Band Fan Beam Antenna (2 reqd)	1.00		0.0	
Low Gain X-Band Patch Antenna (2 reqd)	1.50		0.0	
AVIONICS SUBSYSTEM		12.80		56.3
TIMED IEM (9 card design)	12.30		55.5	
MSX type Ultra Stable Oscillator	0.50		0.8	
THERMAL SUBSYSTEM		17.10		21.5
MLI Blankets, Heaters and Thermostats	16.00		20.0	
TIMED Remote Interface Unit (RIU) (5 reqd)	1.10		1.5	
INSTRUMENT SUBSYSTEM		66.00		60.0
Solar Coronal Imaging Package (SCIP)	30.00		20.0	
Energetic Particle Detector (EPD)	3.00		2.0	
Solar Wind Plasma Analyzer (SWPA) and Elec.	7.00		4.0	
Radio Burst Tracker (RBT) Electronics	4.00		12.0	
Hingelock 621 Deployer (10 meter) for RBT (3 req'd)	4.00		0.0	
CME Heliospheric Imager (HI)	6.80		20.0	
GSFC Magnetometer and Electronics	2.00		2.0	
Astro Bi-Stem Actuator (6 meter) for Magnetometer	4.00		0.0	
Instrument Bench (SCIP and IMU)	5.20		0.0	
SPACECRAFT BUS SUBSYSTEM		64.80		0.0
Prim. and Sec. Structure @ 12% of 350 kg	42.00		0.0	
Wiring Harness @ 5% of 350 kg	17.50		0.0	
Spin Balance Weights @ 1.5% of 350 kg	5.30		0.0	
	Total	286.70		317.6
	Margin	18.1%		35%
			Power System Capability	429.1

with the Cg by extending a “trimmable flap” or flaps. The apparent size (as viewed from the +x direction) of the flap(s) would be changed as the HGA is slewed, thus removing the Cg–Cp offset created by the HGA. The flap can also be used in lieu of the propulsion system for de-saturating the RWAs. This type of implementation has two benefits. First, it greatly reduces the frequency of propulsive events. Propulsive events will most likely be preceded by the instruments ceasing data taking and covering sensitive optics. By reducing the frequency of the propulsive events, there will be fewer interruptions in data acquisition. There will also be fewer times that the cover mechanisms are actuated on some of the instruments. There is a finite risk that every time a cover closes it may not open.

The second advantage of this method is that propulsion mass can be reduced. This can be of great benefit to other spacecraft in the Sun-Earth Connection Program that rely on large and expensive propulsion systems to counter the same effects. The flaps could also act as a momentum dumping back-up for the propulsion system.

Non-coherent Navigation. The non-coherent navigation process has been designed and demonstrated from a technical point of view, and is part of the STEREO baseline. It has the potential of replacing an expensive transponder with a simple and cheaper transceiver. The process involves making two one-way measurements instead of the usual two-way measurements. The uplink frequency is measured against an onboard oscillator using counters in the receiver card. This measurement is placed in the spacecraft telemetry and used to correct the downlink Doppler measurement. Further details are explained in Section 4.5.

Recovery of the navigation data requires some interaction between the DSN radiometric data center, the APL navigation team, and the DSN navigation team. Phase A activity will define a data processing architecture to incorporate the processing steps of the technology demonstration into a routine and smooth part of the day-to-day STEREO data processing flow.

LiIon Battery. The battery baselined for the STEREO spacecraft is a 21 AH super-NiCd battery. This battery is used during spacecraft fly-out, propulsive events and during the process of safing the spacecraft (if the spacecraft is not Sun pointing). The NiCd battery weighs 23.8 kg. The NiCd battery also has a relatively high self discharge rate, which means it must be kept on a trickle charge until just before launch.

Trickle charging of the battery is a normal procedure for an Athena II launch. However if the launch vehicle is changed to the Space Transportation System (STS), trickle charging the battery becomes a complicated and costly process, because it requires an electrical interface with the STS.

An alternative chemistry is LiIon which has the benefit of having a higher energy density and a lower self discharge rates. LiIon batteries are currently being developed and qualified for space. STEREO is an ideal candidate mission for this technology due to its few charge/discharge cycles.

Advanced Recorder Management. Each of the STEREO spacecraft carries a 7.5 Gbit SSR that is used to store both scientific and house keeping data. Data is downloaded from the recorder during DSN passes. As data blocks are received at the MOC, the MOC generates commands that tell the recorder to open those blocks on the SSR for re-writing. If a data block does not make it successfully to the MOC due to an intermittent or marginal downlink channel, the data block isn’t cleared and a re-send is sent to the spacecraft.

In order to enable this functionality it is necessary for the MOC to keep read and write pointers for each of the SSRs. These pointers tell which point in the memory to start reading and writing from. For previous APL missions, these pointers were tracked manually, a very cumbersome task.

The Pre-Phase-A effort has identified some possible techniques for automating solid state recorder management to ensure maximum telemetry data recovery. During the next phase these will be evaluated for cost effectiveness in the context of the STEREO mission environment.

3.10 Launch Vehicle

A survey of available launch vehicles was conducted to determine which ones could satisfy the STEREO payload volume and 350 kg lift mass to a C3 of 1.0 km^2/sec^2 requirements. Initially payload users guides were used to ballpark the various capabilities. Requests For Information (RFI's) were then sent to the top candidates to obtain the latest configurations and capabilities. Finally the best candidates were invited to present their capabilities to the STEREO staff at APL. Information was obtained or requested from Taurus, Athena, Delta and Shuttle Via United Space Alliance [USA]). The usable fairing dimensions and lift capability of all present or near term planned configurations were obtained and are summarized in Figures 3-3 to 3-6.

Since Taurus could not meet the lift mass requirement, two single Deltas exceeded the allocated launch vehicle budget and dual STEREOs on a single Dual Payload Adapter Fitting (DPAF) Delta with their required STAR-37FM kick stages exceeded the available volume; Taurus and Delta options were dropped from further consideration.

ROM proposals were then solicited from Athena and USA for a more in-depth performance and cost evaluation. The results of that evaluation are summarized in Table 3-10. Giving maximum importance-weighted points to the better technical option and downgrading the points given to the other option proportional to the amount of difference resulted in a pseudo-quantitative evaluation.

Total points show the two options to be equal. However, mission science is paramount and overall cost runs a close second in importance. The other issues are just engineering challenges. The Shuttle's mass and volume advantage that allows a much larger high-gain antenna (which greatly improves science data downlink and minimizes DSN time and data void issues), plus its slightly lower cost would seem to tip the scales in its favor. A final launch vehicle decision will be made within 60 days after the start of Phase A.

Table 3-10 Launch Vehicle Discriminator Summary

CHARACTERISTIC	ATHENA II STAR 37FM	Pts	SHUTTLE STAR 48V	Pts
Lift Mass to a C3 of 1.0 km2/sec2	350 kg	18	**500 kg**	**20**
Orbit Inclination	28.5°	4	**28.5-57°**	**5**
Usable Fairing Dimensions	78.1" D x 77" L + 79" L cone to 36" D –STAR-37FM	13	**Essentially Unlimited**	**15**
Maximum High Gain Antenna Diameter	1.1 m - no inst. blockage 1.3 m - some inst. blockage	10	**Essentially Unlimited**	**15**
Maximum Design Axial Load Factor	5.8 ±5.0 g's	4	Liftoff **X(Axial) +6.4 g's** Y +2.0 g's Z +5.0 g's	**5**
Maximum Design Lateral Load Factor	**0.3 ±1.5 g's**	**5**	Landing X -3.6 Y +4.0 Z -8.4	4
Required Design Stiffness	>30 Hz and not between 45 - 70 Hz	7	**> 10 Hz**	**10**
Peak Separation Shock	LV/STAR-37FM 3000 g's STAR-37FM/S/C 6000 g's	4	**STAR 48/S/C 6000 g's**	**5**
Spin Balance and Balance Weights Required	Yes (STAR-37FM Burns Spinning)	6	**No (STAR 48V burns 3 axis)**	**10**
Manifesting	**36 months ARO**	**10**	**USA has 2 reimbursable flights**	**10**
Maximum Time From Launch To Deployment	**180 minutes**	**10**	Up to 10 days	7
Time From spacecraft (S/C) to launch vehicle (L/V) Mate to Launch	**2 weeks**	**10**	5 weeks	7
Fracture Analysis Req'd on All Structure	**No**	**10**	Yes	6
Structure Test/Analysis Correlation Req'd	**No**	**10**	Yes	6
Qualification Status	**Lunar Prospector has flown**	**10**	All parts flown or qualified, system not	7
Cost Impact to S/C	**+5% (tight mass & vol)**	**10**	+20% (safety and bureaucracy)	7
LV Cost to STEREO Program	2002 - 1.53 XX $M + ELVS 2004 - 1.55 YY $M + ELVS	13	**2002 - XX $M** **2004 - YY $M**	**15**
TOTAL		**154**		**154**

Version	kg to C3 = 1.0	
	Fairing size	
	63”	92”
Std 4 Stg	289	242
XL 4 Stg[1]	343[2]	296
XL 5 Stg[1]	374[2]	327

[1] Not Qualified
[2] Estimated

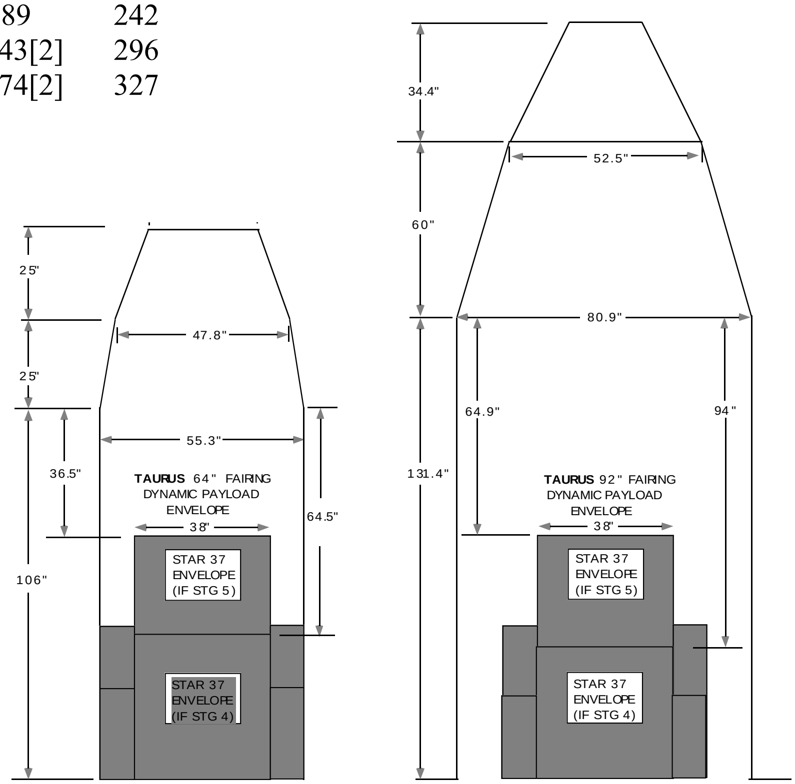


Figure 3-3 Taurus Launch Vehicle

Version	kg to C3 = 1.0
II 6T MP w STAR-37FMV [1]	300
II 6T MP w STAR-37FM [2]	**350**
II 6T MP w STAR 48AV [1]	420

[1] Not qualified
[2] Lunar Prospector configuration

92” Fairing

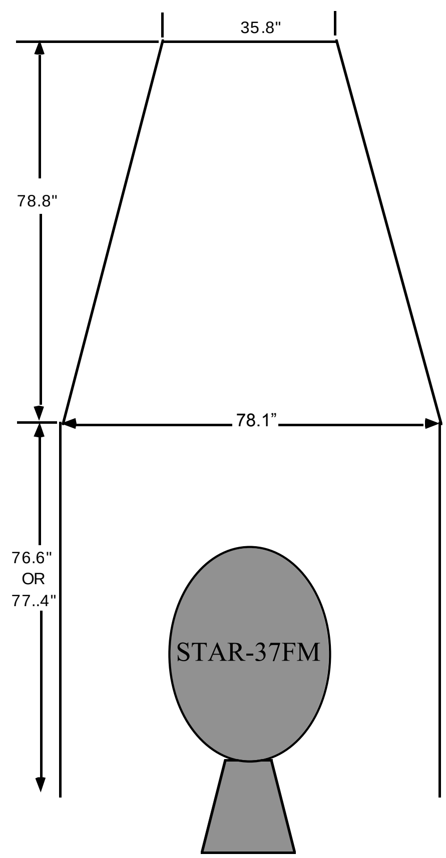


Figure 3-4 Athena II

Version	**kg to C3 = 1.0**
<u>**Single S/C per Launch**</u>	
7326-9.5 (STAR 37FM)	**600[1]**
7920-9.5[4]	**650[1]**
7325-9.5	**710**
7925-9.5	**1300**
<u>**Dual S/C Launch**</u>	
7920-10L DPAF[2]	**413 (4970[3])**
7320-10L DPAF[2]	413 (2735[3])

[1] OLS number
[2] To 100 nmi Park Orbit
[3] 3450 kg required for 3310 m/sec
[4] 3-axis stabilized release

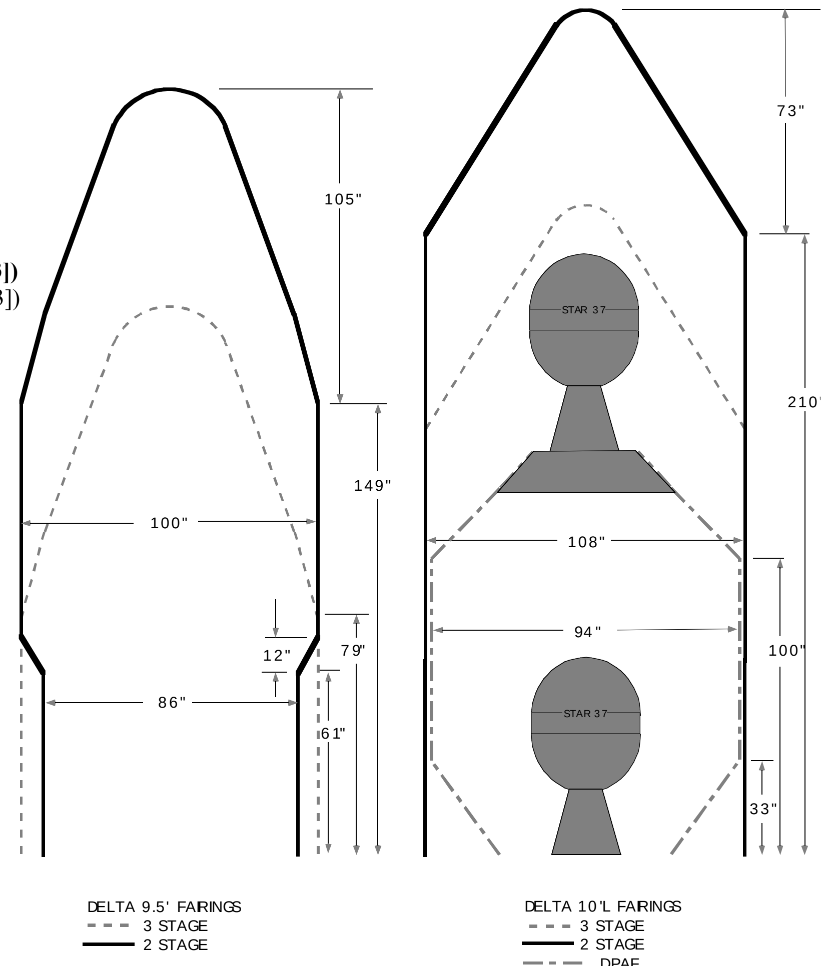
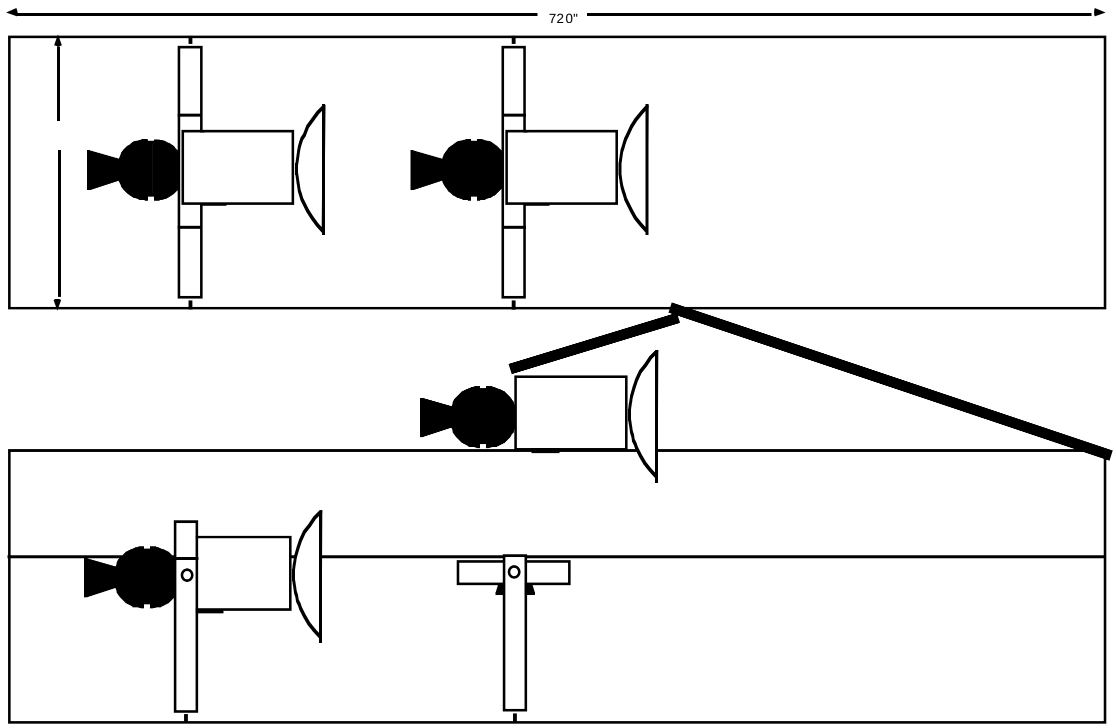


Figure 3-5 Delta II



Version	**kg to C3= 1.0**
STAR-37FM[1]	<350[2]
STAR 48V	**500[2]**

[1] GSFC FSS cradles for STAR 48 exist by require modification
[2] 3310m/sec from 100 nmi park orbit

Figure 3-6 Shuttle

4.0 SUBSYSTEM CONCEPTUAL DESIGN

4.1 Command and Data Handling Subsystem (C&DH)

This section describes the requirements and implementation for the Solar TErrestrial RElations Observatory (STEREO) C&DH subsystem.

4.1.1 C&DH Subsystem Requirements

The STEREO Command and Data Handling (C&DH) System baseline design implements the same architecture as that used on the Thermosphere, Ionosphere, Mesosphere, Energetics and Dynamics (TIMED) spacecraft. The goal is to contain cost and risk by using identical designs where practical and upgrading existing elements where necessary. A listing and brief description of the C&DH system functional requirements follows.

Uplink Command and Stored Command Management. The C&DH system must be able to decode and process Consultative Committee for Space Data Systems (CCSDS) compatible commands received via the uplink as well as ones stored on-board. The uplink must be able to support two data rates. The nominal rate will be 125 bits/second with a 7.8125 bits/second emergency mode. A third data rate of 500 bits/ second is being evaluated.

Telemetry Data Processing. Telemetry transfer frames generated by the C&DH system must be CCSDS compatible. Science data is to be gathered and stored at the combined maximum data generation rate of the instruments, approximately 410Kbits/second. The downlink system must be capable of supporting a real-time downlink mode for each of the instruments individually. The requirement for the maximum downlink data rate is derived as the rate required to transmit 5 Gbits of science data (plus housekeeping) in a two-hour DSN contact (actual downlink time). Providing for margin, the maximum required downlink data rate is 800 Kbits/second.

Other requirements placed on the telemetry processing function include the ability to provide a variable bandwidth allocation for instrument data which would be selectable by the science team. This would be an in-flight variable parameter which would allow the science data collection to be tailored to better view and study solar events throughout the mission and to accommodate differing mission phases. A “Broadcast Mode” is also supported, in which a subset of instrument data will be collected, framed and transmitted continuously at a 500 bps rate, except during high rate data transmission periods. While playing back recorder contents, the downlink system must also be able to support interleaving real-time data with recorded data.

Mass Storage Of Science And Engineering Data. The mass storage requirement calls for 5 Gbits of data volume reserved for science data with additional room for housekeeping data, overhead and margin. There is also a requirement for simultaneous read/write capability to accommodate data collection while performing downlink operations. In addition, random access capability is desired to support re-transmission of lost downlink data without having to perform full recorder contents dumping. Error management within the recorder will contain the error rate to $<10^{-9}$ bit errors for data held for three days. There must also be a means of supporting graceful degradation within the recorder to mitigate the potential loss of data over time due to decreased recorder performance.

Execute Autonomous Fault Protection. The C&DH system must support any spacecraft autonomous operations as implemented in hardware and software. Some examples would be single-event-upset recovery andresponse to monitored voltage or current telemetry conditions with pre-programmed or hardwired actions.

Maintain and Distribute Universal Time (UT). The C&DH hardware and software is required to maintain UT to within a 0.1-second accuracy and distribute it to instruments and other subsystems as required.

Provide For Subsystem Intercommunication. The C&DH system is the bus controller for the MIL-STD-1553B bus referred to as the C&DH 1553 bus. It is required to manage this bus and maintain the communication schedule between the C&DH system, power system, attitude control system and the instruments.

4.1.2 Baseline Design Solution

Integrated Electronics Module Definition. The C&DH electronics will be contained within a single enclosure referred to as the Integrated Electronics Module (IEM). It is a nine card system, partitioned into the functions identified in Table 4-1. Communication between the cards is accomplished via interconnections routed on a printed circuit board backplane.

Table 4-1 IEM Subsystem Partitioning

IEM Subsystem	No. of Cards	Function
C&DH Processor/ 1553 Card	1	C&DH
Command and Telemetry Card	1	C&DH
Solid State Recorder	3	C&DH
Downlink Card	1	C&DH and RF
Uplink Card	1	C&DH and RF
DC/DC Converter Card	2	Power Subsystem

Note that not all of the cards contain exclusively C&DH functions. The Downlink Card, for example, contains primarily telecommunications electronics, but also contains telemetry frame formatting electronics as part of the C&DH operations. A command decoder, also a C&DH function, resides within the Uplink Card. Figure 4-1 illustrates these functions within the IEM. The bold boxes identify the C&DH functions. The IEM external dimensions are roughly $10 \times 13 \times 7$ inches. The daughter boards are standard SEM-E sized, having dimensions of roughly 8.5×6 inches.

IEM Internal Requirements Flow-down

Processor/1553 Subsystem. The hardware requirements for the Processor/1553 Card are driven by the C&DH software development and execution requirements. The Central Processing Unit (CPU) is required to be a 32-bit architecture with an approximate throughput of 3 million instructions per second (MIPS). The memory requirements, based upon the TIMED C&DH software requirements, are 2 Mbytes of Static Random Access Memory (SRAM) for program execution and 4 Mbytes of Electrically Erasable Programmable Read-Only Memory (EEPROM), for program storage. The processor card is also required to contain a MIL-STD-1553B bus port for communication with some of the instruments and the G&C system. A Peripheral Component Interconnect (PCI) port is necessary for communication with other IEM subsystems across the backplane.

Solid State Recorder. The total mass storage requirement for the recorder is on the order of 5 gigabits for science data plus room for housekeeping data, overhead and margin. Given that the data storage rate is required to support the Solar Corona Imaging Package (SCIP) producing data at 400 kbps, a recorder peak write rate of 450 kbps should be sufficient, providing greater than 10% margin. The peak read rate is driven by a system requirement to be able to dump the entire recorder contents in a two hour (plus setup) Deep Space Network (DSN) pass. To satisfy this requirement, a 7.5 Gbit recorder will have to be read at approximately 800 Kbps, allowing for margin. Random access as well as simultaneous read and write operation must be supported. In order to support graceful degradation, the recorder must also provide a means for mapping around bad memory elements.

Downlink Framer Subsystem. The Downlink Framer is a digital subsection of the downlink electronics. This logic is required to perform the final transfer frame assembly to yield real-time, recorder and null telemetry frames which are then passed on to the downlink modulator. The telemetry data is received from the C&DH Processor and locally buffered. The Framer must then serialize the data and add Reed-Solomon encoding prior to transferring the data to the modulator. Other Framer requirements include the ability to pass modulator mode information from the C&DH processor, such as bit rate and convolutional encoding selection. 1/2 or 1/6 convolutional encoding will be selectable.

Uplink Command Decoder. The command decoder is a digital function within the Uplink Card. It is required to take CCSDS compatible commands from the uplink receiver command detector (or from the ground support equipment) and route them to the C&DH processor via the Command and Telemetry (C&T) Card. All relay commands received by the command decoder, whether from the uplink receiver or from the C&DH processor, are routed to the power switching subsystem under control of this circuitry. The command decoder is required to detect errors in any given Command Link Transmission Unit (CLTU). If there is an error the command is rejected by terminating its transfer to the C&T Processor and flagging it as being in error. If desired for STEREO, the command decoder can also provide a hardwired relay command sequence which is executed to perform an autonomous orderly load reduction in the event of a low bus voltage indication from the power subsystem.

Remote Interface Units. Spacecraft temperature information will be monitored, collected, converted from analog to digital and buffered by five remote telemetry units, each of which is

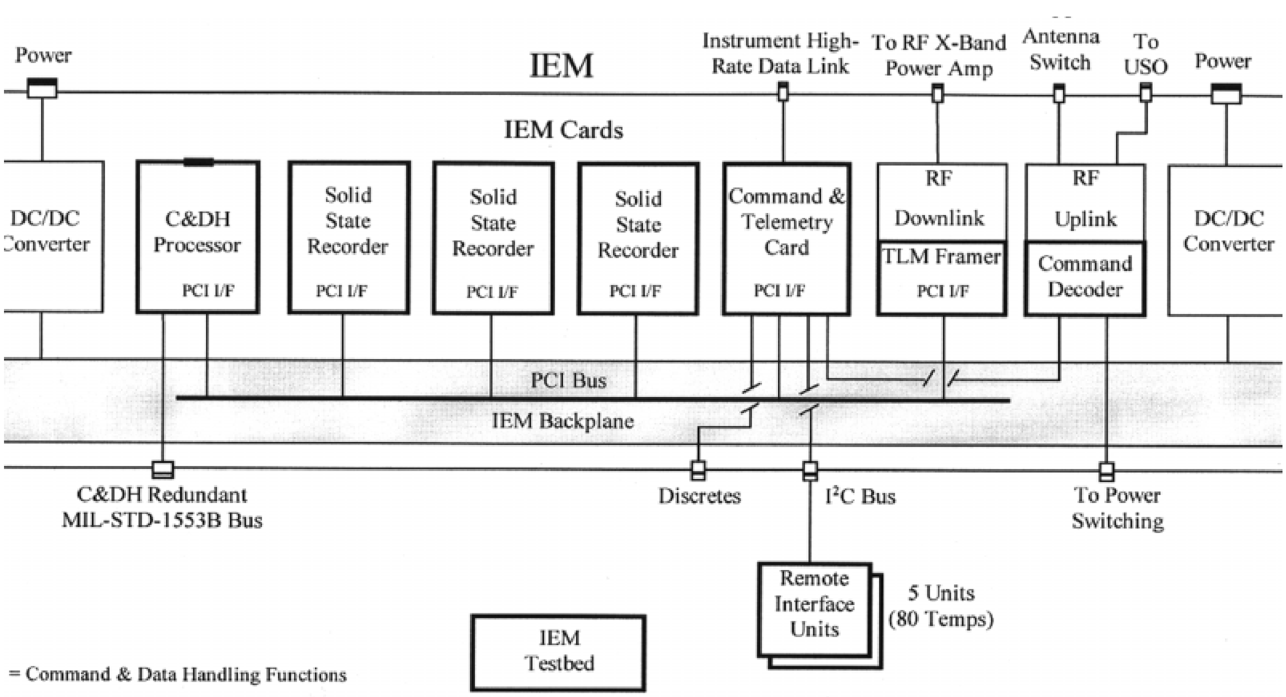


Figure 4-1 Integrated Electronics Module Configuration

capable of acquiring data from 16 temperature sensors. A total of 80 temperature thermistors can be monitored in this fashion. The five units will communicate with the C&T subsystem in the IEM via a serial digitalInter-Integrated Circuit (I^2C) bus. Each of the five units are daisy-chained together via the bus and connected to the IEM. Precautions are required to mitigate the likelihood of a failure in a single RIU that disables the entire I^2C bus. The baseline STEREO Remote Interface Unit (RIU) design is an existing TIMED design which can be replicated with no required changes.

4.1.3 Candidate C&DH Trade Studies

Solid-State Recorder—Make vs. Buy. The justification for initiating a make-versus-buy process is based upon the assertion that there is a potential for reducing the cost and perhaps cost-risk by purchasing some items available through military and aerospace system manufacturers. In the case of the solid-state recorder, the intent would be to seek an off-the-shelf component with a standard interface (e.g., MIL-STD-1553 or RS-422) that would satisfy the mass storage requirements as a self-contained unit. This would have to be weighed against the fabrication and testing of three copies of the existing TIMED design (six copies for the entire mission). It will also be weighed against another option; upgrading the existing TIMED Solid State Recorder (SSR) design by integrating it on to a single board.

Solid-State Recorder—Upgrade **.** By making use of newer memory technologies and packaging techniques, the SSR memory density can be improved from 2.5 Gbit/card to 10 Gbit/card. The advantage is that the number of cards tested is reduced from six to two. New memory devices will have to be identified and incorporated. This involves Floating Point Gate Array (FPGA) and board layout modifications. Since the functional aspects of the design would remain mostly unchanged, the most significant change is accommodating the wider address bus. These changes are considered to be moderate in terms of the effort and risk. This study would weigh the risk involved against the cost of building and testing three copies of the existing TIMED design or purchasing an off-the-shelf design.

4.1.4 Subsystem Mass, Power, and Heritage

Based upon values extrapolated from the TIMED IEM design, the STEREO IEM configuration will have an average power of approximately 54 watts and a peak of 62 watts. The total IEM mass is expected to weigh approximately 10.5 kg, to which is added the mass of five RIUs (0.23 kg each), for a total ofabout 11.7 kg. As indicated throughout this document, the IEM baseline design is to a large extent predicated upon TIMED heritage. Designs are being migrated intact where possible, and modified or upgraded to the extent necessary in order to accommodate the STEREO mission requirements.

4.2 Flight and Ground Software

4.2.1 Introduction

APL has responsibility for two primary elements of the STEREO mission: the Spacecraft Bus (including integration and test with science instruments), and the Mission Operations Center. Science instruments, the Science Operations Center, and their corresponding flight and ground software are the responsibility of the Goddard Space Flight Center (GSFC).

4.2.2 Overview of STEREO Mission Components

Figure 4-2 shows the primary STEREO software components. APL responsibilities are shown in color. The flight components consist of the C&DH and G&C subsystems and instruments for two identical spacecraft.

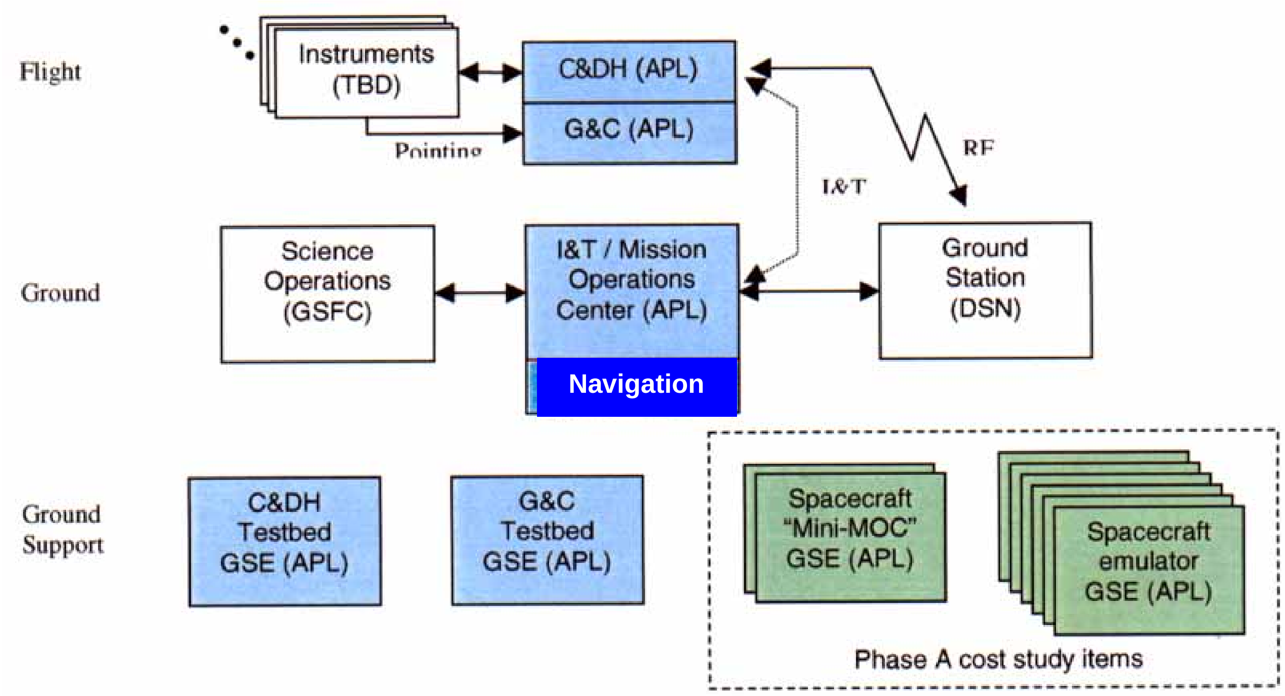


Figure 4-2 STEREO Mission Software Components

The major ground components consist of the Science Operations Center (SOC), developed and run by GSFC, the Mission Operations Center (MOC), developed and run by APL during spacecraft Integration and Test and flight operations, and NASA's Deep Space Network (DSN) ground stations. In addition to performing mission command, control, and monitoring functions, the MOC also includes a real time simulator for testing software upload and spacecraft scenarios. APL also has a navigation team that works closely with the MOC and DSN to perform orbit determination, predictions, and updates for the mission.

In addition to the primary flight and ground system deliverables, a number of Ground Support Equipment (GSE) systems are required to support development and testing deliverable hardware and software. These items, which include simulators, stimulators, and bench test equipment, are generally software-based subsystems in their own right.

4.2.3 STEREO APL Software Requirements

This section outlines the top level software requirements for the flight and ground based software systems.

Flight Systems. The STEREO flight segment consists of two identical spacecraft, each of which contains a number of programmable subsystems: a Command and Data Handling (C&DH) subsystem, a Guidance and Control (G&C) subsystem, and a suite of instruments some of which contain processors. The data system and instruments for each spacecraft are "single string" with no redundant or backup processors. Although the instruments may be powered off, the C&DH and G&C subsystems are mission critical and will be powered continuously throughout the mission.

This section discusses the C&DH and G&C subsystems, for which APL will develop the flight software. The requirements described below are identical for both spacecraft, and in fact the two spacecraft will contain identical software loads.

Command and Data Handling (C&DH) Subsystem. The C&DH subsystem provides support services to the spacecraft bus and the instruments.

- overall spacecraft safety via three spacecraft modes: operational, safe hold, Earth acquisition
- command and telemetry services:
  - CCSDS protocols
  - uplink: 125 bits/sec (normal); 7.8125 bits/sec (emergency)
  - downlink: variable rates up to 800,000 bits/sec during DSN passes
  - 500 bits/sec "broadcast" mode at other times
- data collection from spacecraft bus and instruments
- dissemination of spacecraft bus status to instruments
- spacecraft bus health and status monitoring
- solid state recorder management (7.5 Gigabits)
- non-coherent navigation support (Doppler count measurement and reporting)
- power management (peak power tracking)
- time tagged commands, macros and rule-based autonomy
- time maintenance and distribution
- software upgrade support for programmable spacecraft bus devices

Guidance and Control (G&C) System. The STEREO guidance and control system components consist of :

- processors and their associated software and interface electronics
- star tracker
- inertial measurement unit
- digital solar attitude detectors
- cold gas propulsion system
- reaction wheels

In addition to controlling these attitude system components, the G&C system is also responsible for autonomously controlling the communication system's High Gain Antenna (HGA) gimbal to keep the antenna pointed toward Earth.

G&C services support both the spacecraft bus and the science instruments. Reaction wheel speeds may be changed at any time to maintain required pointing accuracy without degrading science data. However, propulsion system firings to dump system momentum may interfere with science data collection for some instruments. Therefore in Operational propulsion system firings will be limited to short preplanned "spacecraft bus activity" time windows commanded from the ground via the C&DH.

The G&C system must use its sensors, processors, and actuators to support low level testing and commanding of individual components. In flight it implements high level pointing modes that require a closed loop dynamic model of the spacecraft, G&C subsystem, and the universe (with varying degrees of detail). These high level modes support the spacecraft bus and instruments by providing the following services:

- attitude safety (coordinated with C&DH's spacecraft safety function):
  - maintain overall attitude control
  - maintain battery charge
  - conserve cold gas propellant
- body axis control
- high gain antenna pointing
- momentum management
- high precision instrument pointing with closed loop feedback
- G&C subsystem status reporting and history maintenance

Ground Systems

Mission Operations Center. Housed at APL, the STEREO Mission Operations Center (MOC) will communicate with the two STEREO spacecraft via the NASA Deep Space Network (DSN), and with the STEREO science users through the Science Operations Center (SOC) at GSFC. It will provide the following services:

- command and telemetry
- maintains all command and telemetry definitions for spacecraft bus
- controls the flow of all commands to the spacecraft via the DSN
- generates commands for the spacecraft bus
- forwards commands for the instruments received from the SOC
- receives all telemetry data from the spacecraft
- forwards telemetry data to the SOC
- spacecraft bus health and safety monitoring
- time and navigation maintenance (the MOC and the navigation team at APL work together to produce time correlation and navigation data)
- spacecraft configuration management
- spacecraft activity planning
- real time spacecraft simulation

Integration and Test Equipment. The following ground support equipment items are used during subsystem testing and integration with the spacecraft

(1) IEM Testbed: tests the IEM by simulating its environment, including C&DH interfaces:
  - instruments
  - G&C subsystem
  - power system
  - uplink/downlink system
  - housekeeping inputs

(2) G&C Testbed: tests the G&C and its control algorithms by simulating its environment, including
  - star tracker
  - IMU
  - DSADs
  - propulsion system
  - reaction wheels
  - high gain antenna gimbal
  - C&DH subsystem
  - instrument-supplied pointing error signal
  - the universe

Note: the G&C testbed becomes part of the real time spacecraft simulator in the MOC after launch.

(3) Spacecraft emulator: simulates instrument interfaces to the spacecraft to test the instruments before delivery. A Phase A cost tradeoff study will determine whether to implement this item.

(4) “Mini-MOC”: a subset of the Mission Operations Center that allows other subsystems to be tested in the MOC environment during development. A Phase A cost tradeoff study will determine whether to implement this item.

4.2.4 Baseline Architecture

The architecture for all STEREO software, both flight and ground, is based on the software designed by APL for the TIMED mission. The largest differences in the flight software are in the hardware interfaces and requirements for the guidance and control software, and in the fact that the system is single string instead of fully redundant. The main ground software difference is that STEREO uses DSN ground stations instead of low Earth orbit terminals; for this reason the ground software that interfaces with the ground stations also borrows heavily from the APL NEAR mission.

Flight Software. Figure 4-3 shows the baseline hardware architecture for the STEREO spacecraft bus data system. The C&DH subsystem is based on a Mongoose V running the Nucleus+ real time operating system. It

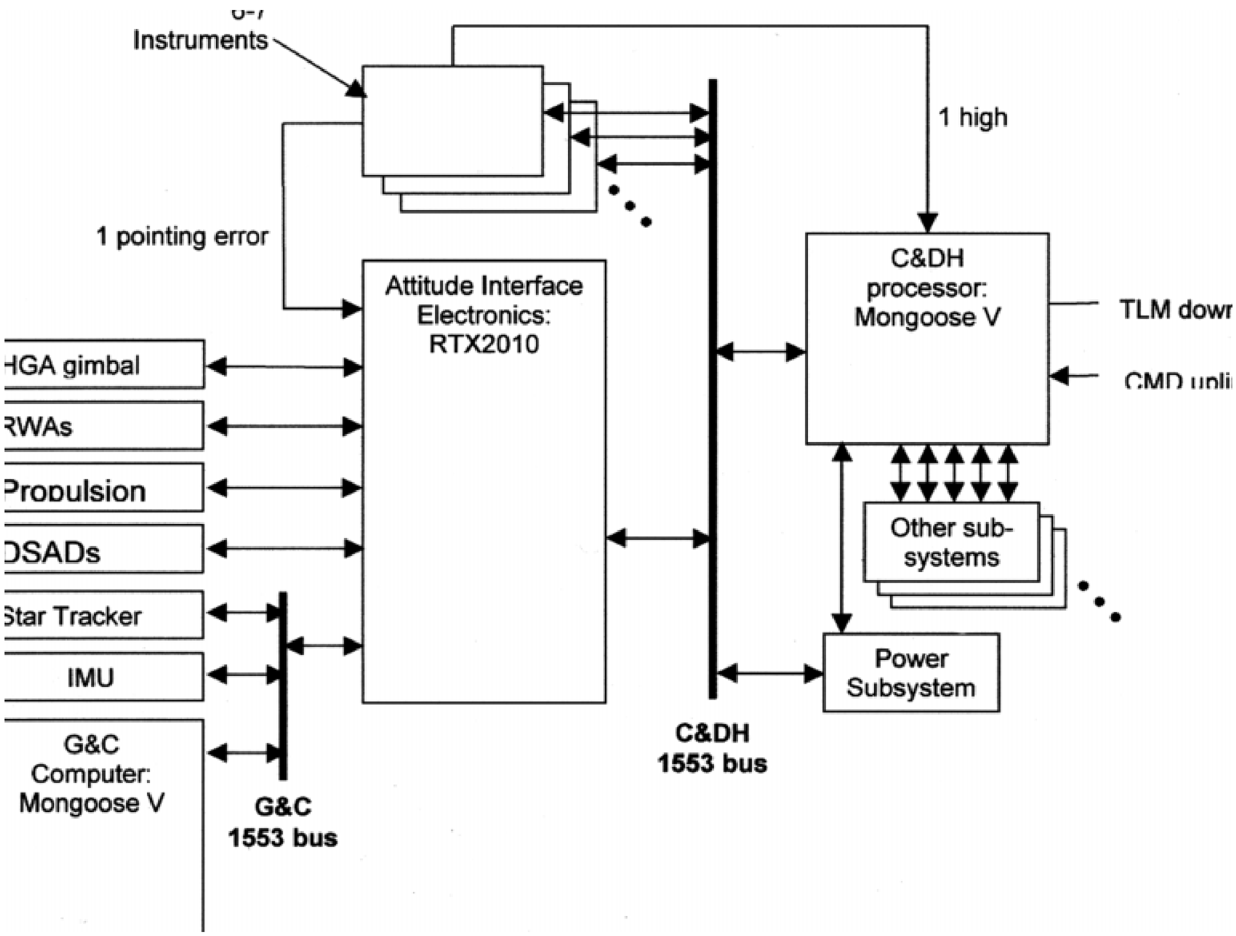


Figure 4-3 Baseline STEREO Data System Hardware Architecture

controls the C&DH 1553 bus which communicates with the instruments, the power system, and the G&C subsystem via the Attitude Interface Electronics (AIE). The AIE contains an RTX2010 processor and the hardware interfaces to all the guidance and control sensors and actuators. It controls the G&C 1553 bus that communicates with the star camera, the Inertial Measurement Unit (IMU), and the G&C Flight Computer. The G&C flight computer is a Mongoose V identical to the C&DH (also running Nucleus+), and runs the mathematical algorithms that control the spacecraft attitude, momentum, and high gain antenna gimbal angle. It receives its sensor inputs from the AIE, and sends its actuator output requests back to the AIE for implementation.

There are two key differences between the TIMED and STEREO architectures that drive significant software changes in all flight processors. First, TIMED uses the Global Positioning System (GPS) to determine time and navigation parameters directly, while STEREO will implement models that will be updated periodically from the ground. This change will eliminate software to interface with the GPS subsystem, but will change the software that manages and distributes the time and navigation parameters.

Secondly, the use of single-string hardware affects safing and software reloading. The safing process will be simpler in some ways because the option to reconfigure redundant hardware is eliminated. However, as a result the overall approach to safing will need to be revisited during the conceptual design phase. Likewise, TIMED achieves software reloads by running the spacecraft with one processor while reloading the other. A new approach will be needed to reload the single available processor while it continues to control the spacecraft.

The STEREO C&DH software will be very similar to the TIMED C&DH software, as both the overall software requirements and the hardware platform are similar. Table 4-2 lists the primary differences.

The TIMED flight software implemented guidance and control algorithms on both the Attitude Interface computer and the G&C flight computer, with "C" code generated automatically from simulation models. This worked well for the G&C flight computer, but was a problem for the AIE computer due to the limitations of the RTX2010 architecture, the compilers and other tools available to support it. The STEREO software will limit the AIE software to controlling the G&C hardware interfaces (without auto-code) while leaving guidance and control algorithms to the G&C flight computer. Reconfiguration of the G&C subsystem architecture is another option that will be studied during Phase A to further simplify the flight software.

In addition to this change based on a TIMED "lesson learned", the STEREO G&C software will also differ because of differences in mission requirements and hardware interfaces. Table 4-3 summarizes these differences.

4.2.5 I&T and MOC Software

The STEREO I&T and MOC software will also be based on the TIMED design, which is a combination of the commercial EPOCH-2000 product from Integral Systems, Inc. (ISI) and

Table 4-2 Differences between TIMED and STEREO C&DH Software

Feature	TIMED	STEREO	Software Impact
Redundant Hardware	Yes	No	Requires new software (S/W) loading approach
GPS	Yes	No	Requires new timekeeping S/W
Number of Instruments	4	6-7	More 1553 remote terminals to Manage
RS-422	No	Yes	New driver needed, high speed Input/Output (I/O)
Max Science Rate	55 kbps	450 kbps	Higher recorder rates
Max Downlink Rate	4 Mbps	800 kbps	Increased SSR management flexibility
Broadcast Mode	No	Yes	New S/W to collect broadcast data

Table 4-3 Differences between TIMED and STEREO G&C Software

Feature	TIMED	STEREO	Software Impact
Redundant Hardware	Yes	No	New S/W loading approach
GPS	Yes	No	New Orbit Determination S/W
Momentum Mangement	T-Rods	Prop	New I/O and Control S/W
Star Tracker, IMU, etc.	Known	TBD	Possibly new I/O S/W
High Gain Antenna	No	Yes	New gimbal I/O and Control S/W
Error Signal	No	Yes	New I/O and Control S/W
Control Frequency	10 Hz	TBD	TBD

custom enhancements developed by APL and ISI. Much of the software will be reused as is, but several requirement differences will drive differences in the software as Table 4-4 illustrates.

Certainly the change to a deep space mission with two spacecraft with some simultaneous ground contacts has operational impacts, but the core software system that delivers commands to the spacecraft, and receives, processes, and monitors telemetry will be largely unchanged. A larger impact to the software will be the change to DSN ground stations and from four distributed Payload Operations Control Centers (POCCs) to a single science interface through GSFC. The elimination of onboard GPS will also require more navigation and time-keeping support on the ground, and the non-coherent navigation system STEREO will use requires some additional software and interfaces with the DSN navigation team as well.

STEREO will use the same philosophy of decoupled spacecraft and instrument operations in general that TIMED uses, so that most spacecraft bus and science instrument operations can be carried out independently of each other. However, some coordination is clearly necessary in overall activity planning; the MOC software will include a Web-based "STEREO Data Server" to serve as a focal point for this coordination information. This server will be based on a similar TIMED server.

Ground Support Software. The following ground support items are used for subsystem testing during development, but represent significant software development efforts on their own. Note that decisions on whether to implement the spacecraft emulators and "Mini-MOC" will be the subject of cost/benefit trade studies during Phase A.

G&C testbed

- PC/NT-based
- connects to G&C subsystem via 1553 bus; accessible via Ethernet
- simulates G&C system components, C&DH, and environment
- allows real time closed loop tests of the attitude system
- becomes part of the real time spacecraft simulator after launch

C&DH testbed

- PC/NT-based
- connects to C&DH computer; accessible via Ethernet
- simulates C&DH interfaces and environment
- allows real time tests of C&DH hardware and software

Spacecraft emulator (if implemented)

- PC/NT-based
- accessible via Ethernet

Table 4-4 Differences between TIMED and STEREO MOC Software

Feature	TIMED	STEREO	Software Impact
Number of Spacecraft	1	2	Requires separate C&T databases, sorting of C&T by spacecraft and support of two simultaneous passes
Ground Station	APL	DSN	Changes to contact planning
Science Interface	4 POCs	1 SOC	TBD - Data product differences
GPS	Yes	No	Ground based Navigation S/W
Unsupported passes	Yes	No	Commitment to automated Ops

- connects to instruments via their spacecraft interfaces (1553, serial)
- emulates spacecraft functions that support instruments
- allows instrument checkout before spacecraft integration

Mini-MOC (if implemented)

- a subset of the MOC, including EPOCH
- available early, for use in G&C and C&DH subsystem testing
- can command and receive telemetry from both the subsystem under test and supporting GSE
- uses the same command and telemetry dictionaries, command procedures, and display pages as the MOC

4.2.6 Phase A Trade Studies

Phase A of the STEREO program will include the following trade studies, with cost reduction and cost risk management as the evaluation criteria.

G&C Processor Architecture. The G&C processor hardware architecture shown in Figure 4-3 is the baseline for STEREO. This basic configuration, inherited from TIMED, was in turn inherited by TIMED from the Near Earth Asteroid Rendezvous (NEAR) mission. However NEAR used a Honeywell 1750 processor for the G&C computer, and this part was not adequately radiation hard to be entrusted with all mission critical G&C functions. Therefore the very hard RTX2010 was put in control of G&C sensors and actuators, and implemented the most critical G&C safing modes.

Since the Mongoose V processor is as radiation hard or harder than the RTX2010, this study will revisit the assumptions that resulted in this architecture. It is possible that two processors are no longer necessary in the G&C subsystem, and/or that an off-the-shelf solution is now available. Software considerations will play a major role in evaluating the cost/benefit tradeoffs of this hardware architecture study.

Operating Systems and Tools. The Pre-Phase-A STEREO baseline is to use the Nucleus+ real time operating system on the Mongoose-based C&DH and G&C computers, and a simple APL developed real time kernel on the RTX2010-based AIE. All processors are to be programmed in the "C" language, with a minimum of assembly language where absolutely necessary. The G&C computer will run "C" code generated automatically from the Matlab^{TM}/Simulink Real Time Workshop, but the AIE will not.

This study will evaluate whether changing any of these baseline assumptions will be cost effective. This study is partly linked with the G&C processor architecture study, since availability of software and support tools is an important factor in choosing processors. In addition to evaluating real time operating systems and support tools, this study will also consider software development environment tools such as requirements trackers and software design and analysis tools.

I&T/MOC Software. EPOCH is the Pre-Phase-A baseline for the I&T/MOC core software. This study will evaluate whether other packages are available that would perform the EPOCH functions for lower overall cost, including the costs of modifying the additional software that APL has developed to work with EPOCH.

Mini-MOC and Spacecraft Emulator. These studies will evaluate whether it is cost effective overall to build "Mini-MOCs" and spacecraft emulators.

4.3 Guidance and Control (G&C) Subsystem

4.3.1 System Baseline

Figure 4-4 is a top-level block diagram of the STEREO G&C system.

Only the white-background boxes are actually part of the G&C system, the rest are in different systems but interface with and/or are controlled by G&C. The four G&C equipment items are

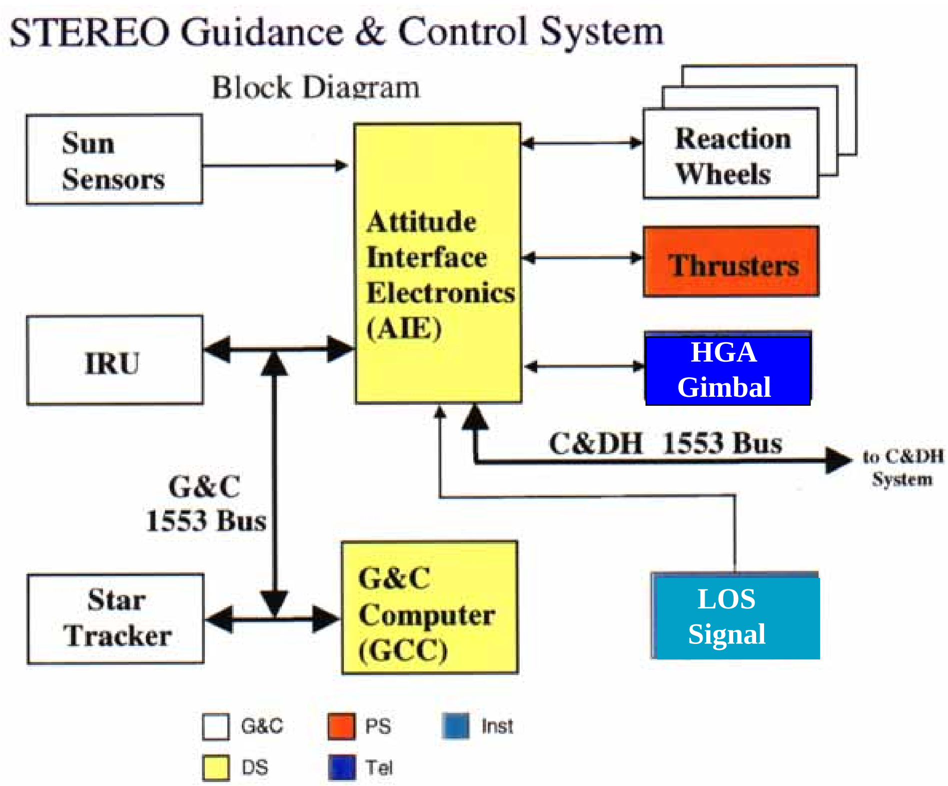


Figure 4-4 Guidance and Control Physical Block Diagram

described below. The baseline equipment selections are strawman candidates for resources (power, weight, cost, and performance) and budgeting purposes.

They are existing items with known characteristics which will meet the STEREO requirements. Actual equipment will be chosen in preliminary design via competitive procurements; better performance and/or cost is likely. Alternates exist to all these items, assuring minimal risk.

Algorithms in the Guidance and Control Computer (GCC) carry out most of the G&C control functions. The AIE distributes the signals and data between G&C equipment and interfaces. Some level of safe-mode control may be implemented in the AIE. Reaction wheels control spacecraft pointing. One instrument (assumed to be the SCIP) will provide pointing error signals in two axes (Loss of Sun (LOS) Signal in Figure 4-4). The cold-gas propulsion system will be used for momentum management, with firings scheduled as necessary (about once or twice per week). De-tumble (removal of excess angular rate) will be possible using thrusters. The HGA gimbal will be moved incrementally, in small steps appropriate for its pointing requirement. These will occur, typically about once per day, up to about 10 per day early in the mission.

Inertial Reference Unit (IRU). Baseline is the NEAR IMU from Delco Electronics (Litton), usingDelco 130Y Hemispherical Resonator Gyros (HRG). These gyros have rate bias stability < 0.001˚/hr, over 16 hr, and an angular random walk (ARW) less than <0.01˚/hr1/2. The NEAR IMU has redundant CPU and power supplies, with four gyros, any three of which provide 3-axis attitude reference. A single-string IRU similar to the one flying on Cassini would be the STEREO baseline. This IRU is extremely reliable, with no moving parts. A four gyro IRU has a projected system-function probability of success (P_S) of 0.9996 for mission life. Alternate gyros will be considered in preliminary design, including mechanical, ring-laser gyros (RLG) and Fiber-optic gyros (FOG), and the best candidate chosen.

Star Tracker. Baseline is the TIMED star tracker from Lockheed-Martin. This tracker has 3 arcsec accuracy (1s) in pitch and yaw (i.e., normal to its boresight) and 32 arcsec in roll. Sensitivity is 7.5 magnitude (Magnitude–Variable (M_v)) stars, and field of view (FOV) is 8.8° square. It includes autonomous star identification and attitude determination, with quaternion outputs at 5 Hz on the 1553 bus. Autonomous star identification can be achieved within ~2 sec. It has been flown on DS1 and P59, and is scheduled to fly on numerous missions including TIMED, EO1, Microwave Anisotropy Probe (MAP), and Imager for Magnetopause-to-Aurora Global Exploration (IMAGE).

Reaction Wheels. The NEAR reaction wheels from Ithaco, Inc. are baselined. NEAR used the Ithaco Type A wheels, and TIMED uses the larger Type B wheels. Both have a brushless DC motor, bipolar tachometer, and separate electronics, stacked to save weight and space. The Type A wheels have maximum angular momentum of 4 Nms (@ 5100 rotations per minute (RPM)) and maximum torque of 0.025 Nm. The Type B wheels have significantly higher momentum capacity and somewhat higher torque. Static unbalance is less than 1.5 gm cm, and dynamic unbalance is less than 40 gm cm^2, with torque noise Power Spectral Density (PSD) of less than 10^{-11} $(Nm)^2/Hz$, 0.1 to 1 Hz. Continuous operating life is specified as greater than four years. The NEAR wheels have been operating continuously now for over three years with no sign of degradation. The Small Explorer (SMEX) wheels (as used on Transitional Region and Coronal Explorer (TRACE) and Sub-millimeter Wave Astronomy Satellite (SWAS)) are an especially attractive alternate, with higher torque and significantly lower unbalance, both of which are desirable for low jitter. It appears likely that at least 0.14 Nm of torque should be available, in a reasonably sized wheel.

Sun Sensors. The Adcole Digital Solar Attitude Detector (DSAD) system is baselined. This system is small, very mature, and flight proven, many times. It has five detector heads, each of which measures 2-axis Sun vector in a ±64˚ FOV. Accuracy is 0.5˚quantization, with bit transition-angles calibrated to 0.25˚ accuracy. A fine Sun-sensor system is also available from Adcole, and is under consideration either in addition to or as an alternate to the standard DSAD system.

4.3.2 G&C Functions

Figure 4-5 is a top-level functional flow diagram of the G&C system.

In this diagram, the upper blocks are the system being controlled (color-keyed System; with outputs on right or bottom). These are spacecraft hardware components, except for dynamics which can be considered hardware plus Physics. Other blocks map to software modules in the GCC. The white (or unshaded) boxes are in the foreground ("Fast") or inner control loop, which will run at 10 to 25 Hz. Gyros (Inertial reference unit, IRU) are the primary short-term attitude reference, and are corrected for drift and other errors by signals from the background ("Slow") loop and/or the SCIP. A software switch is shown to indicate that Guidance, Navigation, and Attitude Estimation will not be used in Safe Mode. Gyro data is used with Sun sensors and star tracker for attitude determination. Thruster control will ordinarily only be used for momentum management, not for pointing.

A top-level diagram such as this forms the starting point for system model development in Simulink™, a graphical simulation

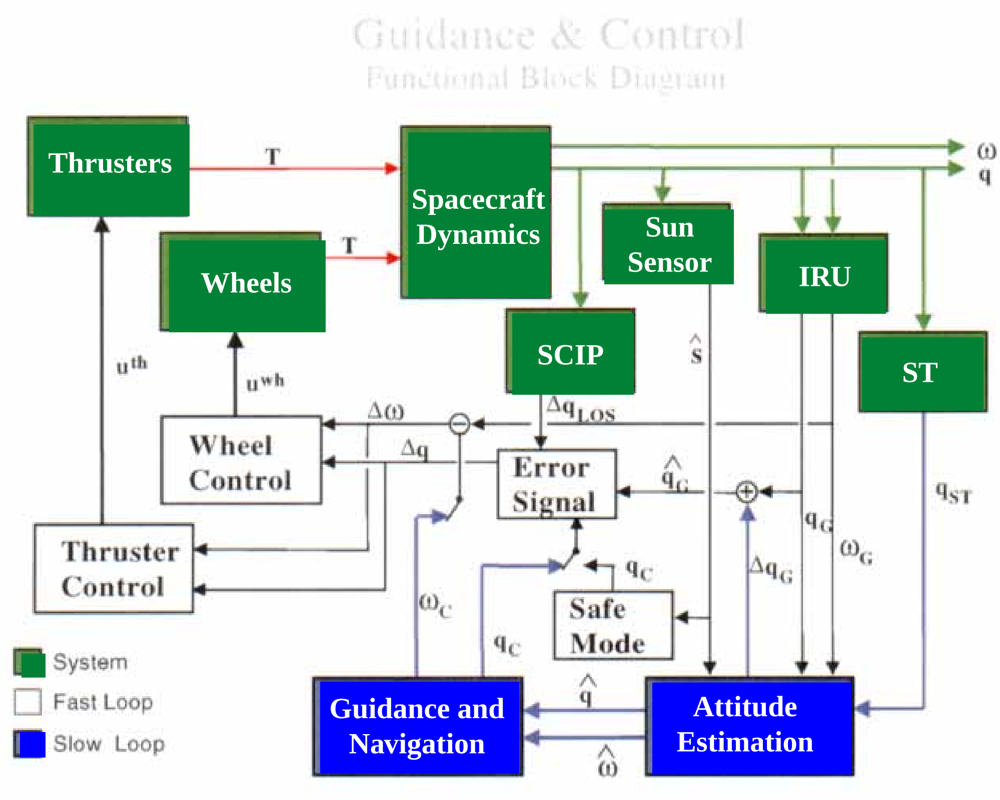


Figure 4-5 Guidance and Control Functional Block Diagram

development tool which is part of the Matlab™ software suite. In Simulink™, individual blocks can be opened and expanded, to as many levels of detail as needed, until a high fidelity model of the system is developed. This model can then be exercised directly from the graphical environment, greatly facilitating design tradeoffs. Once the models are sufficiently mature, an additional Matlab™ tool called Real-Time-Workshop™ (RTW) can be used to on part or all of the model to produce code which is directly useable as flight code in the GCC, and for code for Integration and Test (I&T) simulations. Simulink™ and RTW have been successfully used to develop simulations and ultimately flight and I&T code on TIMED.

4.3.3 Requirements Summary and Discussion

Design Drivers. The principal requirements driving the G&C system design are for precise spacecraft pointing and jitter control. Pointing requirements are summarized in Table 4-5.

Jitter is the most challenging requirement. In order to meet the requirement for low jitter, a high

Table 4-5 Pointing Requirements

Spacecraft Pointing Requirements (3 σ)	Roll	Pitch/Yaw
Knowledge: Control: Jitter:	±20 arcsec ±0.1° 30 arcsec RMS	± 0.1 arcsec ±20 arcsec 1.5 arcsec (0.1 to TBD Hz)

control bandwidth is desirable. This in turn drives the design toward high wheel torque and a fast sampling rate. Additionally, disturbances will need to be minimized. Modern control techniques can also play an important role. These tradeoffs are discussed in some more detail below.

Other Significant Requirements

(1) Point LOS within 5 arcmin of Sun for SCIP acquisition—this will require good co-alignment of all instruments and G&C sensors with the SCIP.

(2) Nominal HGA pointing to 0.1°; Maintain HGA pointing during thrusting—The HGA gimbal angle will be changed in steps, not continuously. This requirement sets the gimbal step frequency. A small thruster impulse bit and small on-time will be needed. In-flight HGA alignment calibration may be necessary to meet the 0.1° requirement. On-board ephemeris will be needed for HGA pointing vectors.

(3) Complete autonomous thruster firings within 300 seconds—this will require that momentum dumping be fairly aggressive. Any autonomous use of thrusters will have to be very carefully designed, verified, and tested. A very gradual, conservative momentum dumping approach is safest and preferred if possible.

(4) Momentum storage capacity at least four days in Operational Mode—This directly sizes the wheel momentum.

(5) Return from any attitude in less than 12 minutes—It is assumed this time limit starts after any detumble is complete, i.e., is the maximum allowable time for a rest-to-rest 180° slew. This requirement may size wheel torque, although a high wheel torque is desirable anyway for control bandwidth. If the time limit is strictly enforced and proves difficult to meet, this requirement may force consideration of thruster attitude control.

(6) Solar pressure momentum bias within Sun-angle limit—use of solar pressure for momentum management can only be done with some kind of center-of-pressure trim devices (flaps), due to the 0.1° overall pointing constraint. Solar pressure torque was used successfully on NEAR for momentum management, but required off-Sun pointing of several degrees.

4.3.4 Jitter Considerations

The instrument requiring fine-pointing control is assumed to be the coronagraph (SCIP). For the purposes of this study the SCIP has been assumed to be accurate to 0.1 arcsec (3σ) and supplied at 10 Hz. This instrument will set the control and knowledge requirements for the entire spacecraft. The assumed jitter requirement placed on the G&C system by the SCIP is:

Roll:	30 arcsec Root–Mean–Square (RMS),
Pitch/Yaw:	1.5 arcsec (0.1 to TBD Hz)

Pointing control of other instruments will essentially the same as for the SCIP, except for low-frequency jitter. Knowledge for the other instruments will differ from the SCIP mainly by the misalignments.

To meet this requirement, the control bandwidth (BW) must be as high as possible consistent with the baseline design of the spacecraft, instruments and nominal pointing requirements. As a goal it will be 1 Hz, though it may not be possible to make it that high. Limiting factors are wheel torque and linear range of fine pointing control. The NEAR spacecraft can be used for comparison (possessing similar weight, inertia and quiescent pointing). Using the NEAR reaction wheels and current STEREO inertia estimates, a linear range of 0.1° would give a BW of about 0.5 Hz. With the SMEX wheels (0.14 Nm torque) 1 Hz BW can be achieved with a linear control range of about 0.2°, which is probably workable for a 0.1° overall pointing requirement.

We can assume for now that the control system will be able to virtually eliminate the effects on

SCIP pointing by disturbances at frequencies less than BW, down to the pointing control requirement. However, other instruments will experience pointing errors relative to the SCIP at all frequencies. It's not possible to accurately quantify these errors at this stage of design. Typical experience with similar spacecraft (e.g., NEAR) suggests that jitter due to structural non-rigidity will be on order of 10 mrad, at the disturbance frequency. There are low frequency disturbances (e.g., from thermal distortion) and higher frequency disturbances from onboard equipment, notably the reaction wheels and moving mechanisms in other instruments and subsystems. All these issues will have to be studied in some detail before meaningful quantitative estimates can be made. The technology to deal with jitter at reasonable levels is available, but not without resource impacts (cost, schedule, and weight).

All instruments including the SCIP will experience disturbances at frequencies above the BW. Only minimizing the disturbances themselves, through careful mechanical design can control the effects of these disturbances on the instruments.

The Radio Burst Tracker (RBT) will also affect jitter, possibly significantly. The booms should in general be as short, light, and stiff as possible. It is also desirable that they have as much inherent damping of bending modes as possible, although typically such booms don't have much damping and are often modeled as (nearly) undamped. If the booms have bending frequencies less than BW, the control system should be able to control those modes, or at least limit them to levels consistent with the control performance (notional 0.1 arcsec). If excited, frequencies higher than the BW will not be damped actively and will contribute to jitter for all spacecraft. Although this may seem to suggest that low frequency is desired, in general stiff booms are preferred because of the response's roll-off with frequency. Since low-frequency disturbances can only imperfectly be controlled, it may be better to have high frequency disturbances that are small enough to ignore.

4.3.5 Flexible Spacecraft Modeling and Jitter Study

Spacecraft with several flexible attachments has been modeled in Matlab™/Simulink™ to study the jitter effect. The flexible structures modeled include two solar panels, three RBT booms and one magnetometer boom. Each flexible attachment is modeled as an uniformly distributed beam-like structure, based on preliminary structure parameters. The dynamics of flexible spacecraft is described by the following equations.

$$I\ddot{\theta} + M_I\ddot{\xi} = N - \dot{h} - \dot{\theta} \times (I\dot{\theta} + h + M_I\dot{\xi})$$
$$\ddot{\xi} + D\dot{\xi} + \Lambda\xi + M_I^T\ddot{\theta} = 0$$

where:

θ— rotation angle of spacecraft;

ξ— modal coordinates of flexible structures;

$M_I\xi$— interaction matrix between flexible structures and rigid body of spacecraft

$M_i\dot{\xi}$— total momentum from flexible structures;

$M_i\ddot{\xi}$— total acceleration from flexible structures;

$D = 2k \times diag\{\omega_{f1},...,\omega_{fN}\}$ — natural damping matrix of flexible structure;

$\Lambda = diag\{\omega_{f1}^2,...,\omega_{fN}^2\}$— stiffness matrix of flexible structure.

The first equation is the flexible spacecraft equation of motion, which shows that the interaction from flexible structure to spacecraft rigid body is through momentum and acceleration of flexible structure deformations. The second equation describes the dynamics of flexible structure which is driven by the rotational acceleration of rigid body.

The coefficient matrices in flexible structure dynamics can be obtained through structure analysis using finite element methods. Since we do not have them now, we modeled the flexible attachments with the following assumptions:

- Normalized modal model, with lowest frequency $f_0 = \frac{1}{2\pi}\sqrt{\frac{3EI}{(M+0.243\rho l)l^3}}Hz$, and $f_i = i ? f_0, i = 1,2,...N$, based on preliminary parameters.
- Simplified interaction matrix, based on preliminary geometry of flexible structures
- Uncoupled flexible structure models
- No external force on flexible structures

Various simulations have been done to study the jitter effect on the attitude of spacecraft: mainly focusing on the jitter effect excited by firing thrusters about X, Y and Z axes respectively. Simulations are set up as firing thrusters about one axis only in each simulation, when spacecraft is in stable status (controlled by wheels). Thrusters fire once (10% duty cycle) per second consecutively for 20 seconds. The dynamics of angular rate of spacecraft, pointing error, flexible momentum and acceleration are studied based on simulation results.

A few observations from the simulations are summarized as follows:

(1) The axis which thrusters fire about is affected most, however, the other two axes react too. This is caused by the coupling interaction from flexible attachment: the booms are not deployed symmetrically.
(2) The pointing error on the axis which thrusters act on can be quickly decreased by wheel control to within the jitter requirement and the point errors on other two axes are much smaller than jitter requirement during thruster firing.
(3) Although there is no controller specifically designed to suppress flexible deflection used in the simulation, the regular wheel controller handles the jitter effect in the simulation fairly well.

4.4 Power Subsystem

4.4.1 Power Subsystem Requirements

The STEREO power subsystem is required to provide power to all spacecraft loads for the duration of the two-year mission. The bus voltage is required to be in the range of 22 to 35V, with load shedding occurring at 25V. The full load compliment with all instruments on shall be supported in solar-only operation. Battery discharge is permitted to accommodate load transients associated with attitude and propulsive maneuvers. (Ref. Margins Section)

The power system electronics for each spacecraft shall be identical, and shall be designed so as not to require continuous ground intervention for normal operation. The electronics shall regulate power flow to the bus to maintain power to the loads and safeguard the battery. The power system electronics shall provide ON/OFF power to the spacecraft loads, including the individual instruments.

The solar array is required to support the full load compliment (excluding attitude and propulsive load transients) for the two-year mission life while pointing within 5° of the Sun. The solar array shall be sized to allow operation at solar distances of 0.85 to 1.03 Astronomical Unit (AU) for the leading spacecraft, and 0.99 to 1.18 AU for the trailing spacecraft. The array shall meet aforementioned power requirements after a 1-MeV electron radiation exposure equivalent to 1.4 e+14 e/cm^2 (with a 6 mil CMX coverglass). The solar array shall require no gimbals, nor intra-panel hinges. The array shall be tolerant of partial shadowing. A surface ESD requirement for the solar panels is currently in definition. The solar array layout is permitted differ between the two spacecraft, but each array shall have identical harness and mechanical interfaces.

The battery shall not restrict the launch window, and shall not require reconditioning later than 28 days prior to launch. The battery shall provide power to the loads from three minutes before launch until 10 minutes after Radio Frequency (RF) acquisition without reliance upon incidental solar power. The battery shall provide power to the loads during load transients associated with attitude or propulsive activities, and shall provide emergency power in the event of temporary attitude loss or load fault.

4.4.2 Power Subsystem Design

System/Electronics. The STEREO power subsystem has a peak power tracker (PPT) architecture with an unregulated bus. This design has direct heritage from the TIMED program. The battery is connected directly to the main bus, maintaining an unregulated bus voltage of 22 to 35 volts. The power system electronics (PSE) regulate the flow of power from the solar array by controlling of a bank of PPT modules between the solar array and main bus. The PPT modules are buck converters whose output voltage is set by the bus voltage. The input voltage, or array operating voltage, is set by pulse width modulation of the converter. The PSE implements three concurrent control loops to manage the array power with the PPT modules:

(1) A PPT algorithm is processed in the IEM, which sets the array voltage to operate at its maximum efficiency. This algorithm uses solar array current and voltage telemetry to measure array power. The Pulse Width Modulation (PWM) is incrementally adjusted to dither about the operating voltage that produces maximum array power. This control loop only dominates the PPT control when the combined load requirement and battery charge limit exceed the array power capability.

(2) A current control loop monitors the battery charge current and adjusts the PPT to restrict power flow from the array to maintain battery charge current to one of two preset limits. A high rate charge limit is used to recharge the battery after a discharge. A trickle rate charge limit is used to maintain a full battery again self-discharge.

(3) A temperature compensated voltage control loop monitors the battery voltage and temperature to prevent overcharge of the battery. The control loop reduces charge current to maintain the battery at one of 8 ground-selected NASA voltage-limit curves. The voltage control loop is expected to dominate the PPT control the majority of the mission time.

The battery is further protected against an over-temperature condition by spacecraft autonomy which will automatically reduce the battery charge limit to trickle if the battery over-heats.

The PSE also houses the power switching functions for the spacecraft. A dedicated TIMED-heritage serial command bus conveys relay commands from the IEM. Commands are decoded and sent to the appropriate relay card. Relays are controlled via a functionally redundant control matrix to turn power ON or OFF to individual loads. The relay cards are also equipped with shunts to measure load current. Current telemetry on each relay card is multiplexed and amplified to a 0-5V signal. Relay card telemetry is then multiplexed by an analog mux card and digitized with a 12-bit Analog to Digital (A/D). The resulting telemetry is then forwarded to the IEM via a 1553 interface. Relay tell-tales are similarly sent to the IEM via three-state buffers which are polled by the analog mux card to forward the tell tale telemetry over the 1553 interface. Load fusing is located in the PSE using NEAR/TIMED heritage fuse plugs to facilitate I&T.

Solar Array. The STEREO solar array consists of two co-planar panels totaling 3.35 m^2 in area. The panels are populated with GaAs cells of a to be determined size with 6 mil CMX cover glasses. Cells are connected into serial strings, each with diode isolation. Panels are sized to fit within volume constraints of an Athena launch vehicle, and provide 35% margin in solar only operation to the full load contingent at mission aphelion at End of Life (EOL).

Battery. The STEREO battery baseline is a 21-Ah Super-nickel cadmium (S-NiCd) battery. S-NiCd technology is well suited for lengthy stand-by use of an interplanetary mission as has been well demonstrated by the NEAR spacecraft. The battery assembly is a build-to-print SWAS design, with possible minor wiring modifications to suit STEREO power electronics and telemetry. Battery reconditioning need not be performed within 28 days of launch, though it is strongly desired that final reconditioning occur within 14 days of launch. The battery size was selected to provide power to the loads during an Athena launch. The 21-Ah S-NiCd is sufficient to provide load power during launch for 95% of candidate Athena launch scenarios to a maximum depth of discharge of 55%. As launch scenarios are refined, battery sizing requirements shall be revisited.

Reliability. The power system is fault resistant in concert with the IEM's execution of the spacecraft autonomy algorithm. All control circuits are A/B redundant, as are all power switching/telemetry circuits except the relay matrices, which are functionally redundant. PPT modules are sized such that five of the six modules can meet mission power requirements. The battery is capable of meeting mission requirements with the failure of one cell, and the solar array is capable of meeting mission requirements with the failure of one string.

4.4.3 Make vs. Buy Decisions

All power electronics shall be designed and fabricated in-house by APL. All electronics designs derive strong heritage from the TIMED program currently finishing fabrication. Environmental test of the electronics shall be done in-house.

The battery is a build-to-print subcontract. Thermal vacuum and vibration testing of the battery assembly shall be subcontracted to the battery vendor. Battery fabrication and test is expected to be approximately 15 months in duration.

The solar array substrates will be designed in house, with subcontracted fabrication. Solar cell fabrication and laydown shall be subcontracted. Thermal vacuum testing of the panels shall be subcontracted to the solar array vendor. Solar panel fabrication and test is expected to be approximately 12 months in duration. Vibration and acoustic testing shall occur on the spacecraft level.

4.4.4 Trade-Off and Study Areas

S-NiCd vs. Lithium Ion Battery. Lithium-ion battery technology has recently made significant advances and is beginning to demonstrate flight suitability. Use of this technology can result in significant mass saving. Additionally, the space shuttle launch option will result in long periods in which the battery cannot be trickle charged prior to deployment. The low self-discharge rate of lithium ion technology is well suited for this prolonged open circuit stand without requiring a pre-deployment top-off.

Single Junction vs. Multi-junction Photovoltaics. Given that the array size is currently constrained by the Athena fairing, allowable load growth is limited if margin is to be maintained. Use of multi-junction cells could offer an improvement of 10% to 20% in array power within the same panel size. Current vendor estimates indicate that the cost per watt of power is roughly equal between the two technologies. System and program evaluation is required for final selection.

PPT vs. DET Architecture. While each of these architecture options has its own advantages and disadvantages, neither is strongly superior to the other for the existing mission. The baseline PPT architecture has been selected primarily because of its cost advantage from its strong TIMED heritage. However, evolution of requirements could eventually favor direct energy transfer (DET) if mass savings, bus voltage regulation, or a desire to isolate the battery from the bus becomes more compelling. The possible selection of lithium ion technology would likely favor a DET architecture to isolate the battery from the bus.

Athena vs. Shuttle Launch. While not directly a power subsystem trade-off, the selection will likely impact the power subsystem design. Should a shuttle launch be selected, the battery will need to remain in an open circuit state for a lengthy period prior to deployment. With the S-NiCd battery baseline, a brief discharge and top-off charge via the shuttle arm will be required prior to deployment. This scenario will likely have operations and hardware impact. Options are being evaluated which include the addition of circuitry on either the spacecraft or the shuttle to control the top-off, or changing the battery technology to Lithium-ion to achieve a favorable self-discharge characteristics to eliminate the need for a top-off.

Indium Tin Oxide (ITO) Coating. The new requirement for ESD cleanliness to 1V has initiated review of the option to coat solar array coverglasses with a conductive indium-tin oxide coating to reduce accumulation of charge.

4.5 Telecommunications Subsystem

During normal operations, the telecommunications subsystem is required to provide 200 kbps data downlink for the two year mission. The data are transmitted to the existing DSN 34m Beam Wave Guide (BWG) antenna system, supplemented by the 34 m High Efficiency Front-end (HEF) and 70 m systems, when necessary. The telecommunications subsystem provides for simultaneous X-band uplink, X-band downlink and tracking.

To save mass and cost, an innovative approach pioneered by the APL Space Department and scheduled to be flown on the TIMED spacecraft will be used. Transponders have been replaced by simpler RF transceiver based telecommunications cards that plug into the spacecraft Integrated Electronic Module (IEM). The TIMED S-band cards will be modified for X-band operation on STEREO.

Figure 4-6 is a preliminary block diagram of the telecommunications subsystem. The uplink and downlink RF cards reside in the IEM; all other RF components are external to the IEM. An APL built Ultra-Stable Oscillator (USO) provides the frequency reference. Presently, the baseline spacecraft design uses a 40 w Travelling Wave Tube Assembly (TWTA) to provide the power amplification. The antenna switching assembly consists of four switches. In phase A, switching assembly and cabling alternatives will be investigated.

A 1.1m gimbaled high gain dish is used for the high rate data downlink when the Sun-probe-Earth (SPE) angle is between -5 and 115˚. For normal operation, the spacecraft is oriented about the Sun-pointing axis toward Earth within ±0.1˚ for maximum antenna gain. The 1.1m dish is the maximum that can be accommodated by the Athena II launch vehicle. We investigated replacing the gimbaled dish with a phased array and distributed amplifiers, but concluded that the DC power requirement could not be accommodated. During phase A, we will canvass the antenna industry to identify other options for increasing the reflector, such as deployable antennas. The ‘front’ low gain antenna and the ‘normal mode’ medium gain antenna (Figure 4-6) are used during the early part of the mission for the leading spacecraft when the high gain

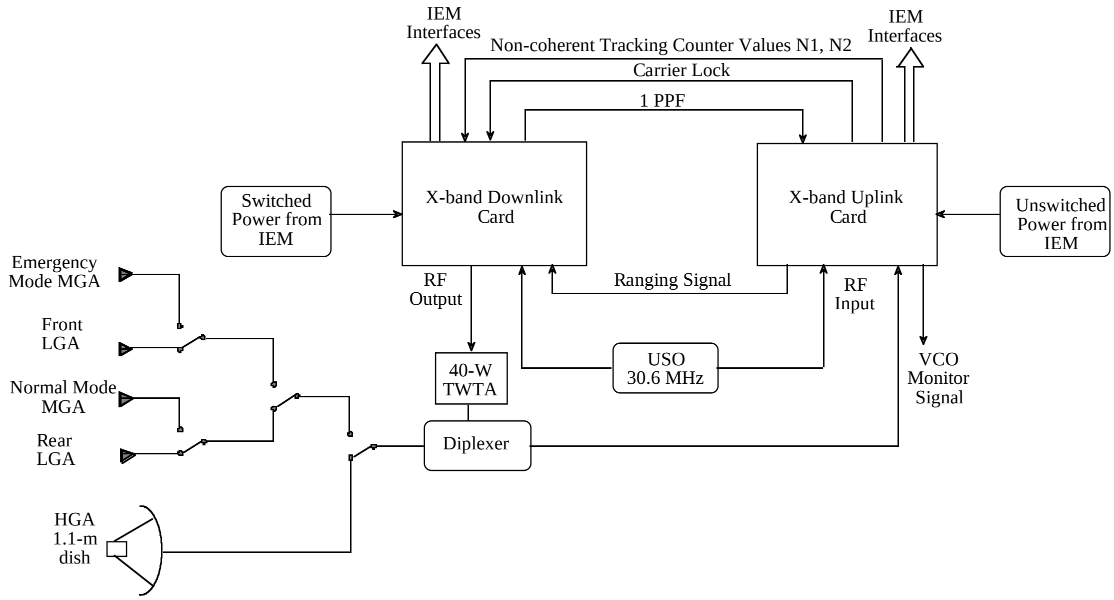


Figure 4-6 Preliminary STEREO X-band RF Telecommunications Block Diagram

antenna is unavailable. The 'front' low gain antenna will also be used for a few days after launch by the lagging spacecraft. The front and rear low gain antennas also serve for emergency uplink communications for the two-year mission, providing coverage in all directions.

The emergency medium gain antenna is used to establish downlink communications. All antennas are selected autonomously by the spacecraft or through ground command. The front and rear low gain antennas are so named based on their location on the spacecraft.

For the purpose of the pre-phase A study, the performance of the NEAR low gain and fan-beam antennas have been used in all calculations to estimate the downlink capability. The antenna designs will be determined in phase A and optimized, based on our antenna design experience, for the gains required by the STEREO trajectories.

The link performance for a number of trajectories has been analyzed, prior to selection of the baseline. The baseline is a leading spacecraft at 20°/year and a lagging spacecraft at 28°/year. The mission life is two years with the leading spacecraft launched 60 days prior to the lagging. Therefore the analysis has been performed for 792 days for the leading spacecraft and 732 days for the lagging spacecraft as shown in Table 4-6. The performance results presented are preliminary since the antenna gain, all spacecraft losses and link losses are estimations. These will be further refined in phase A for a day-to-day trajectory analysis.

Table 4-6 Range at Required Mission Life

	Years	Days	Range (AU)
Leading spacecraft	2+60 days	792 (requirement)	0.71
	5	1890	1.57
Lagging spacecraft	2	732 (requirement)	1.03
	5	1830	1.98

The emergency uplink rate is 7.8125 bps and emergency downlink rate is 10 bps. The high power DSN HEF 34 m system will be used for the uplink and the 70 m for the downlink. Rate 1/2, k=7 convolutional coding plus RS will be used for the downlink. The up and down links will have $\geq$3 dB margin ($\geq$6 dB goal on uplink) and Bit Error Rate (BER) <$10E^{-6}$. The spacecraft Low Gain Antennas (LGA) will provide coverage in all directions and support the emergency uplink for the two year mission. The emergency Medium Gain Antenna (MGA) is used to establish downlink communications based on the NEAR concept where the spacecraft transmits a beacon signal through the emergency MGA and is rotated about the Sun line to sweep through the Earth direction. When the signal is detected, commands can be sent to stop rotation and trouble-shooting can commence. Jet Propulsion Laboratory (JPL) is implementing a 70 m uplink capability which will extend the use of the LGAs.

Because the spacecraft is not redundant, the luxury of one receiver continuously connected to an omni-directional antenna is not available. The STEREO spacecraft will autonomously switch to the low or medium gain antenna (depending on distance) if there has been no communications from the ground for a pre-determined length of time. For example, the Advanced Composition Explorer (ACE) spacecraft switches between two sets of antennas every pre-selected number of days if no uplink is received.

The normal high data rate science link uplink rate is 125 bps and downlink rate is 200 kbps (to obtain a total of $\geq$5 Gbit per 8 hr DSN contact). The existing DSN 34m BWG system will be used for the uplink and downlink, supplemented by the 34m HEF and the 70 m when required. JPL is planning to upgrade to the 34m BWG system so the BWG system performance is similar to the 34 m HEF. All calculations assume the existing BWG performance (for a 2002 launch). Rate 1/6, k=15 convolutional coding plus RS will be used for the normal downlink. During the early part of the mission, when the data rate capability is very high (>300 kbps), R=1/2, k=7 convolutional coding plus Reed Solomon (RS) will be used for compatibility with the DSN stations. The link has $\geq$3 dB margin and BER <$10E^{-6}$.

One of the major considerations of the HGA design is the complex structure around the antenna, which can become scatterers because of the significant illumination from the antenna. The resulting scattered fields can interfere with the direct antenna radiation and cause degradations in the antenna directivity and sidelobe levels. The HGA field of view is clear of obstructions from 0° to 90°. As the HGA is gimballed larger than 90°, the spacecraft structures impinges on its field of view and reduces its gain. We have used the Ohio State University (OSU) reflector code to estimate the antenna performance when the reflector is blocked by the spacecraft structure. This program uses an extended aperture integration technique to calculate the pattern of the antenna without any blockage. The same technique is used to calculate the farfield pattern of the blockage due to the structure separately. Subtraction of the two patterns yields the pattern of the antenna with blockage. When the HGA is in its extended orientation, from 90 to 115°, the gain is reduced by approximately 2 dB. It is important to verify the gain reduction when the antenna in this orientation. In phase A, we will develop a mockup of the spacecraft/antenna structure in order to verify the OSU reflector code calculation. Beyond 115°, the dish physically interferes with the structure.

During the times data is not downlinked through the HGA to DSN for the primary STEREO mission, data will be ‘broadcast’ at 500 bits/sec at X-band to undefined ground stations. In phase A, the possibility of using commercial ground stations will be evaluated. Such stations receive a downlink for a customer and distribute the data as required by the customer. An optional 500 bits/sec S-band broadcast to a National Oceanographic and Atmospheric Administration (NOAA) ground station using the HGA has also been analyzed and the results given later in this section.

4.5.1 Lagging Spacecraft

Figure 4-7 gives the range and Sun-probe-Earth (SPE) angle for the 28˚/year lagging spacecraft. Note that the SPE remains <115˚ so that the HGA is usable throughout the mission (after post launch checkout on the low gain antenna). The range at the two year mission life is 1.0 AU.

Figure 4-8 shows the link performance for the lagging spacecraft. Table 2 shows that the link will support 200 kbps with the 34m BWG to ~day 603. A number of options are then available:

(1) From ~ day 603 to ~661 use 34m HEF (if 34m upgraded BWG is not available). From ~ day 661 to ~732 use 70 m DSN system. (cost)

(2) Increase transmitter power to 110 watts (this impacts cost, dc power, thermal).

(3) Increase HGA size to 1.9 m (there are mechanical limitations with present launch vehicle, increase is possible with shuttle).

(4) Accept lower link margin (and impact on risk).

(5) Accept lower bit rate.

4.5.2 Leading Spacecraft

Figure 4-9 gives the range and Sun-probe-Earth (SPE) angle for the 20˚/year leading spacecraft. The SPE is >115˚ from day 1 to 125. After ~day 125, the high gain antenna can be used. The SPE has two peaks; at day 1 SPE=167˚ and at day 55

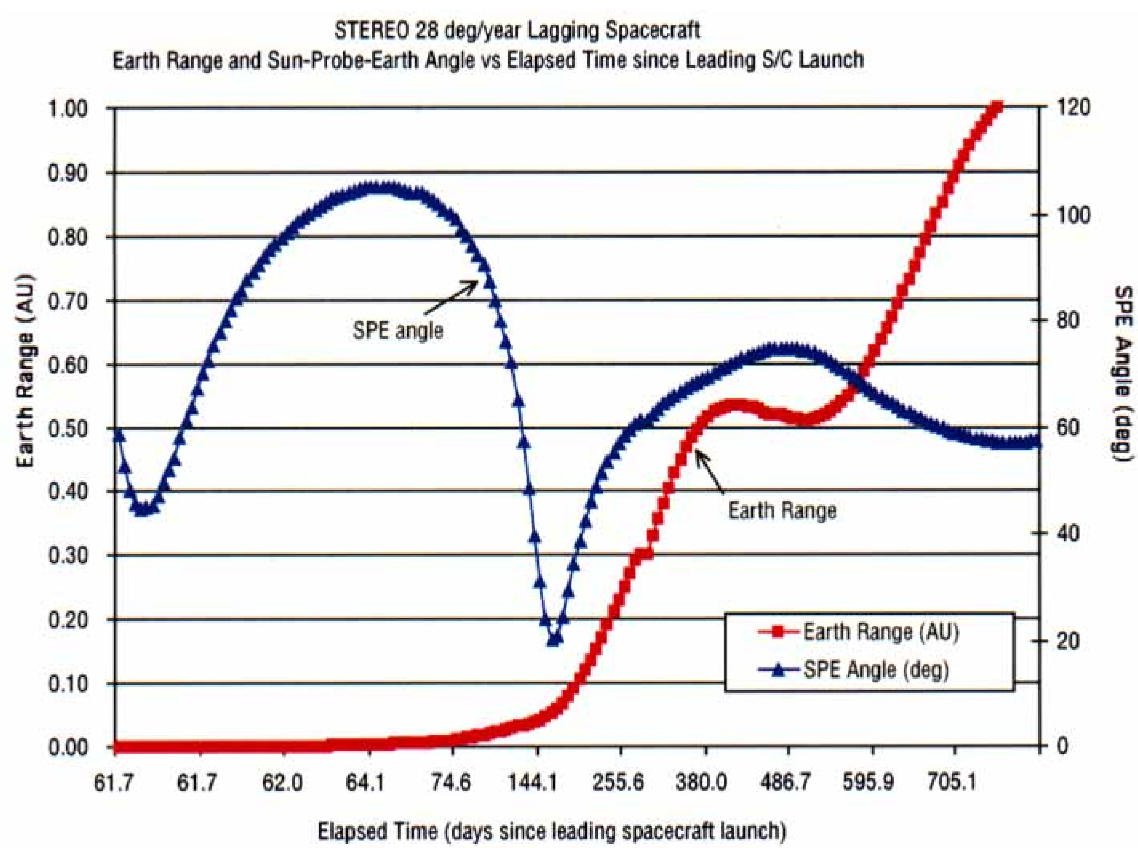


Figure 4-7 Lagging Spacecraft Earth Range and SPE Angle vs. Elapsed Time

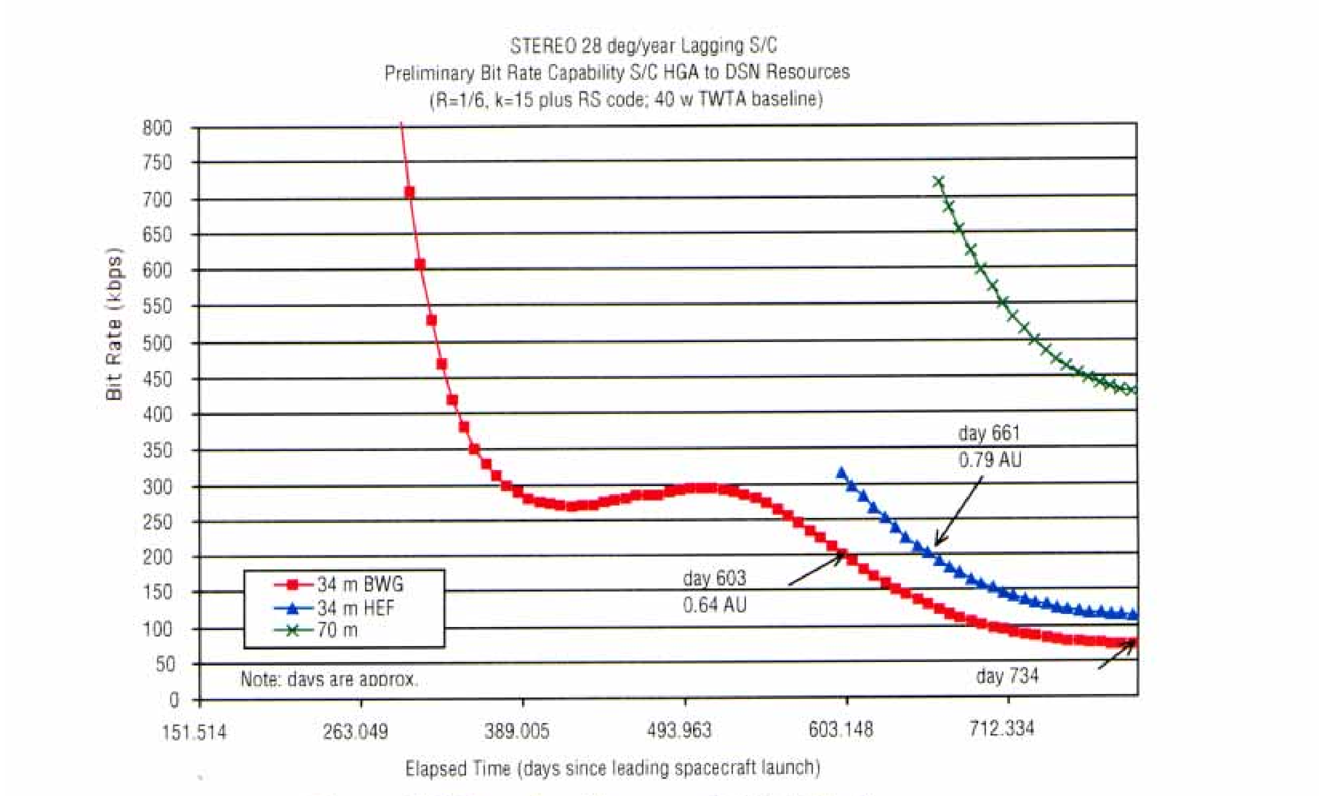


Figure 4-8 Lagging Spacecraft Link Performance

Table 4-7 Preliminary Bit rate Performance Summary for 28°/yr Lagging Spacecraft (Baseline is 40 w TWTA; 1.1m HGA)

DSN antenna system	S/C HGA (m)	TWTA (w)	~Day since launch	Earth Range (AU)	Bit Rate (kbps)
34m BWG	1.1	40	<603	<0.64	≥200
34m HEF	1.1	40	603–661	0.64–0.79	297–200
70 m	1.1	40	661–732	0.79–1.0	>300
34m BWG	1.1	110	1-732	≤1.0	≥200
34m BWG	1.9	40	1-732	≤1.0	≥200

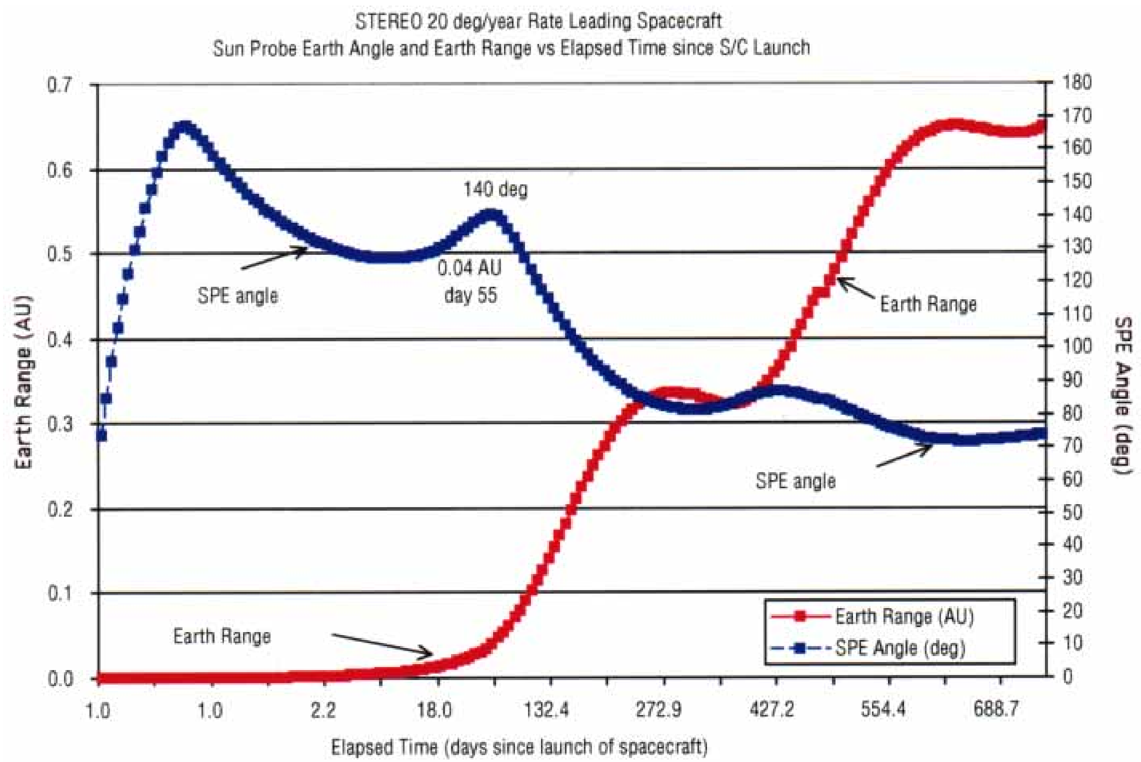


Figure 4-9 Leading Spacecraft Earth Range and SPE Angle vs. Elapsed Time

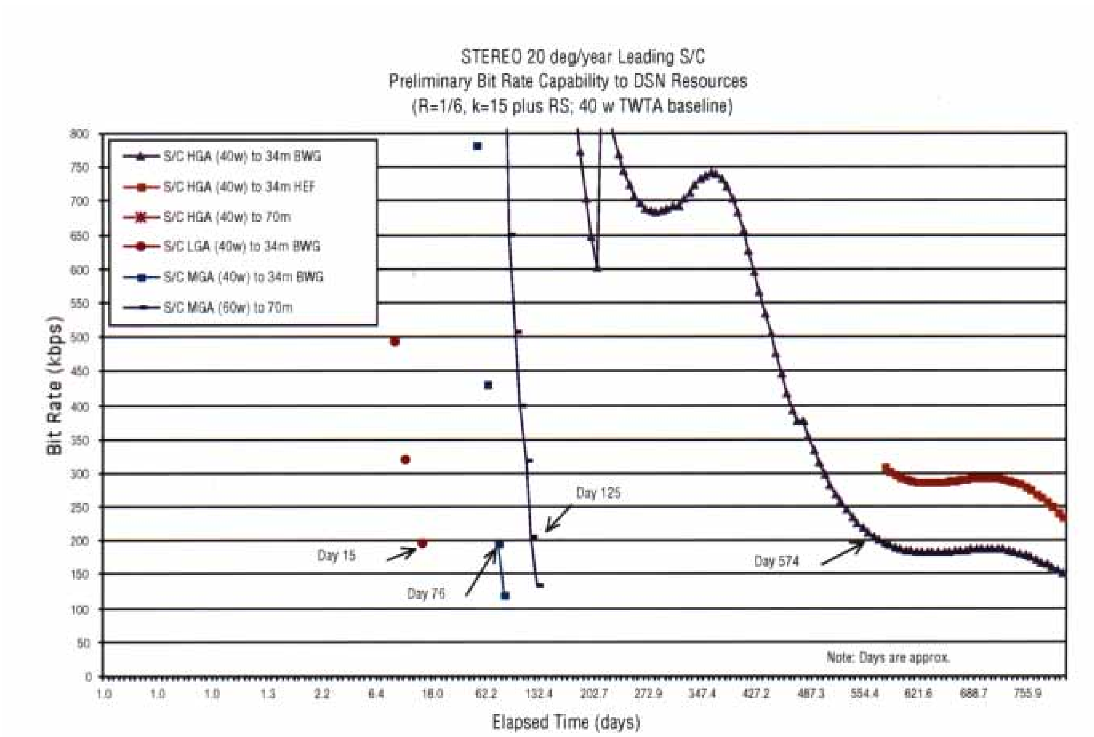


Figure 4-10 Bit Rate Performance for Leading Spacecraft

SPE= 140.3˚. The range at the two year mission life is 0.72 AU.

Figure 4-10 shows the bit rate performance for the leading spacecraft. Table 4-7 gives a bit rate performance summary. The LGA is used until ~day 15 after launch; then the 'normal mode' MGA is used until the SPE has reduced below 115˚.

Table 4-8 shows that the link will support 200 kbps with the 34m BWG during the first ~76days and from ~day 125 to 574. A number of options are then available for the remaining times:

(1) For ~day 574-823,
- use 34m HEF (if upgraded 34m BWG is not available).
- increase transmitter power (cost, dc power).
- increase HGA size (mechanical limitations with present launch vehicle, although 1.3m may be possible once the mechanical design is finalized)
- accept a lower bit rate

(2) For ~days 76-125,
- design antenna to provide required gain (cost)
- use 70 m DSN system and higher power (cost, dc power).
- slew spacecraft to direct HGA at Earth (loose science during downlink)
- redesign trajectory to move second peak of high SPE earlier in mission (may not be possible, science impact).
- have second small (~9 in) gimballed dish (mechanical constraints on spacecraft, costly)
- accept a lower bit rate

During phase A, antenna designs will be optimized for STEREO and the above options will be further investigated.

4.5.3 Navigation Support

We will use the two-way non-coherent Doppler tracking technique developed by APL. It involves making two one-way measurements instead of the usual two-way measurements. The difference in uplink frequency is measured against an onboard reference oscillator and stored in counters in the receiver card (Figure 4-6). This measurement is placed in the spacecraft telemetry and used to correct the downlink Doppler measurement. The correction is made to the tracking file received by the APL navigation team as described below. 'Differenced Doppler' can be used to validate the non-coherent doppler tracking. This is enabled by the USO and done through one-way tracking at two DSN stations simultaneously. The onboard oscillator drift is cancelled by

Table 4-8 Preliminary Bit rate Performance Summary for 20˚/yr Leading Spacecraft (Baseline is 40 w TWTA; 1.1m HGA)

DSN antenna system	S/C antenna	TWTA (w)	~Day	Earth Range (AU)	Bit Rate (kbps)
34m BWG	LGA	40	<15	$\leq$0.01	$\geq$200
34m BWG	MGA	40	15-76	0.01-0.06	>>200-200
70m	MGA	60	76-125	0.06-0.14	>>200-204
34m BWG	1.1m HGA	40	125-574	0.14-0.63	$\geq$200-200
34m HEF	1.1m HGA	40	574-823	0.63-0.77	300-200
34m BWG	1.1m HGA	54	125-792	0.14-0.72	$\geq$200
34m BWG	1.3m HGA	40	125-792	0.14-0.72	$\geq$200

differencing the data from the two stations, resulting in very accurate Doppler data.

For the two year mission, the technique can provide navigation support as follows:

- 0.1 mm/s (over 60 sec measurement interval) Doppler accuracy
- ±7500 km position

The non-coherent navigation technique requires some additional hardware in the spacecraft receiver, but does not require any changes to the DSN systems that generate the uplink signals and receive the downlink signals. The uplink signal will be generated using standard DSN capability. The operation of the DSN receiver is affected only by the fact that the exact downlink frequency is determined by the spacecraft oscillator and not by the uplink frequency and a transponder turn-around ratio. The Doppler measurement is made within the DSN receiver in the normal manner. The use of the non-coherent navigation therefore has no significant effect on the DSN station or its operations. During phase A, the technique will be discussed with DSN. This technique is familiar to DSN as breadboard testing has been done at their facility. Experiments were performed at DTF-21 side-by-side with transponder to show the performance of the non-coherent navigation technique. The COmet Nucleus TOUR (CONTOUR) program plans to use this technique.

The impact on the spacecraft is that two 16-bit counters and a small amount of digital logic are included in the spacecraft receiver to perform a comparison of the uplink signal and the downlink signal at the spacecraft. The counter values are latched at the start of each telemetry frame transmission, regardless of the frame type. These counter values are subsequently placed into the telemetry for use on the ground.

The Radiometric Data Center (RMDC) of the DSN will process the observed Doppler phase in the normal manner. Its operation is unaffected by the use of noncoherent navigation. The RMDC produces files containing Doppler frequency over specific intervals of time for use by the navigation team. Rather than being used directly by the navigation team, this Doppler data will be delivered to the STEREO project along with the telemetry frame time-stamps, the telemetered counter values and the means of associating counter pairs with the proper telemetry frame time-stamp. Software will be used to make a correction to the Doppler, so that it will be identical to that which would have been observed if the spacecraft had employed a transponder. This corrected Doppler data will then be delivered to the navigation team, in the format of the files produced by the RMDC.

The methodology of the navigation team is unaffected by use of the noncoherent navigation method. Therefore, the operation of the RMDC and the navigation team are unaffected, although an additional computational step has been inserted between them. The software needed to perform this computation will be developed and applied by the Stereo project. This software can reside at either the DSN or APL.

4.5.4 Broadcast Mode

No uplink is required. The 500 bits/sec broadcast downlink at X-band is supported through the high gain antenna to a distance of 1.98 AU using R=1/2, k=7 convolutional coding plus RS. This assumes a ground system temperature of 440 K and a G/T ~ 38.4 dB/K.

An optional 500 bits/sec downlink at S-band to NOAA resources has been analyzed. This option would require an S-band downlink card, S-band amplifier (40 w has been assumed) and design of a dual frequency X and S-band feed for the HGA. The NOAA ground system antenna gain is 45.8 dBic, system noise temperature is 100 K and R=1/6, k=15 convolutional coding plus RS are available (private communications, Mr.

Richard Grubb). Typical receiver performance has been assumed. The link through the HGA supports 500 bits/sec to 0.70 AU. This is ~789 days for the leading spacecraft and ~632 days for the lagging spacecraft.

4.6 Navigation

The Pre-Phase A portion of the navigation task involved finding potential software packages to be used operationally for orbit determination. It also involved generating STEREO's navigation requirements. Available packages were narrowed down to two candidates - GTDS and OCEAN. Requirements generation is on-going and will be reported at the end of Pre-Phase A.

GTDS (Goddard Trajectory Determination System) was developed in the 1970s and is currently being used to support ~40 missions. Source code, make-files, and documentation have been installed on APL computers, and there are no known licensing problems for using GTDS operationally. GTDS is currently being evaluated against the swingby package used for mission design.

OCEAN (Orbit/Covariance Estimation and Analysis) is a relatively new package developed by Naval Research Laboratory (NRL) circa 1995. NRL evaluation is still ongoing. It is currently backup operational support for 12 Low Earth Orbit (LEO) missions. An executable version has been installed on APL computers. Certain licensing agreements will have to be addressed for it to be used operationally. Evaluation of OCEAN will commence once the GTDS evaluation has been completed.

A unique feature of both STEREO spacecraft will be their capability to telemeter high-fidelity spacecraft (S/C) to Sun unit vectors (good to a few micro-radians) for inclusion in ground processing. This angle data can serve as surrogate ranging data, but also provides a two-dimensional input to the tracking filter, making it more useful than one-dimensional ranging data (although not yet proven to be as accurate). Both GTDS and OCEAN will be assessed for inclusion of angle tracking data, and possible code modifications will be determined.

4.7 Mechanical Subsystem

4.7.1 STEREO Spacecraft Configuration

Spacecraft Structure Description (Figure 4-11). The STEREO spacecraft structure is rectangular (56 $\times$46 inches) in shape with two hinged solar panel arrays attached to the 46 inch long sides. The spacecraft structure is composed of five basic elements; the X-frame structure, the picture frame structure, honeycomb side and endpanels, the solar array panels and the close-out panels.

X-Frame. The primary structural element serves as the backbone for transferring all spacecraft loads directly into the four point attachment to the STAR-37FM solid rocket motor adapter. The X-frame consists of three 0.780 inch thick honeycomb panels bolted together to form the "X" shape. The vertical and bottom edges of the panels contain bonded inserts to interface with the baseplate panel and the vertical side panels described in the paragraphs below. The X-frame panels are fabricated with 0.015 inch thick 6061-T6 aluminum alloy face sheets and 0. 750 inch thick 5056 aluminum alloy core material.

Picture Frame. The second structural element consists of a lightweight open framework which is closed at the bottom with a honeycomb panel baseplate. The 1.530 inch thick baseplate contains bonded inserts around the perimeter and through the center to interface with the picture frame and X-frame, respectively. The honeycomb panel baseplate also provides the spacecraft separation plane interface with the solid rocket motor adapter. The four chamfered corners of the rectangular framework work as vertical struts adding stiffness to the honeycomb panel openings. The picture frame structure is fabricated from 0.093 inch thick 6061-T651 aluminum alloy material and the honeycomb panel baseplate is fabricated with 0.015 inch thick 6061-T6 aluminum alloy face sheets and 1.500 inch thick 5056 aluminum alloy core material.

Honeycomb Panels. The third structural element is the four (4) large honeycomb vertical panels that fill the openings in the picture frame, thus providing torsional stiffness to the assembled elements described above. The side panels adjacent to the solar arrays, are 0.520 inch thick and contain bonded inserts around the perimeter and through the center of the panel to interface with the picture frame and the X-frame, respectively. The 1.280 inch thick back panel and the 0.780 inch thick front panel both contain bonded inserts as well. The honeycomb side panels are fabricated with 0.010″ thick 6061-T6 aluminum alloy face sheets and 5056 aluminum alloy core material. The honeycomb front and back panels are fabricated with 0.015 inch thick 6061-T6 aluminum alloy face sheets, 0.750 inch thick and 1.250 inch thick 5056 aluminum alloy core material, respectively.

Solar Array Panels. The fourth structural element is thehoneycomb panel substrates for the solar cells. The 1.260 inch thick solar array panels are attached to the top edge of the main structure via spring loaded hinge assemblies and stowed in a vertical position for launch. The honeycomb panels, to which the solar cells are attached, are fabricated with 0.005 inch thick 2024-T81 aluminum alloy face sheets and 1.250 inch thick 5056 aluminum alloy core material. The solar panels are preloaded against the structure for launch and are released for deployment by pyrotechnic devices. The torsion

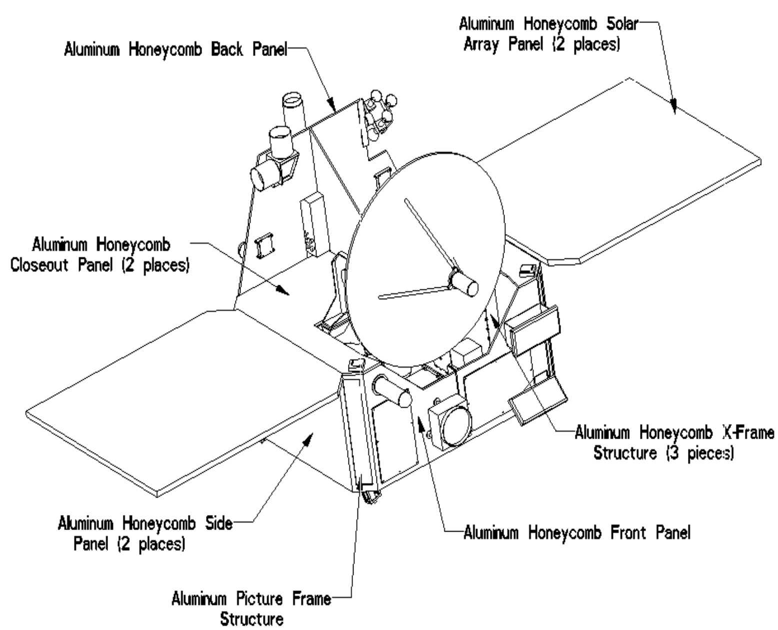


Figure 4-11 STEREO Spacecraft Structural Members

spring hinges allow the panels to rotate 90˚ to the deployed position where they are locked in place.

Close-Out Panels. The final structural element is the honeycomb panels used to enclose a large portion of the upper end of the spacecraft structure. Two 0.520 inch thick panels provide stiffness to the open end of the spacecraft structure to minimize wracking. The close-out panels The panels are fabricated with 0.010 inch thick 6061-T6 aluminum alloy face sheets and 0.500 inch thick 5056 aluminum alloy core material.

The STEREO spacecraft is attached at four points to the STAR-37FM orbit injection stage of the launch vehicle (Figure 4-12). The upper flange of the orbit injection stage, is equipped with four equally spaced separation nut assemblies forming the separation plane interface with the STEREO spacecraft. The separation nut system is then used to despin the STEREO spacecraft after orbit assemblies are positioned beneath the X-frame portion of the structure thus providing direct load paths from the spacecraft into the adapter. The orbit injection stage is equipped with a cold gas thruster system to spin up the spacecraft after separation from the Athena II fourth stage. The cold gas thruster injection and for the evasive maneuver after spacecraft release.

The STEREO spacecraft is attached to the orbit injection stage by four bolts through the baseplate panel into the separation nut assemblies (Figure 4-12). The baseplate panel is fitted with a bonded-in ring which provides the proper spacecraft mechanical interface with the STAR-37FM adapter structure. At payload separation, the STEREO spacecraft will retain the released bolt portion of the mating hardware at each location, thus allowing the spacecraft to move smoothly away from the orbit injection stage.

4.7.2 STEREO Payload Description (Figure 4-14 for Spacecraft Axes Designation)

The rectangular spacecraft bus is layed out in a manner that prioritizes the fields of view for the Solar Corona Imaging Package (SCIP), the Heliospheric Imager (HI) and the high gain communications antenna. The SCIP instrument is mounted centrally on the outside face of the +Z panel at a location that makes it the highest point on the payload. This location allows the imager to have approximately a 180˚ clear field of view and is pointed directly at the Sun at all times (Figure 4-16). It may be necessary to mount the SCIP imager on an instrument deck, which is isolated both thermally and structurally from the rest of the spacecraft structure. This would be done only if the inherent spacecraft jitter and thermal distortion would jeopardize the instrument pointing accuracy. A passive radiator is provided at the aft end of the imager to cool the Charged Coupled Device (CCD) detector to –70˚C during operation.

The HI instrument is mounted on the outside face of the –Z panel and pointed 90˚to the Sun-Earth line. The imager is located on the panel to provide the required 165˚ clear field of view and a passive radiator to cool the CCD detector to –70˚C during operation.

The parabolic dish high gain antenna is mounted along the spacecraft Z-axis and is positioned approximately in the center of the payload. The antenna is driven through its 115˚ of rotation (along the Z-axis) by a Tecstar rotary actuator capable of a 0.0094˚ step size and 0.009˚position resolution. The location of the antenna was selected to minimize the effect of the moving dish on the spacecraft center of mass and on the spacecraft center of pressure. The top edge of the –Z panel has been notched to allow the antenna dish to operate effectively at the extreme limit of its 115˚ travel. The Solar Wind Plasma Analyzer (SWPA) is mounted near the top inside corner of the +Z panel with the Faraday cups positioned to collect samples from all directions along the ecliptic plane. The Energetic Particle Detector (EPD) is also mounted near the top inside corner of the +Z panel opposite the SWPA. The EPD is equipped with a rotary actuator so that the detector can be repositioned to operate properly when the spacecraft pass behind the Earth during their two-year mission. This actuator also allows both payloads to be identical regardless of whether it is a leading or lagging spacecraft. The EPD is positioned at 45˚ to the right of the Sun-Spacecraft line to collect samples carried by the magnetic flux lines.

The Radio Burst Tracker (RBT) is composed of three plasma antennas that are deployed orthogonal to one another. Two RBT antenna are mounted on the outside face of the +Z panel at an angle of 45˚ to the Y-axis (Figure 4-16). The Orbital Sciences Corporation hingelock deployers are used to extend the antennas to a length of 10 meters from the spacecraft structure. The third hingelock deployer is mounted on the inside surface of the –X panel and deploys the antenna to a length of 10 meters through a penetration in the –X panel. The third antenna is deployed at a 45˚ angle to the Z-axis.

The Magnetometer (MAG) is attached to an Astro bi-stem actuator which is mounted to the inside surface of the –X panel. The magnetometer deployed to a length of 3-6 meters from the spacecraft structure through a penetration in the –X panel (Figure 4-17). The magnetometer boom is deployed at a 90˚ to the –X panel. The exact length of the magnetometer boom will be determined based on the magnetic signature of the payload. The –X panel also contains a penetration for the star tracker camera to look along the X-axis in the anti-Sun direction. The outside surface of the –X panel contains a cold gas thruster near each corner. Each two pound thruster is positioned at a 15˚ x 15˚ compound angle to provide 3-axis attitude control and momentum dumping capability.

The outside surface of the –Z panel contains two mid-gain (fanbeam-type) antennas and a low gain (patch-type) antenna to supplement the high gain dish antenna. A second low gain (patch-type) antenna is mounted on outside surface of the +Z panel. The –Z panel also contains penetrations for mounting the spacecraft battery and the X-band transmitter assembly through the panel from the outside. By rack mounting the battery and the transmitter in this fashion, cooling air can be directed over their mounting plates during spacecraft testing on the ground. These same mounting plates function as passive radiators for the battery and transmitter during orbital operation.

The interior walls of the spacecraft are used to mount the control electronics for the scientific instruments, electronics for attitude control devices and sensors, electronics for power switching/distribution, momentum wheels, inertial measurement unit, cold gas storage tank/ distribution components and electronics for command and data handling.

4.7.3 STEREO Spacecraft Launch Configuration (Figure 4-13, 4-18)

The STEREO Spacecraft fits snuggly into the dynamic envelope of the 92 inch fairing on the Athena II launch vehicle. The spacecraft, with the orbit injection stage attached, pushes the payload deep into the conical section of the fairing. This situation has limited the size of the high gain dish antenna in order to satisfy the science instruments field of view requirements. The spacecraft components have been positioned such that the minimum clearance between any component and the fairing dynamic envelope is 0.500 inch. The high gain dish antenna is shown in the 90˚ position to minimize the center of mass offset during launch and orbit injection. This equates to requiring less spin balance weights during the orbit injection maneuver aboard the STAR-37FM boost motor.

4.8 Structural Analysis

The STEREO spacecraft mass properties are shown in Table 4-9. The mass property calculations do not include the Thiokol STAR-37FM motor and it's structure. The center of gravity is located with respect to the STEREO spacecraft/Thiokol assembly separation plane. The spacecraft is assumed to be statically balanced such that Cgx and Cgy are very close to zero.

The Athena Mission Planner's Guide recommends that the spacecraft structural stiffness produce fundamental frequencies above 12 Hz in the lateral direction, 30 Hz in the thrust direction, and avoiding 45 to 70 Hz in the thrust direction. A NASTRAN finite element model of the launch configured STEREO spacecraft calculated the following primary structural modes:

The un-deformed STEREO finite element model is shown in Figures 4-19 and 4-20. The deformed mode shapes are shown in Figures 4-21 through 4-31.

Table 4-9 STEREO Mass Properties

Parameter	Units	Launch Configuration	Orbit Configuration
Mass	kg	350	350
Cgx	cm	39	43
Cgy	cm	0	0
Cgz	cm	0	0
Ixx	kg*m2	118	153
Iyy	kg*m2	108	108
Izz	kg*m2	115	143

Table 4-10 STEREO Mode Descriptions

STEREO Primary Structural Modes, Launch Configuration	
Frequency (Hz)	**Description**
16.4	Flexure of solar panels
26.0	Flexure of +Z deck
27.5	Flexure of High Gain Antenna support
29.8	Flexure of High Gain Antenna support
36.2	Flexure of High Gain Antenna support
41.4	Minor spacecraft rotation about Z axis
59.3	Spacecraft racking mode (mostly –X deck flexure)
63.0	Major spacecraft rotation about Z axis
63.9	Mostly spacecraft rotation about Y axis
67.6	Spacecraft rotation about Y axis
102.0	Spacecraft thrust (X)

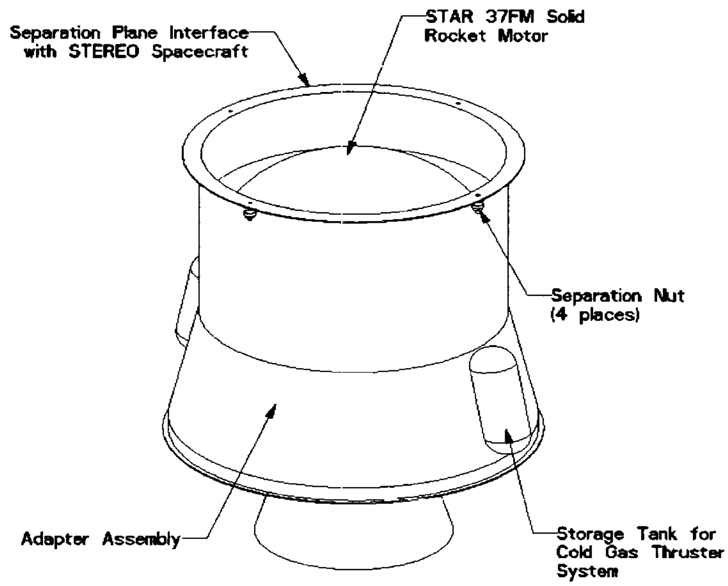


Figure 4-12 Orbit Injection Stage

The STEREO spacecraft is attached to the orbit injection stage by four bolts.

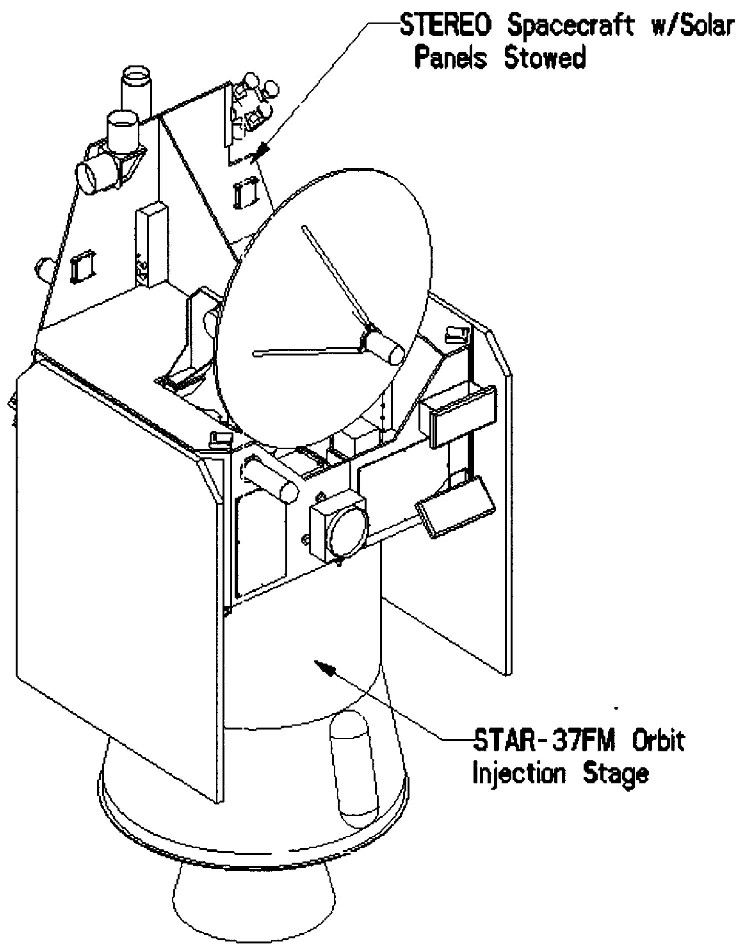


Figure 4-13 Launch Configuration

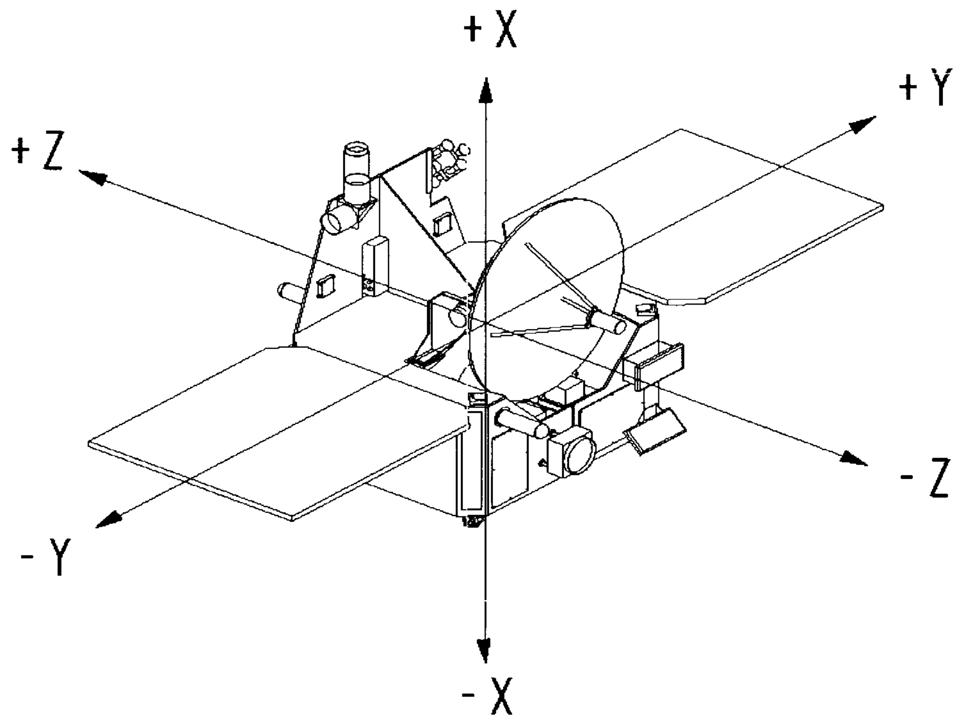


Figure 4-14 Spacecraft Axes

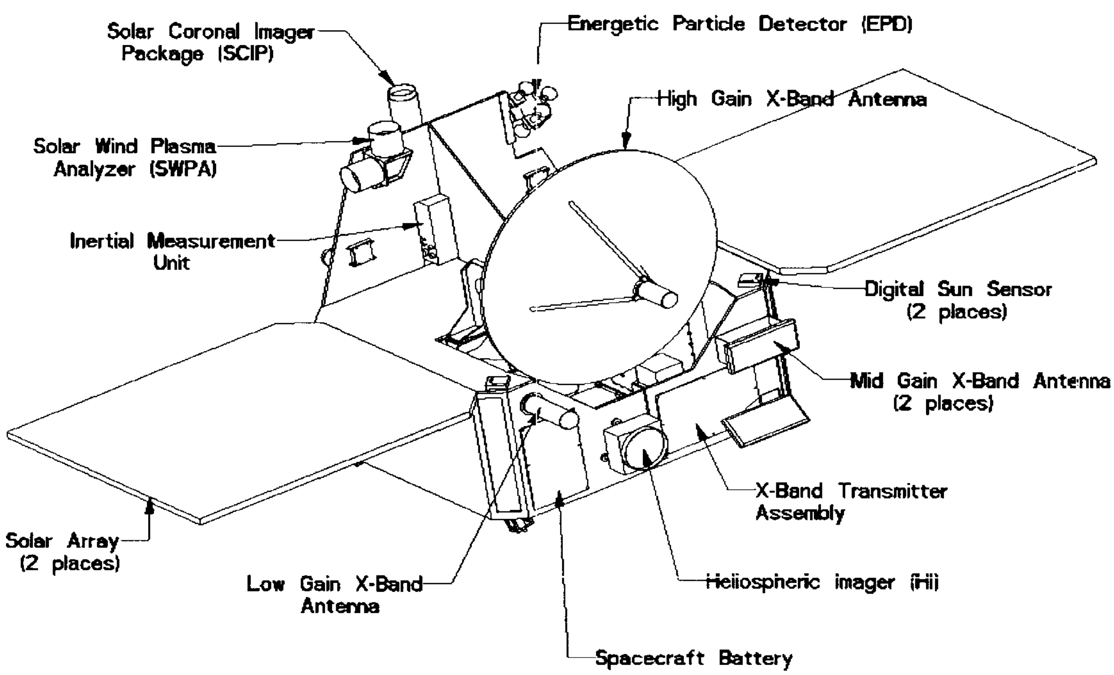


Figure 4-15

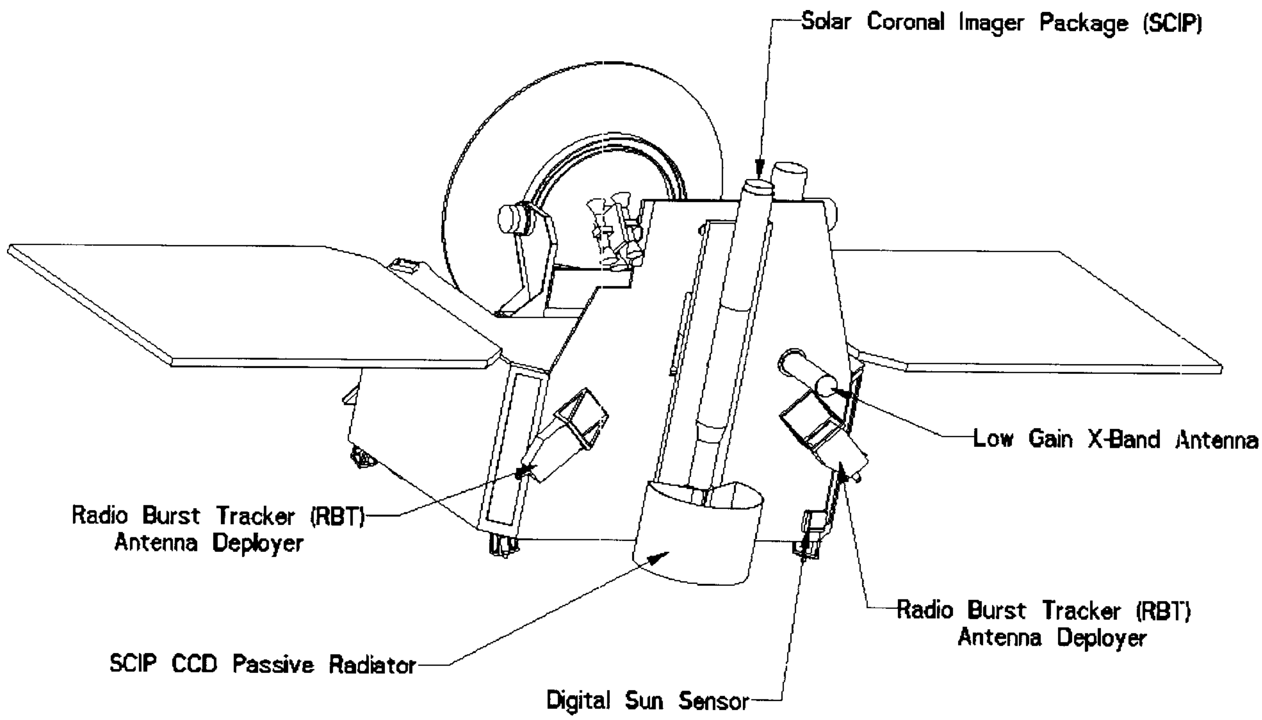


Figure 4-16 STEREO +Z Instruments

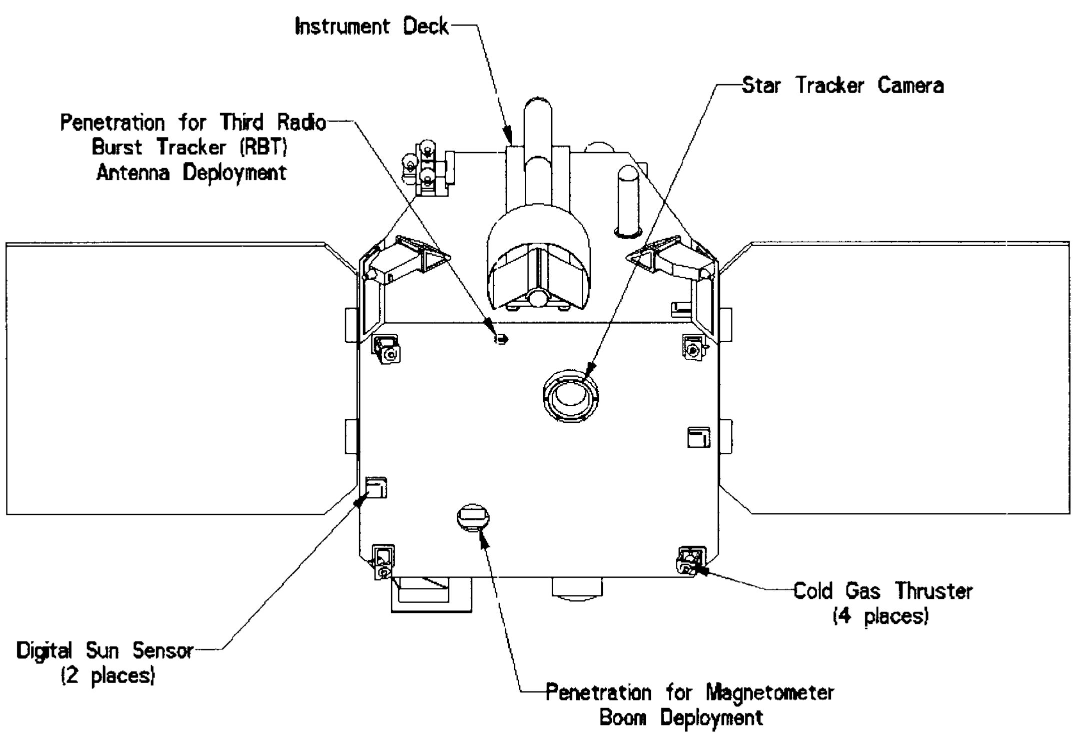


Figure 4-17 STEREO Spacecraft—X Axis

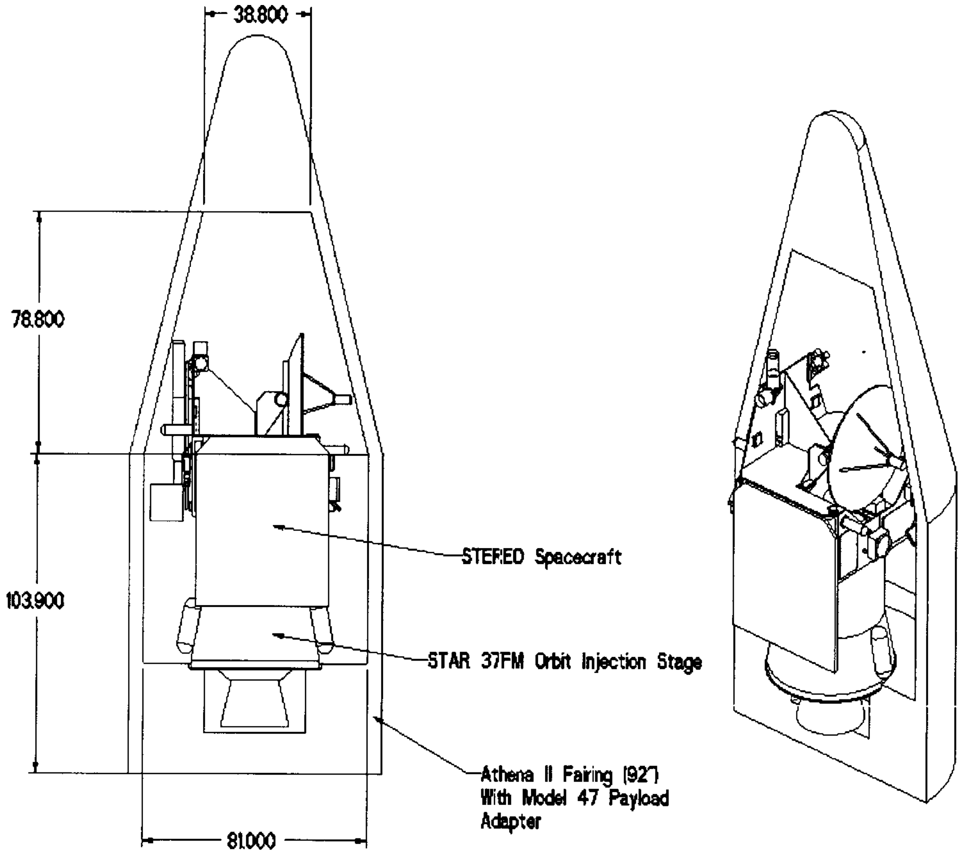


Figure 4-18 STEREO in Launch Shroud

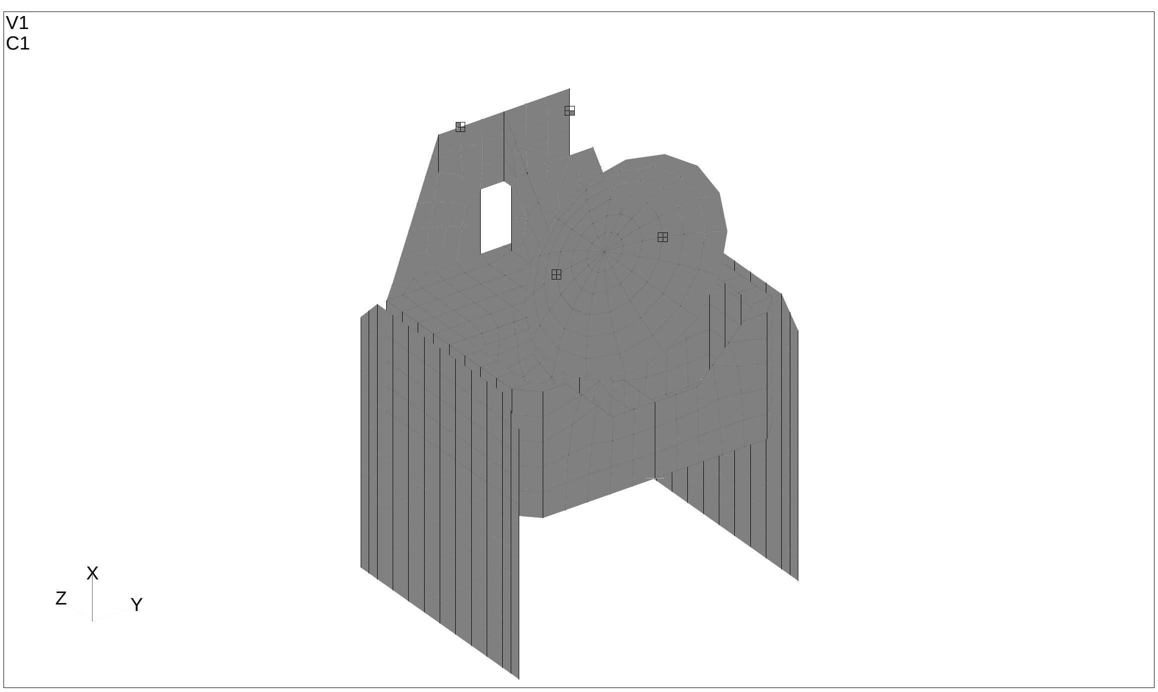


Figure 4-19 STEREO Finite Element Model (hidden lines removed)

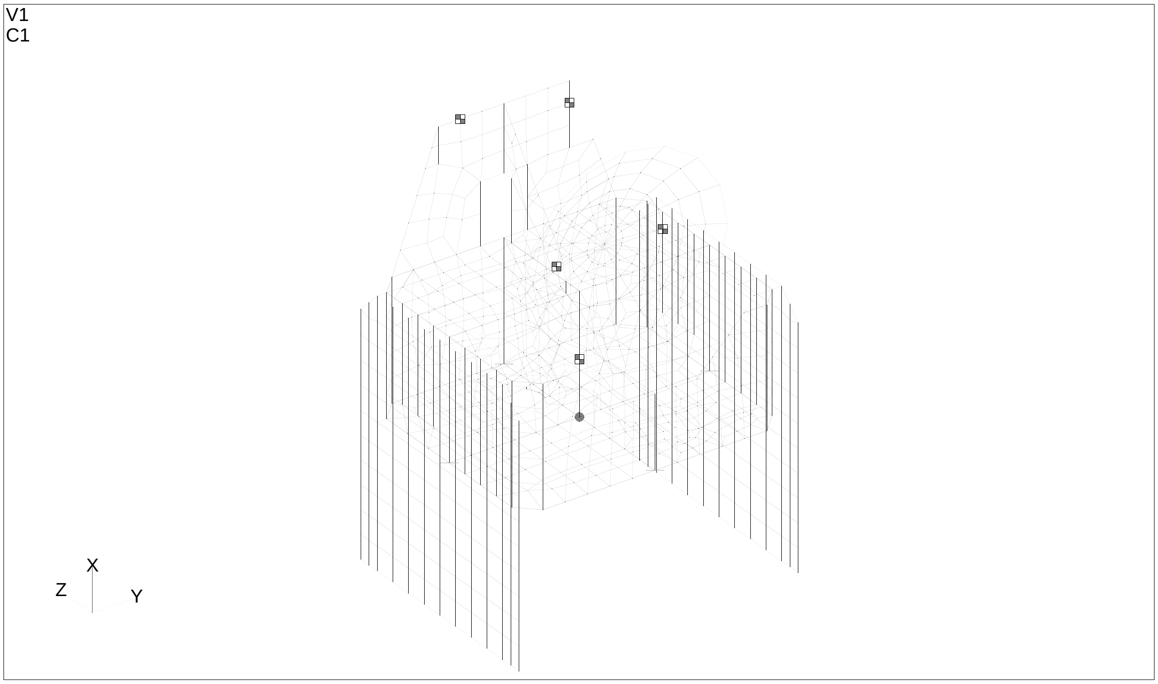


Figure 4-20 STEREO Finite Element Model

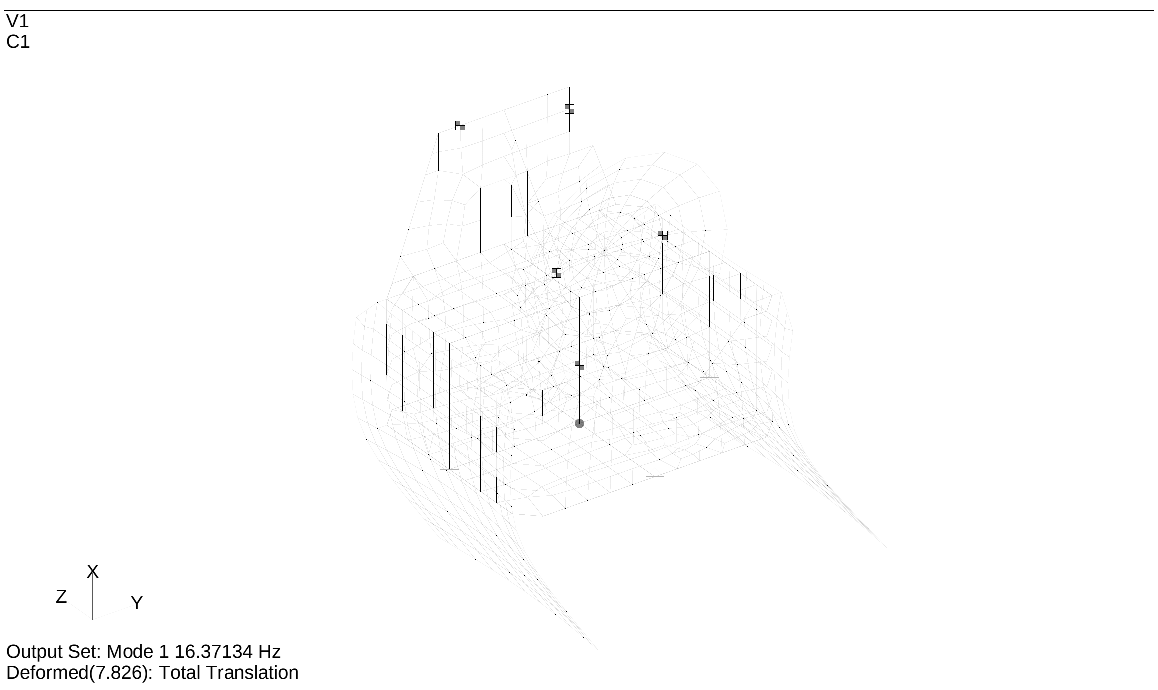


Figure 4-21 16.4 Hz, Solar Panel Flexure

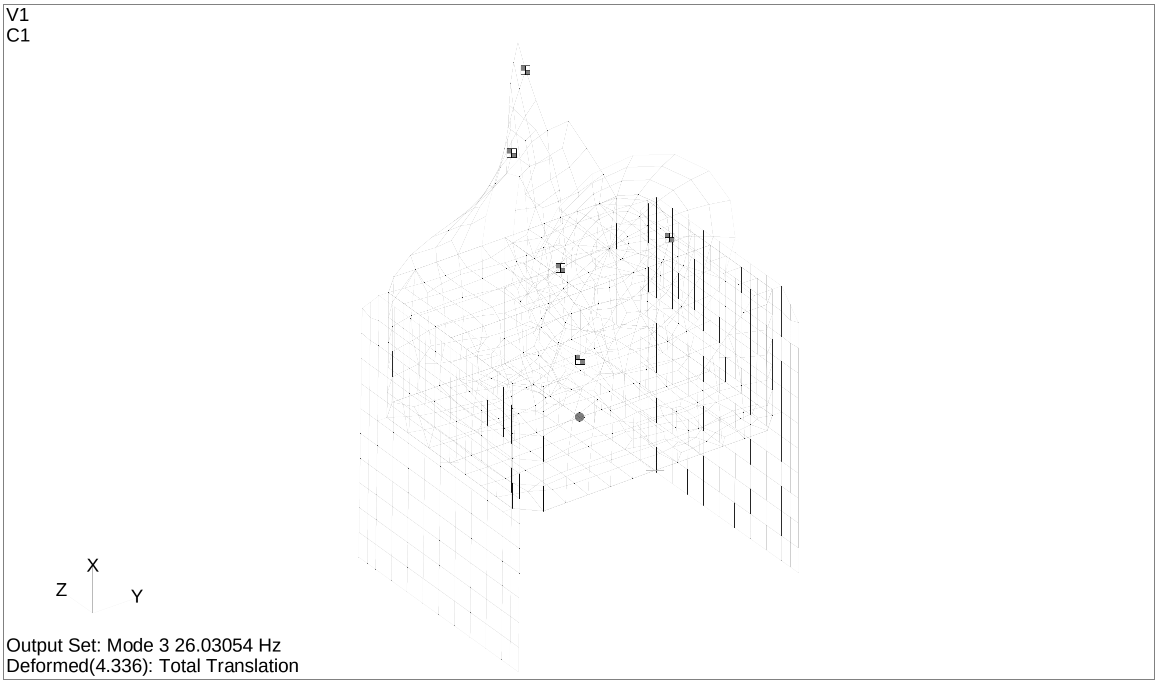


Figure 4-22 26.0 Hz, +Z Deck Flexure

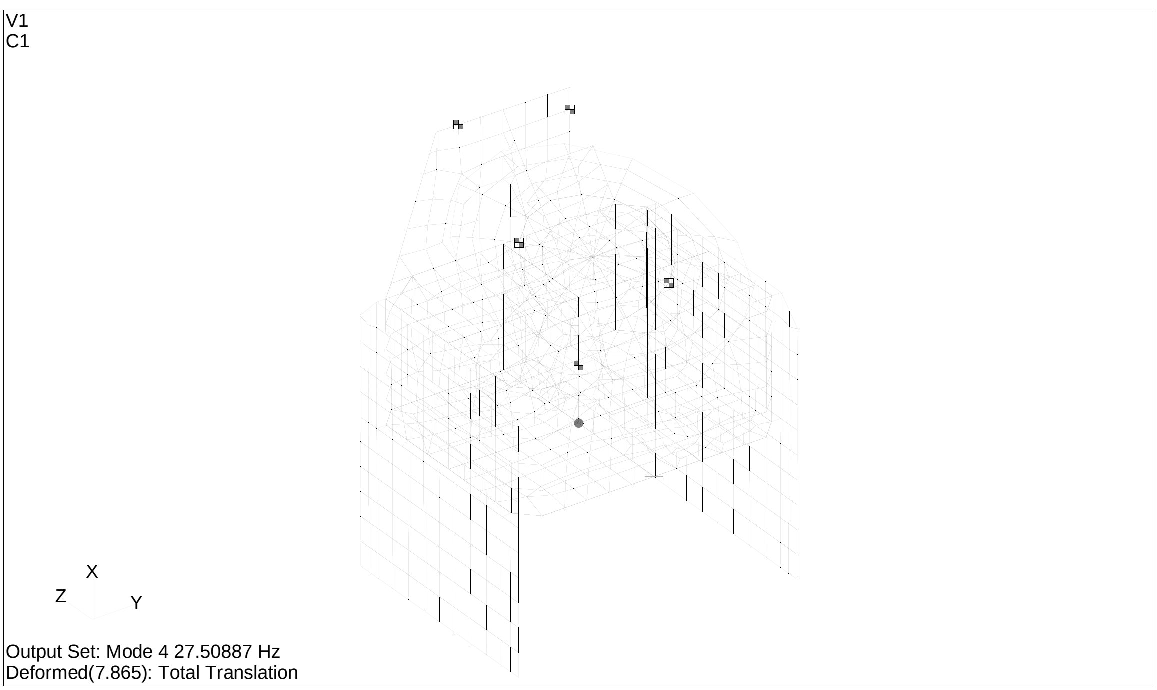


Figure 4-23 27.5 Hz, High Gain Antenna Support Flexure

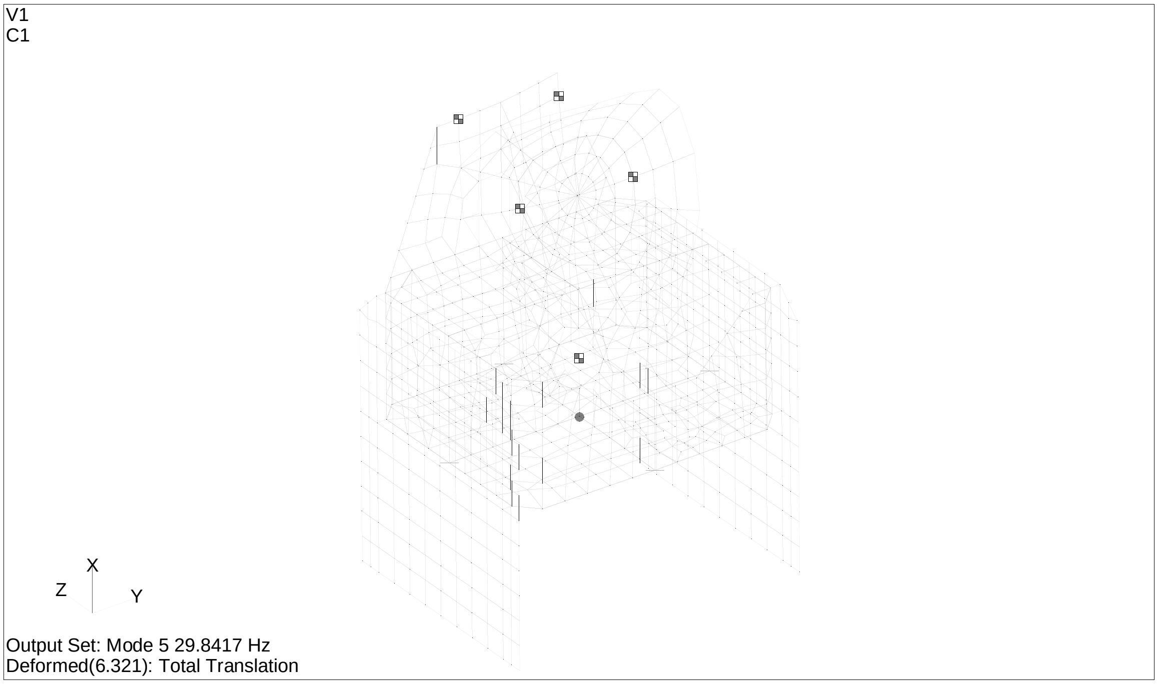


Figure 4-24 29.8 Hz, +Z Deck and High Gain Antenna Rotation about Y Axis

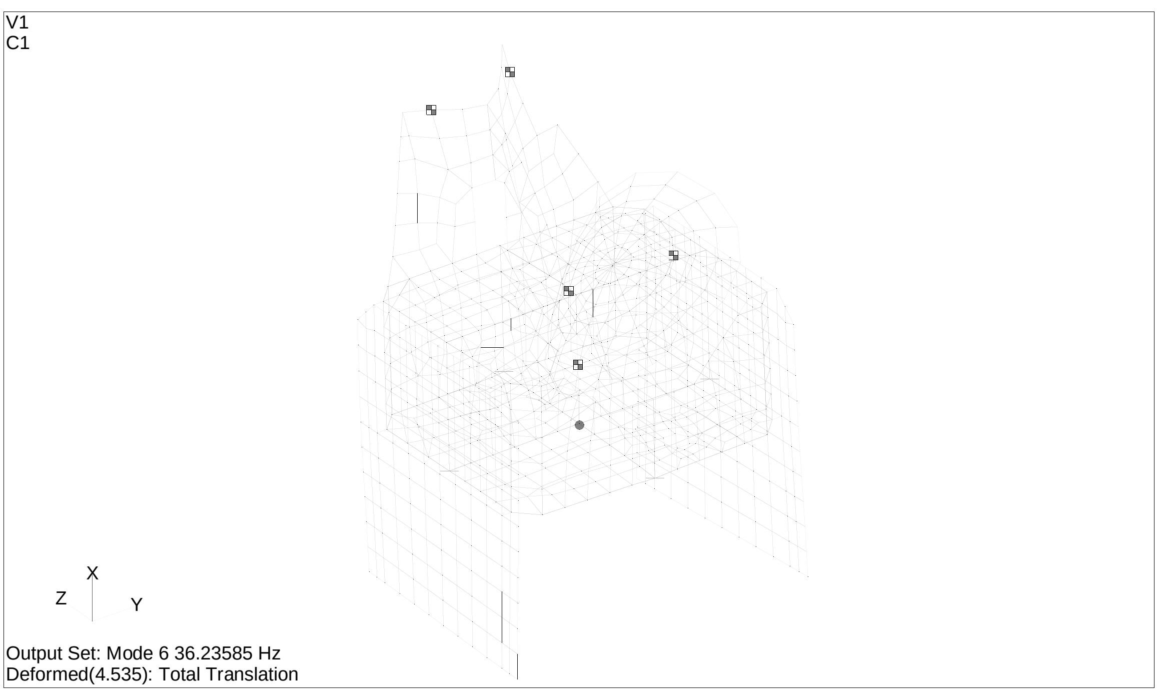


Figure 4-25 36.2 Hz, +Z Deck and High Gain Antenna Rotation about Y Axis

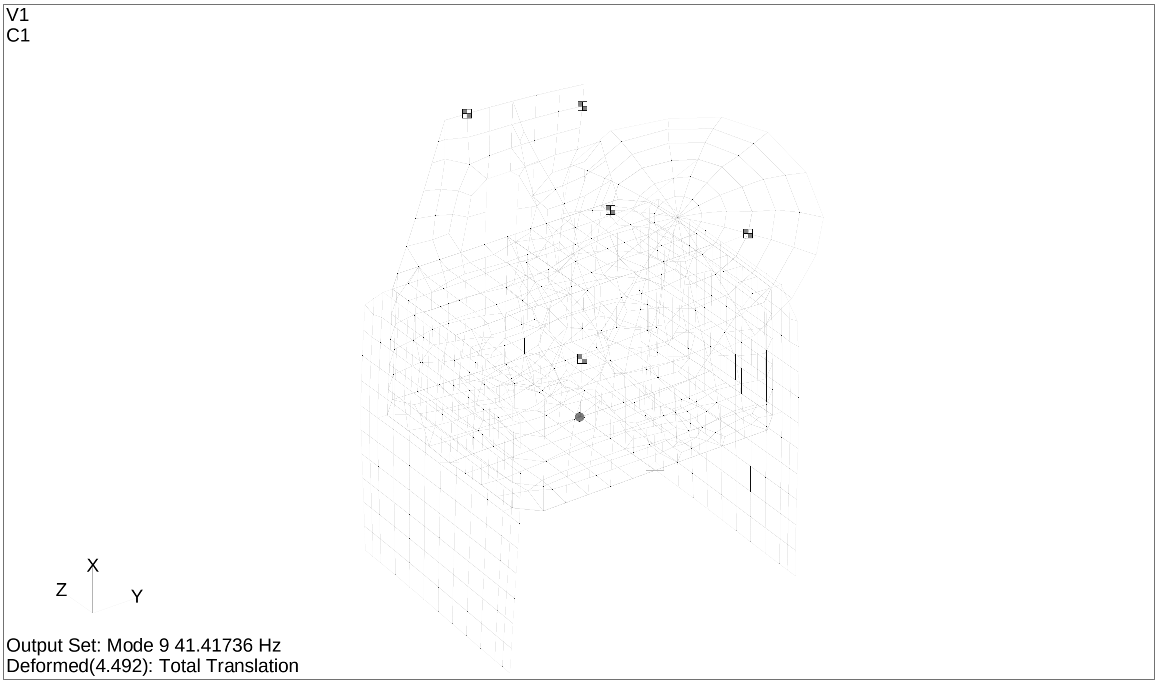


Figure 4-26 41.4 Hz,Spacecraft Minor Rotation about Z Axis

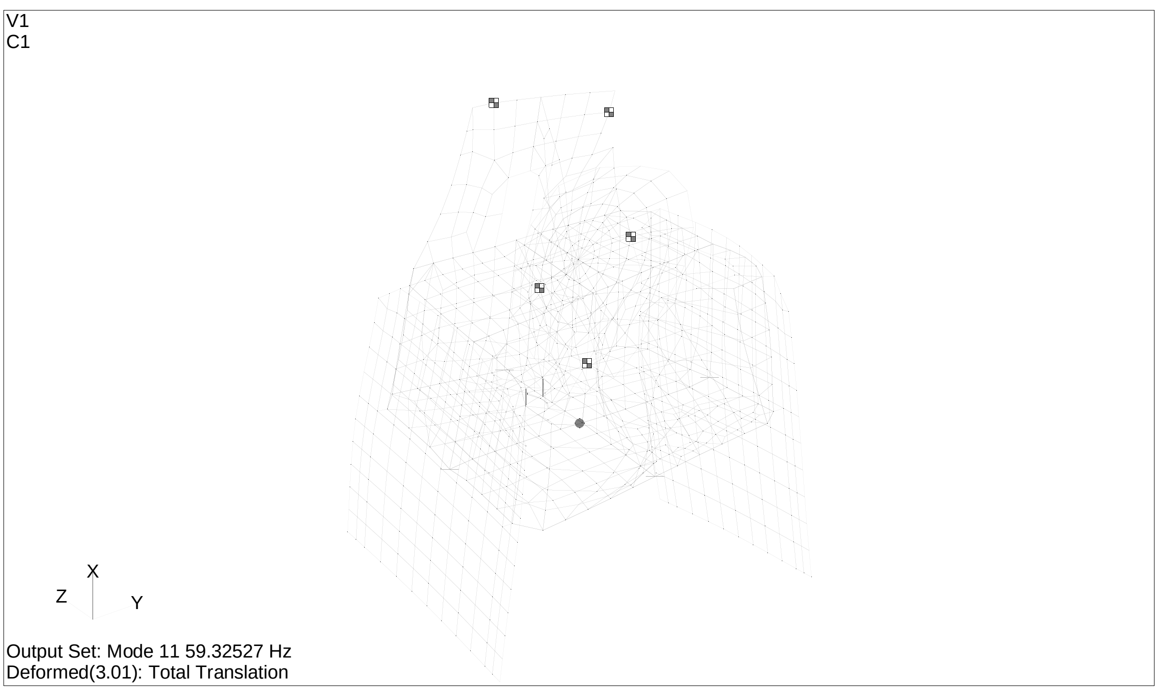


Figure 4-27 59.3 Hz, Spacecraft Racking (mostly –X deck flexure)

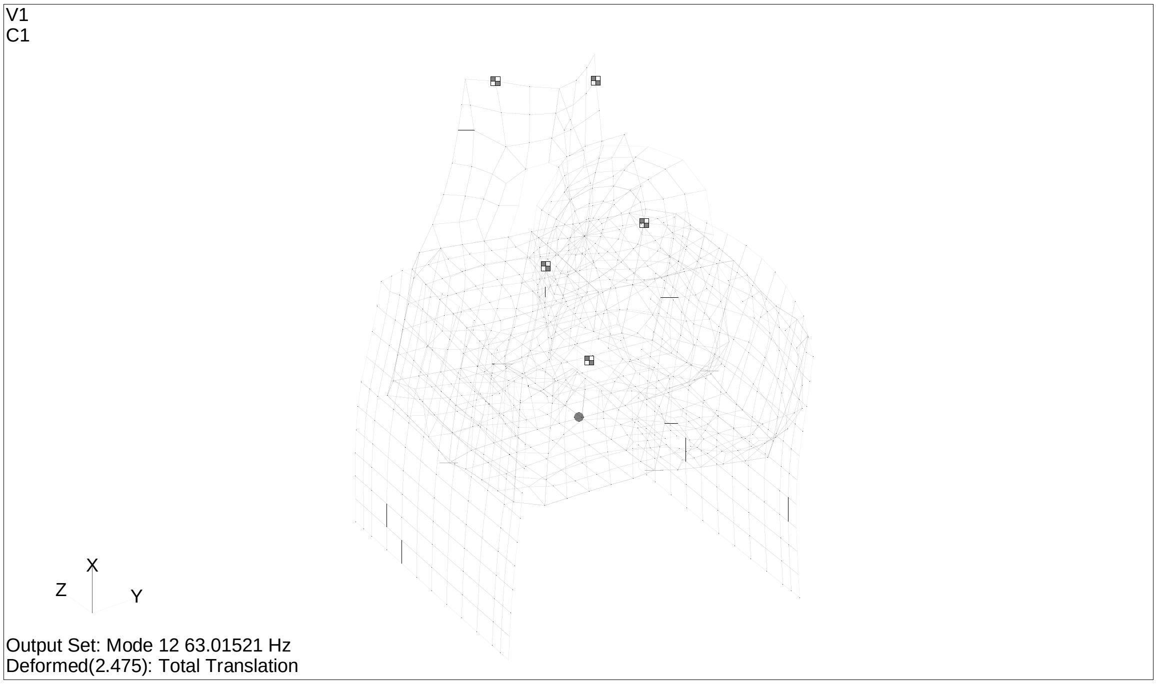


Figure 4-28 63.0 Hz, Major Spacecraft Rotation about Z Axis

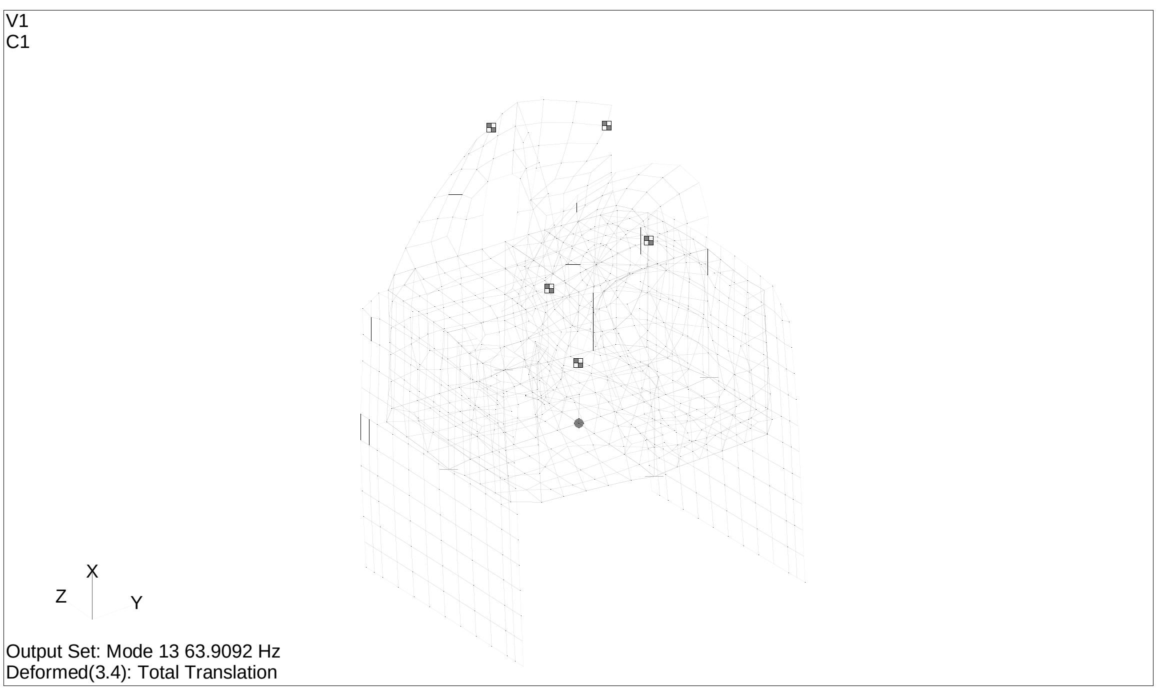


Figure 4-29 63.9 Hz, Spacecraft Rotation about Y Axis

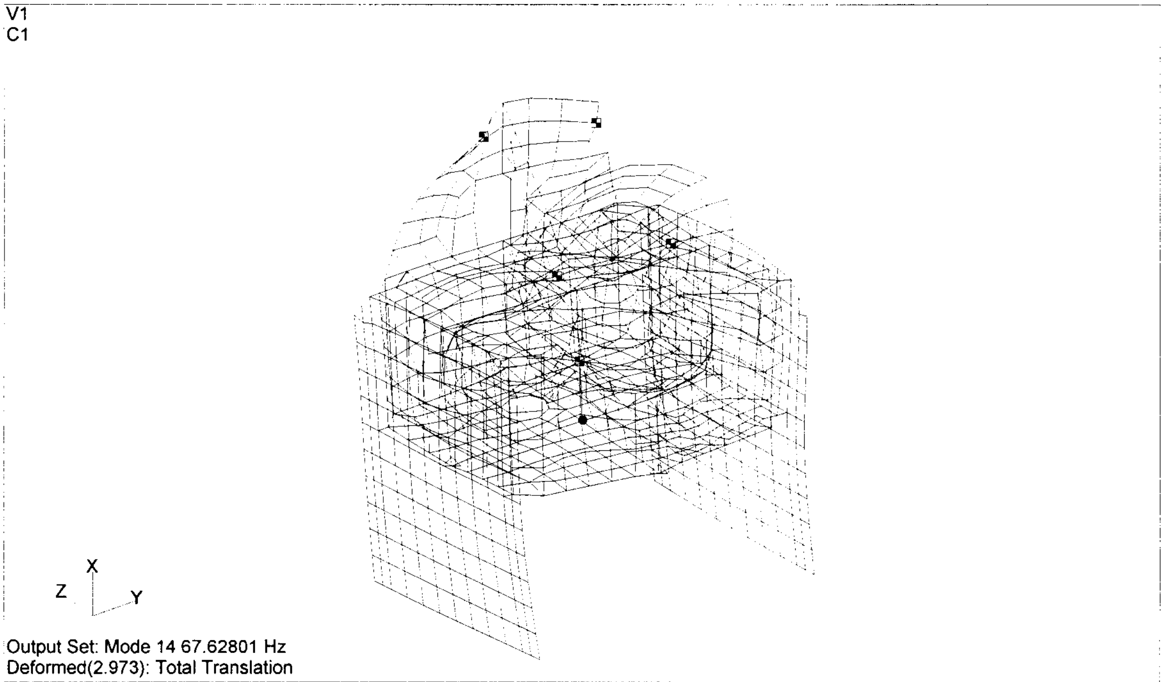


Figure 4-30 67.6 Hz, Spacecraft Rotation about Y Axis

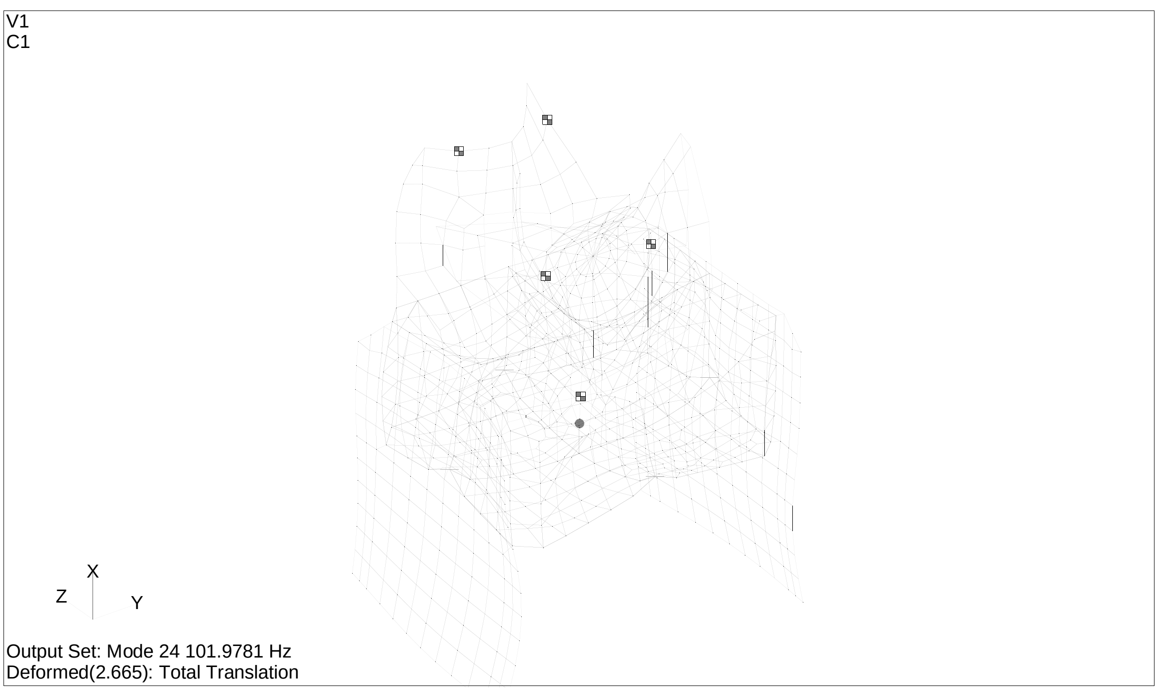


Figure 4-31 102.0 Hz, Spacecraft Thrust (X)

4.9 Thermal Design

The STEREO spacecraft thermal design will be simple and robust using no louvers or heat pipes and very little heater power. The design will accommodate solar distance variations between 0.85 and 1.18 AU, a solar pointing attitude of $\pm 5^{o}$ off Sun line of sight, constant electrical loads during operation, and a two year mission life. The thermal design for each of the two proposed spacecraft will be identical, using an ESD (electrostatic discharge) mitigating coating on the external surface of the Multi-Layer Insulation (MLI), allowing for the complete grounding of the MLI. All thermal hardware will meet program cleanliness requirements.

Spacecraft radiators will be body mounted and located away from environmental heat sources. The radiators will be designed to maintain the internal spacecraft temperature between –10 and $+35^{o}$C during operation, and –25 to $+45^{o}$C during survival conditions. Spacecraft operational heater power will be used sparingly because of constant electrical loads and a wide bus operating temperature range. Survival heater power will be used when electrical loads are reduced and the internal spacecraft temperature falls below the minimum design threshold.

The instruments, in general, will be thermally isolated from the spacecraft. This approach will simplify the spacecraft design and sub-system level testing, allow for wide interface temperatures, and potentially reduce overall heater power requirements. Instruments whose desired interface temperature ranges match of the spacecraft may be candidates for non-isolation. Currently instrument thermal requirements are TBD.

The baseline launch vehicle for STEREO is an Athena II with the potential for a change to the Space Shuttle. There are no foreseeable thermal requirements for Athena that would drive the spacecraft's overall thermal design. However, preliminary thermal analysis shows the potential for severe hot and cold Shuttle Bay environments depending on Shuttle attitude. A Space Shuttle launch will require a much more rigorous thermal analysis to be iterated between STEREO and Space Shuttle thermal personnel in order to determine the most benign acceptable Shuttle attitude for the STEREO mission.

All spacecraft components will be thermally tested per the STEREO Component Environmental Specification. The purpose of the testing is to determine workmanship flaws in flight hardware. As an example, a typical electronics box will be cycled six times between hot and cold operational plateaus with one survival cycle. Typical soaks are four hours at each plateau. The integrated spacecraft level thermal vacuum test will be conducted at Goddard Space Flight Center in chamber 290. The baseline has both spacecraft being tested at one time. Under the baseline, at least one spacecraft would be thermally balanced and both would be thermally cycled. The thermal cycles would achieve a minimum of three hot and cold cycles with a minimum of 108 operational hours accumulated at each plateau.

4.10 Propulsion Subsystem

The STEREO propulsion system is required to provide 3 axis torques to stabilize the spacecraft after separation and to provide 3 axis torques for momentum wheel desaturation periodically throughout the two year mission. The momentum wheels require desaturation because mis-match between the spacecraft center of pressure and center of mass will cause a momentum build-up, resulting in excessive wheel speed. A 1500 N-sec cold gas propulsion system with four double canted thrusters has been selected to satisfy all tip off rate nulling and momentum dumping requirements. System sizing includes margin provided by the requirement to load five years worth of expendables as well as a 10% leakage allowance.

As shown schematically in Figure 4-32 the propulsion system consists of a high pressure gas storage tank, a fill/vent valve, a main system filter, an isolation latch valve, a pressure transducer, a test port and four dual seat solenoid thruster valves. A study of control requirements showed that the system can function unregulated over the sizing pressure range of 34,500 to 690 kPa (5000 to 100 psia). Each thruster will be calibrated to provide 4.448 N (1.00 lb) of thrust at 31000 kPa (4500 psia) within ±3%. Nominal thrust vs. inlet pressure is shown in Figure 4-33. The double canted thruster arrangement shown in Figure 4-34 will provide the required forces and torques. Table 4-11 details the calculations used to determine how much center of pressure (Cp)/center of gravity (Cg) off-set can be accommodated by the selected 8.0 L (490 in 3) pressurant tank. Future iterations will include variable solar pressure tied to the actual launch date as well as updated mass properties and effective surface areas.

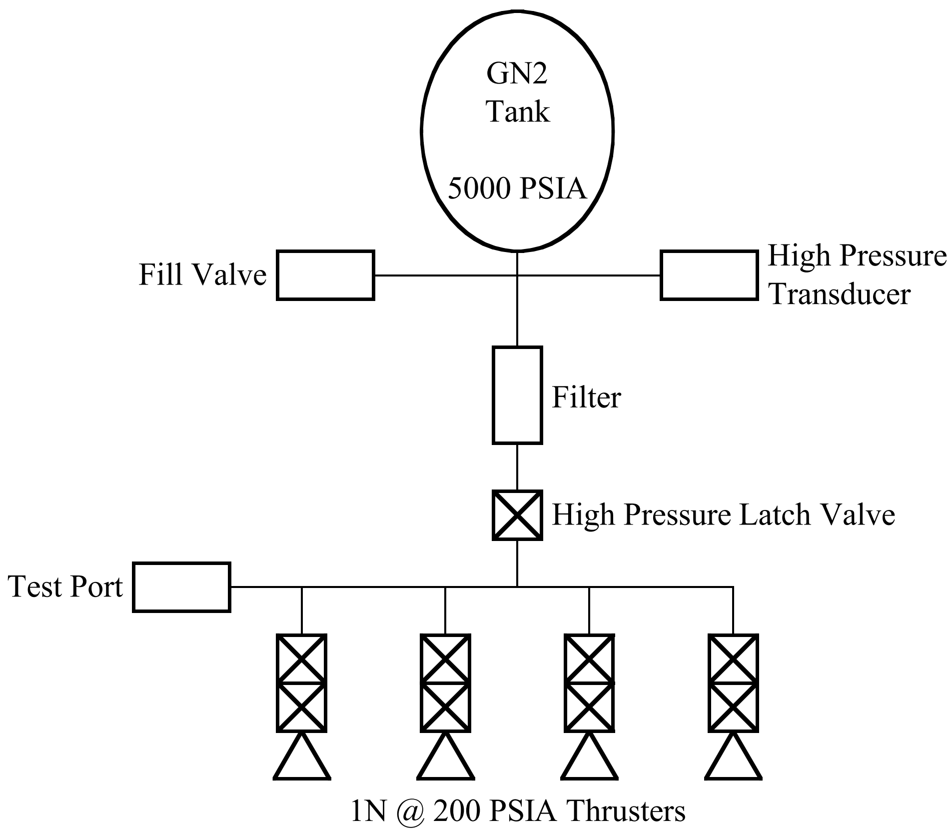


Figure 4-32 Propulsion System Schematic

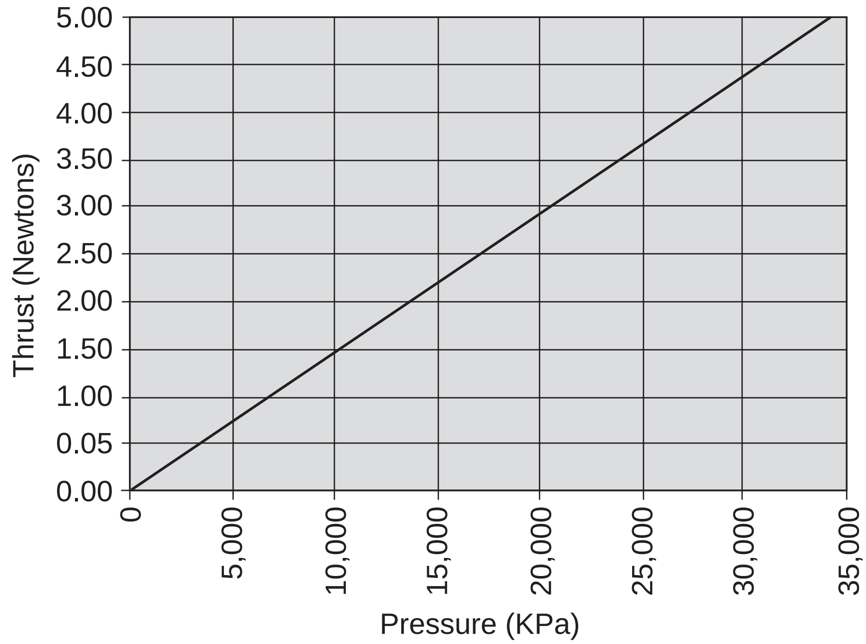
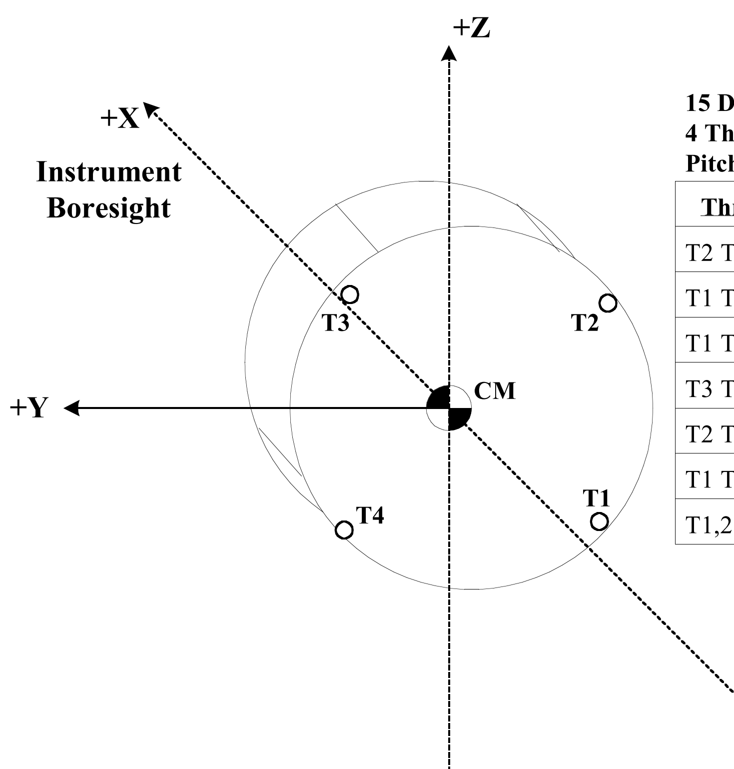


Figure 4-33 Thrust is Linear Over the (5,000–100 PSIA) Range



15 Degree Double Canted
4 Thruster set provides
Pitch, Yaw, Roll and 1 Axis Delta V

Thruster	Torque	Axis
T2 T3	+Pitch	Around Y
T1 T4	-Pitch	Around Y
T1 T2	+Yaw	Around Z
T3 T4	-Yaw	Around Z
T2 T4	+Roll	Around X
T1 T3	-Roll	Around X
T1,2,3,4		Along X

Figure 4-34 Thruster Configuration and Torque

Table 4-11 Sizing Spreadsheet Shows Accommodation of 16.0 cm CG, CP Offset

Parameter	Metric	English
Spacecraft Initial Mass—kg, lb	350.00	771.61
Injection Trim Delta V—m/sec, ft/sec	0.00	0.00
Separation Spin Rate, RPM	0.00	0.00
X Approximate Stowed Spin Inertia—kg-M^2	180.00	
Despin—N-m-sec, lb-ft-sec	0.00	0.00
Max Tip-off Rate—°/sec	2.00	2.00
Y, Z Approximate Stowed Inertia kg-M^2	160.00	
Tip-off Rate Nullification—N-m-sec - lb-ft-sec	5.59	4.12
S/C Projected Area—M2, ft^2	6.00	64.59
Solar Radiation Pressure—N/M2 - lb/ft^2	4.617E-06	9.646E-08
Reflectance Factor	0.60	0.60
Off Normal Sun Angle—Deg	0.00	0.00
CP/CG Offset—cm, ft	16.00	0.525
Mission Duration—years	5.00	5.00
Mission Duration—sec	1.58E+08	1.58E+08
Thruster Moment Arm—M—ft—in	0.75	2.46
GN2 Isp—sec	65.00	65.00
Thrust—N, Lb	4.448	1.000
Flow Rate—g/sec, lb/sec	6.978	0.01538
System Leak Rate—sccs	4.00E-05	
Mission Leak Total—scc	6311.520	
Leakage Allowance—%	10	10
Nominal GN2 Temperature—Deg C, F	21.1	70.0
Maximum GN2 Temperature—Deg C, F	40.0	104.0
Minimum GN2 Temperature—Deg C, F	-28.9	-20.0
Initial Tank Pressure @ Nom. Temp.—kPa, psia	32404	4698.6
Final Tank Pressure @ Min. Temp.—kPa, psia	690	100.0
MEOP @ Max. Temp.—kPa, psia	34483	5000.0
Burst/MEOP Factor of Safety	2.0	2.0
GN2 TANK CALCULATIONS		
Despin—N-sec, Lb-sec	0.0	0.0
Tip-off Nullification—N-sec, lb-sec	7.4	1.7
Solar Radiation Pressure Torque—N-M, lb-ft	7.09E-06	5.23E-06
Momentum Dump Impulse—N-sec, lb-sec	1492.38	335.66
Injection Trim GN2—kg, lb	0.000	0.000
Despin GN2—kg, lb	0.000	0.000
Tip-off Nullification GN2—kg, lb	0.012	0.026
Momentum Dump GN2—kg, lb	2.341	5.164
Leakage Allowance GN2—kg, lb	0.234	0.516
Required Mission GN2 Total—kg, lb	2.587	5.706
Required Tank Volume—L, in3	8.03	490.00
Spherical Tank ID—cm, in	24.84	9.78
Tank OD—cm, in	26.11	10.28
Tank OAL—cm, in	27.38	10.78
Tank Mass—kg, lb	2.40	5.30
Total Loaded GN2 Mass—kg, lb	2.652	5.846

5.0 INTEGRATION AND TEST (I&T)

In order to meet the current Solar TErrestrial RElations Observatory (STEREO) schedule it will be necessary to integrate and test two spacecraft in 10 months. This is an aggressive schedule, almost demanding that the two spacecraft be integrated concurrently. Concurrent integration, however, is not feasible due to the significant cost implications. This implies that the Integration and Test period must be well planned with contingencies taken into account for both spacecraft. The section provides an overview of the I&T portion of the program, detailed planning will occur during the rest of Pre-Phase A and during Phase A/B.

5.1 Overview

The two STEREO spacecraft will be integrated at The Johns Hopkins University Applied Physics Laboratory (JHU/APL) with environmental testing occurring at both APL and the Goddard Space Flight Center (GSFC) in a FED-STD-209E class clean area. The STEREO testing philosophy will be based on APL document SDO 2387-1, MIL-STD-1540B, and GEVS-SE.

In order to meet the tight I&T schedule, the two spacecraft will be integrated and functionally tested as if they were a single redundant spacecraft with a side A and a side B. This means that a subsystem is integrated onto the first spacecraft, functionally tested and then the same subsystem is integrated onto the second spacecraft. This occurs on intervals of about two weeks. The same integration method would apply to instruments as well.

The methodology has several benefits:

- Allows the I&T team to remain in the same testing configuration for both spacecraft.
- Allows the I&T team to apply lessons learned from one spacecraft to the second.
- Permits the use of a single I&T team.

Once the spacecraft are integrated, they will be tested for performance and function concurrently, using a single I&T team and Ground Support System (GSS). Dual Ground Support Equipment (GSE) for power, RF and instruments will be required. Once functional and performance testing is complete, the spacecraft will be environmentally tested in a linear fashion except for thermal-vacuum testing which will occur concurrently. After thermal-vacuum testing at GSFC, the spacecraft are shipped for launch. A Pre-Ship Review is scheduled prior to packing of the spacecraft.

GSE Requirements. It is planned that, each spacecraft will have its own subsystem GSE that will follow the spacecraft throughout the I&T process. This equipment consists of a Block-House Charging Unit (BCU) for the powering spacecraft during ground tests, RF interface equipment racks for actuating the RF equipment, Solar Array Simulators (SAS) for powering the spacecraft, and the GSE for operating all the spacecraft instruments.

Both spacecraft will be operated from one GSS, which will transition into Mission Operations once I&T at APL is finished. The GSS is used to run scripts that functionally test the spacecraft and display spacecraft telemetry data. It can also operate via a network connection.

I&T Personnel. The Integration and Test team will be made up of dedicated I&T personnel, subsystem support, personnel from the Mission Operations Team (MOT) and system engineering. I&T personnel will be used to plan the I&T process, physically integrate and test the subsystem flight hardware as it is integrated on to the spacecraft, validate system level functional and performance requirements and environmentally test the spacecraft. The I&T team is also responsible for all remote activities including testing at GSFC and launch site operations.

The I&T team is supported by the subsystem leads during subsystem integration and system functional/performance testing. The MOT is used during I&T for two reasons. First, the MOT bolsters the I&T team during a relatively intensive portion of the spacecraft development schedule. Secondly, it provides the MOT with experience operating the spacecraft. One of the goals is to operate the spacecraft during I&T as if it were on-orbit. The System Engineer's role during I&T is to make sure that all requirements are validated and that autonomy is fully tested and operates as designed.

Cleanliness. The STEREO spacecraft have two cleanliness requirements. Class 100,000 prior to integration of several of the instruments and class 10,000 during and after these instruments are integrated. This will necessitate that the class 100,000 environment be transitioned to a class 10,000 one for the instruments. The goal is to make this transition as late as possible. With all other things being equal, the class 10,000 instruments will be held until last to be integrated on to the spacecraft

Prior to integration of the class 10,000 instruments, the spacecraft and clean room will be scrubbed to a class 10,000 level. This will be verified by the use of witness plates and air sampling. This verification will be performed by APL's Reliability and Quality Group. Another option of maintaining a Class 10,000 just around the instruments (bagging) will also be evaluated when instrument teams are engaged.

A detailed Contamination Control plan will be written during the next phase of APL's effort. It will address contamination in general as well as providing details from a class 100,000 to a class 10,000 environment.

5.2 Spacecraft Integration and Test Flow

The objectives of I&T portion of the program is to test the two STEREO spacecraft under ambient conditions and simulated environments in order to establish correct operations of all subsystems and instruments, when interconnected as a spacecraft. This confirms that all electrical interfaces between spacecraft and instruments are in compliance with specifications and/or interface control documents. The performance of the spacecraft will be validated and recorded as a baseline by system performance testing at several points in the testing process. Furthermore, the spacecraft is required to display quality of workmanship by demonstrating performance under mission-level environmental stress.

Integration of the STEREO spacecraft starts after the structure, propulsion and harness have been delivered as a unit. Subsystem integration will be completed in the order outlined in Figure 5-1. The integration effort will ping-pong between the two spacecraft. This allows both spacecraft to maintain schedule and lessons learned on the first spacecraft to be applied to the second. The subsystem integration order was chosen to facilitate testing. Instruments are integrated next to last, with the instruments requiring a Class 10,000 environment integrated last.

Spacecraft Integration. The two STEREO spacecraft will undergo integration and performance/functional testing in the APL Clean Room in the Kershner Space Integration and Test Facility (Building 23). Environmental testing will occur at APL and the environmental test areas of GSFC Environmental Test Engineering Branch Facility (Buildings 7 and 10).

Tests that occur in the APL clean room will include functional testing of all components, subsystems, and instruments. Performance testing will then be conducted to establish a "baseline of data" from the spacecraft to compare with testing during the environmental test program and flight operations.

Spacecraft subsystems and instruments for both STEREO spacecraft are to be delivered flight qualified and fully integrated so as to meet the schedule outlined in Figure 5-1. Figure 5-1 is labeled in months prior to launch site delivery. The I&T schedule officially runs from the point when the first subsystem is ready to be integrated on to the spacecraft structure to launch site shipping. I&T preparation and planning starts much earlier and runs from Pre-Phase A through A/B and onto C/D.

Subsystem integration occurs in the following order, on to an already integrated structure, harness, and propulsion system:

1. Power electronics and battery (Solar Array Simulator is used for test, solar arrays are installed just before vibration, "work" batter is install for initial integration, flight battery is installed before performance test period)
2. RF Subsystem (Antennas, TWTA, switches, diplexers and coax)
3. IEM (C&DH, Uplink and Downlink cards)
4. G&C Subsystem (attitude interface electronics, Guidance and Control Computer, RWAs and electronics, IMU, star tracker, propulsion system, DSADs and electronics.
5. Instrument 1
6. Instrument 2
7. Instrument 3
8. Instrument 4
9. Instrument 5 (Class 10,000 assumed)
10. Instrument 6 (Class 10,000 assumed)
11. STEREO 1 & 2 Ordnance and Thermal Systems (all pyro-ordnance devices, ie.: Booms, deployers**,** covers or lids for instruments. Thermal heaters, temperature sensors, and multi-layer insulation (MLI).

Each subsystem is allocated approximately two weeks per spacecraft for physically integrating the subsystem and functional test. Time consuming tasks such as alignments are saved until later in the schedule . Instrument integration can theoretically occur in the order that they are delivered, however, it is preferable that the class 10,000 instruments be integrated last.

The order for subsystem integration was chosen such that the spacecraft can be tested in ever increasing levels of complexity. For example, none of the spacecraft could be tested without a power subsystem, therefore power is integrated first.

The last items to be integrated and functionally tested are ordnance and thermal systems. Ordnance systems are integrated next to last due to the changes in handling procedures when live ordnance is on the spacecraft. Thermal systems are held until last because they limit access to the spacecraft.

Spacecraft Environmental Testing

Once integrated, the spacecraft will go through a series of environmental tests with a goal of ascertaining proof of performance under the environmental loads that the spacecraft will see during launch, fly-out and operations. Mass properties for each spacecraft are also confirmed during this period.

Environmental testing for each spacecraft will also occur in a ping-pong fashion, except for thermal-vacuum testing. Due to the long period of time needed for thermal-vacuum testing, both spacecraft will be tested at the same time. Thermal-vacuum testing will occur at GSFC. A detailed thermal-vacuum study will be performed during the remainder of Pre-Phase A.

Prior to thermal-vacuum testing the spacecraft will undergo, random vibration, sine/load testing, shock testing and acoustic testing (at GSFC). After each of the major environmental tests, spacecraft performance will be verified against the baseline set during integration. Mechanical alignments will also be checked. Prior to thermal-vacuum testing, mass properties and spin balancing occurs for each of the vehicles. Spin balancing is only required for the Athena II launch.

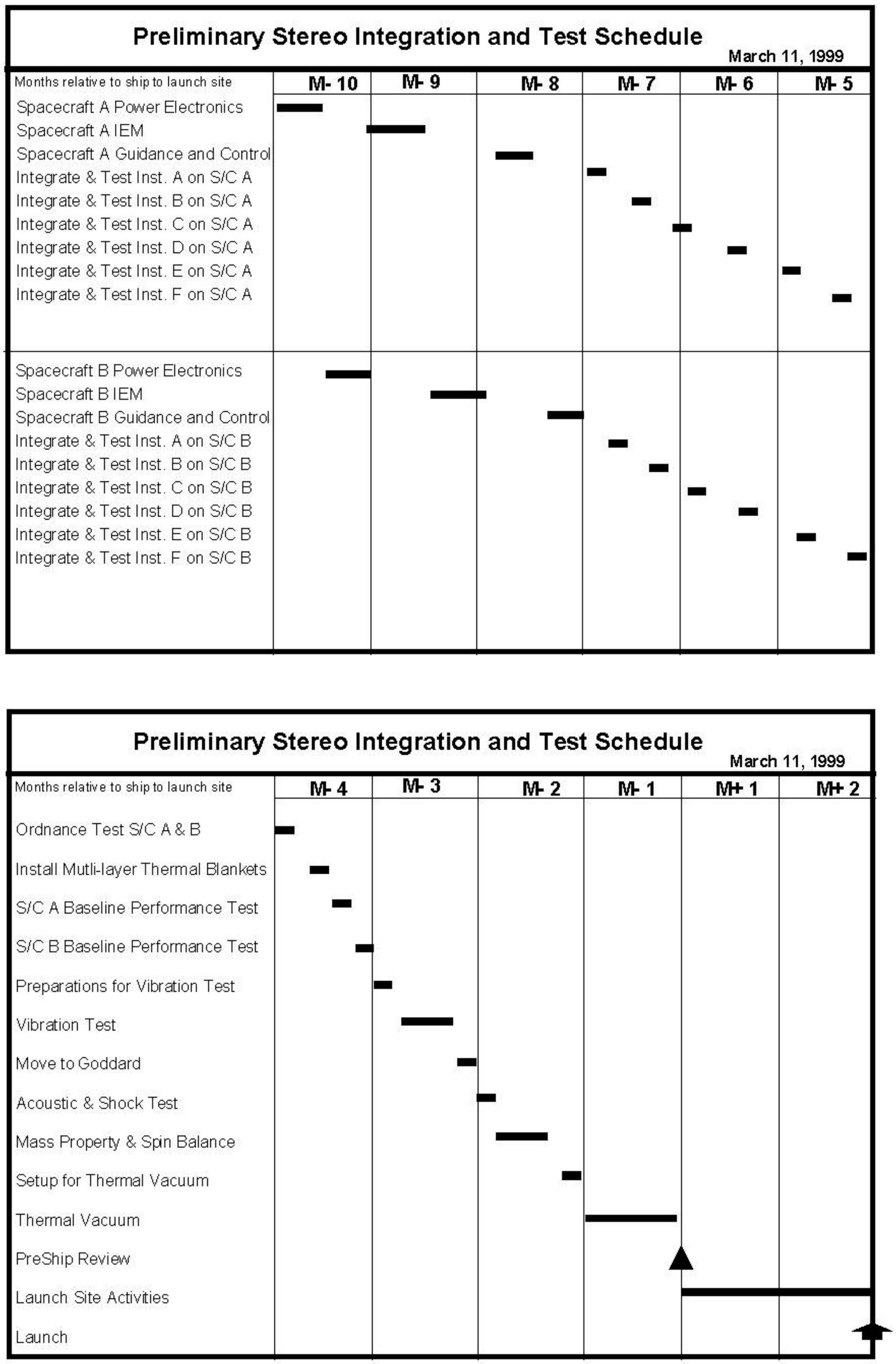


Figure 5-1 STEREO Integration and Test Flow

6.0 MISSION OPERATIONS

Overview. Figure 6-1 shows the STEREO Mission Operations System (MOS) which consists of the two STEREO spacecraft, DSN ground stations, Mission Operations Center (MOC), the Science Operations Center (SOC) and their respective operational teams. The STEREO spacecraft will be operated by APL utilizing the DSN for communications with the spacecraft after launch. The spacecraft bus and the instrument suite will be operated in a decoupled fashion. The MOC will support all spacecraft bus operations and the SOC will operate all instruments on both spacecraft, although communication between the SOC and the spacecraft will necessarily flow through the MOC. All spacecraft servicing, including commanding and data recovery will occur during a single (nominal) ground contact, or track, each day. This track will extend over a two to eight hour window, depending on the spacecraft's range from Earth. Spacecraft command messages will be uploaded and real-time engineering data will be downloaded and evaluated to assess spacecraft health. The Solid State Recorder (SSR) will be played back on each contact and all science data will flow to the SOC in near real-time.

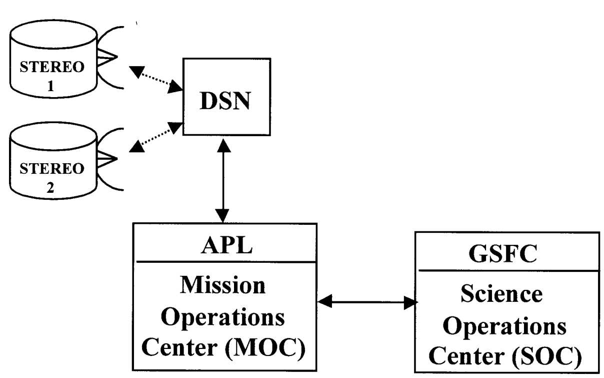


Figure 6-1 STEREO Mission Operations System

6.2 Mission Operations Center

The MOC has primary responsibility for management of the spacecraft bus including the development of operational timelines with associated command sequences and the uplink to the spacecraft by way of the DSN. Recovery of spacecraft bus engineering telemetry and the analysis of this telemetry is also performed at the MOC. The MOC receives instrument command sequences (packets) from the SOC and, after verification, queues them for uplink to the spacecraft based on start and expiration times appended to the command messages by the SOC. The MOC also distributes the downlinked science data and necessary operational data products to the SOC. The MOC is operated by the Mission Operations Team (MOT) and is located at JHU/APL in Laurel, MD.

6.2.1 Mission Operations Team

The MOC is staffed and operated principally by the Mission Operations Team (MOT). Staffing of the MOT will begin during the development phase of the program. Every MOT staff member will have a detailed knowledge of the operation and constraints of both STEREO spacecraft and MOS. The MOT will be assigned functional responsibilities necessary to provide both an education and essential tasks in support of the Spacecraft Bus Engineering Team (SBET) as well as the Integration and Test (I&T) Team. The MOT will support the SBET during the testing of the subsystems prior to delivery to the I&T Team. Components of the actual MOC will be employed to support subsystem testing and the I&T phase of the program. These components will be used to develop of databases, display formats and command sequences which are necessary to support subsystem tests. These items may be brought forward to the spacecraft system level support effort.

During the I&T phase, the MOT will be part of the I&T Team. They will define and produce the necessary system level tests to support the conduct of mission simulation tests. The spacecraft will be tested in the same manner as it will eventually be operated on-orbit. During test, the MOT Spacecraft Specialists will provide direct support to the Test Conductor as members of I&T Team. During this time, the function of the MOT will be to provide an assessment of the performance of the spacecraft subsystem under test. The MOT will assume the role of the Test Conductor during certain times within the I&T phase.

On-orbit mission simulations, where the spacecraft is operated as if it were on-orbit, will be conducted during the I&T phase. These tests will be conducted by the MOT just as they will during the actual on-orbit phase of the mission. All external operations supporting organizations and facilities (DSN and SOC) will be invited to support these tests. These tests will become the rehearsals of the MOT and the entire MOS.

Once on-orbit, the MOT is responsible for all spacecraft commanding, recovery of spacecraft telemetry, assessment of the telemetry, and the control, monitoring and performance assessment of all ground components necessary to support these functions. The MOT is also responsible for supporting SOC mission planning activities. During the Normal Operations mission phase, the MOT staff will be comprised of the following:

- Flight Operations Manager
- Spacecraft Specialists (two/vehicle)
- DSN Scheduler
- System Maintenance Engineer

During the Early Operations mission phase, the MOC will be staffed 24 hours/day and seven days/week. As a goal, during Normal Operations the MOT will work business hours, five days/week. This will require validation of many automated MOC procedures and autonomy rules on the spacecraft. Occasional off-business hours scheduling is likely to occur during some special operations including contingency activities.

6.2.2 Mission Operations Team Activities

Operations planning will consist of the following activities necessary to support a scheduled track:

- Track scheduling
- Maintenance activity scheduling
- Managing the uplinking of instrument commands
- SSR management
- Timekeeping management
- Navigation management
- Track Plan Generation

The STEREO operations' planning consists of planning a week of tracks in advance. The MOT will determine the operational requirements of the spacecraft bus over the next week and will prepare the necessary command packets to satisfy these requirements. Operations planning and assessment activities for all instruments will be conducted by the SOC.

6.3 Operations Control and Assessment Activities

6.3.1 Data Flow

Figure 6-2, illustrates the flow of command and telemetry data between the ground-based spacecraft bus elements, instrument operations elements and the on-orbit STEREO spacecraft. The 'outer-loop' depicts instrument operations. Using a decoupled instrument operations approach, all instruments will be operated by the instrument operations team at the SOC. In Figure 6-2, SOC Planning begins on the far right, where instrument commands are produced. These command messages, which will be packetized along with some additional information needed by the MOC, are transmitted

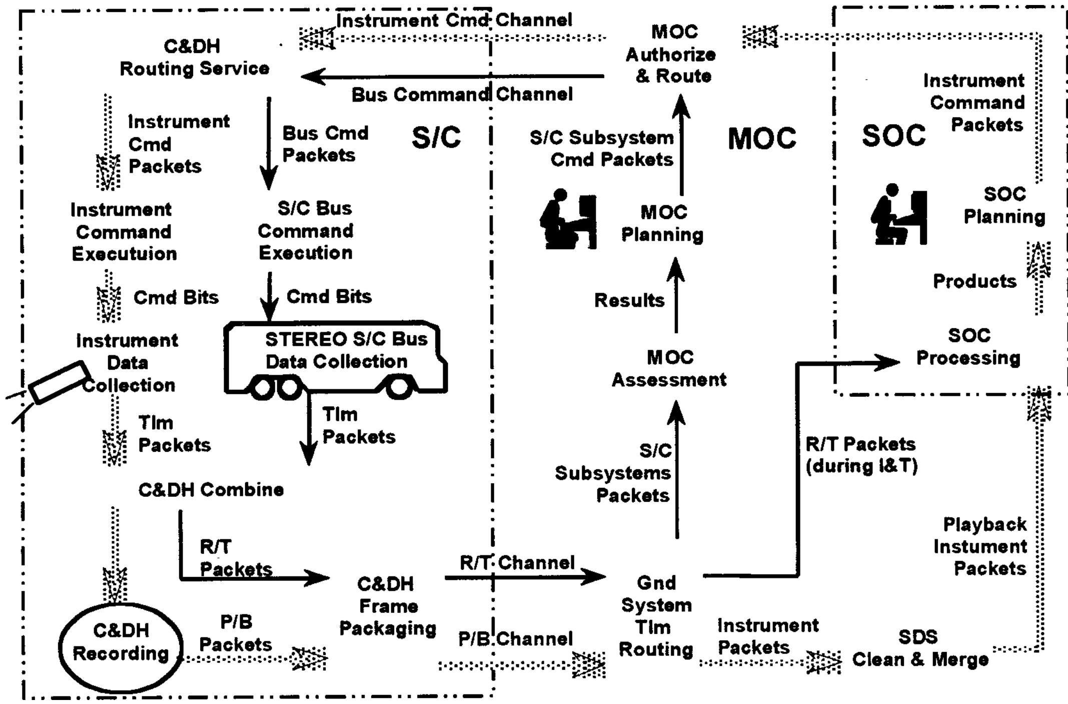


Figure 6-2 STEREO Mission Data Flow

to the MOC via the Internet. At the MOC (MOC Authorize and Route) authorization and checking is performed, then these commands are queued for uplink to the instrument. Along with the command packets, the SOC appends timing information which indicates the time span (earliest and latest times) over which the command packet may be uplinked to the instrument. Real-time command packets, when uplinked to the spacecraft, are immediately routed by the spacecraft bus Command and Data Handling (C&DH) processor (the C&DH Routing Service) to the appropriate instrument. They are time tagged and macro command packets are stored in the instrument's allocated storage locations in the C&DH processor. Conceptually, the command packet goes 'directly' from the SOC to the instrument, since the MOC, ground station and spacecraft bus are merely the delivery system. This delivery system notifies the SOC of the delivery status of the command message.

Whereas the SOC produces instrument commands, the MOC produces spacecraft bus commands. This is depicted in the 'inner-loop' on the Data Flow diagram. Starting at the MOC Planning process, the MOT prepares command messages which operate spacecraft bus during the next day. These command messages are queued for uplink (MOC Authorize and Route) just like the instrument commands, only they go to a different destination (via the C&DH Routing Service). The C&DH processor immediately routs real-time commands, to the appropriate spacecraft subsystem and time tagged and macro commands are stored in the C&DH processor. The MOC receives delivery status of the command packets just as the SOC does.

Once these commands have been executed (Instrument Command Execution and Spacecraft Bus Command Execution) on the spacecraft, the operations become integrated. The instruments produce science and engineering data (Instrument Data Collection) in response to the uplinked command messages. The data produced by the instruments is sent to the spacecraft data system in the form of CCSDS telemetry packets. Similarly, engineering data produced by the spacecraft bus, is also formatted into CCSDS packets. These packets, produced by the instruments and the spacecraft bus, are stored on the SSR within the spacecraft data system (C&DH Recording). During a track with the spacecraft, the contents of the SSR are transmitted to the MOC (C&DH Frame Packaging).

On the ground (Ground System Telemetry Routing), real-time data is forwarded to the MOC and to the STEREO Data Server (SDS), while all recorded data is sent to the server facility (SDS Clean and Merge). All instrument data will be sent to the SOC for processing and analysis. The cycle repeats, with the SOC preparing instrument commands for the next time period. Spacecraft bus data is routed to the MOC (MOC Assessment) where an assessment function is performed. The MOC spacecraft bus planning process then repeats.

Of significance is that the instruments and spacecraft bus are operated (almost) entirely independent of each other. The same can be said about the ground elements (the SOC and the MOC). This decoupling of instrument operations concept greatly simplifies the operations process, which traditionally requires these functions to be merged in a complicated manner. For additional information on Mission Operations, see the Concept of Operations in Appendix H.

6.4 STEREO Ground System forMission Operations, Integration and Test, and Field Operations

The STEREO Ground System (GS) supports the sending of commands to the STEREO spacecraft and the display/distribution of telemetry data. The baseline for the STEREO Ground System is to use a modified TIMED MOC at APL. This system is capable of operating the TIMED mission while the STEREO Mission is

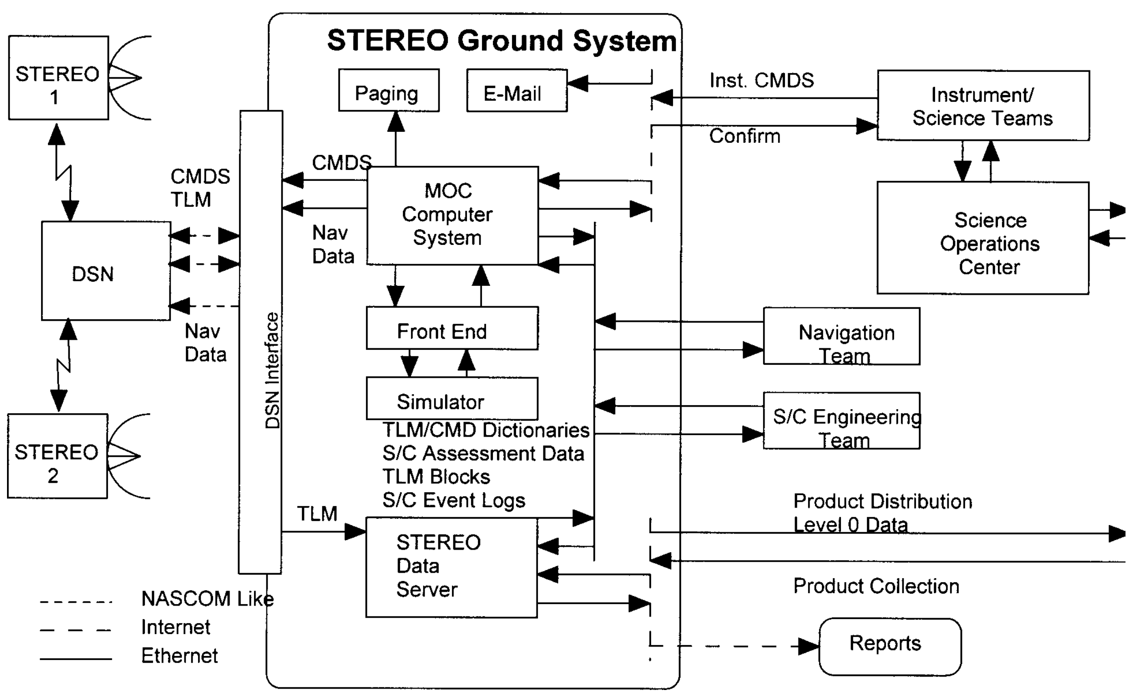


Figure 6-3 STEREO Ground System Normal Operations Functional Block Diagram

performing Integration and Test, Launch, and Normal Operations. The TIMED GS will be compatible with the STEREO Spacecraft due to the similarities of the TIMED and STEREO spacecraft bus. The STEREO Ground System, as configured for Normal Operations, Integration & Test, and Field Operations, is described in the following subsections.

6.4.1 Normal Operations

The STEREO Ground System includes all hardware, software, data links, and facilities used to plan tests and operations; generate and uplink commands; and receive, process, analyze, and disseminate telemetry and test data. The majority of the STEREO GS design is inherited from the TIMED program. The STEREO GS will be capable of performing unattended spacecraft contacts and uplink commands, downlink science and housekeeping telemetry, and perform basic state of health verification from the housekeeping telemetry. The GS will have the capability to detect anomalous conditions, page STEREO spacecraft specialists and send basic information from the anomalous contact to the spacecraft specialist via the Internet. The GS will be capable of sending telemetry data and MOT data products to the SOC and receiving STEREO Instrument commands from the SOC via the Internet. Commands and telemetry will flow to and from the Deep Space Network (DSN) via a DSN.

The transfer of telemetry data and mission operations data products to the Science Data Center will be accomplished by the STEREO Data Server (SDS). For the purpose of trending and assessment, the SDS will archive all STEREO Bus telemetry for the duration of the mission. The Spacecraft Engineering Teams and the Navigation Team will be able to access the STEREO Bus telemetry via the SDS and the MOC Computer System. The MOC Computer System will consist of several workstations and X-Terminals and/or Intel based PCs which will use a COTS based command and telemetry system as well as several custom software packages. Custom software packages will handle the following tasks:

- contact planning and scheduling
- command load generation (including merging the Instrument commands from the Science Teams and sending acknowledgement back to the Science Teams)
- Solid State Recorder Management
- Engineering dump data display
- processor dump data display, analysis and trending
- Spacecraft Timekeeping,
- Ground System and Spacecraft autonomy management.

6.4.2 Integration and Test (I&T)

During the I&T phase, the MOC depicted in Figure 6-3 will be supplemented with Ground Support Equipment (GSE) to enable testing and simulation of the STEREO mission. During this phase, the DSN Interface will be replaced with a Front End which will provide an interface and serve data to the STEREO Ground System and record all raw telemetry. The Instrument Teams will be able to accomplish commanding and telemetry analysis either remotely as they will during the Operations phase or locally at the MOC. The dashed lines indicating how the Instrument Team's data flows during I&T is still being traded. It is expected that the SOC will be on-line and flowing products during I&T on a trial basis. Figure 6-4 depicts the STEREO GS during the I&T phase of the program.

The STEREO GS will be used extensively during this phase of the mission for all STEREO commanding, telemetry displays, analysis, and distribution of spacecraft data. During this phase, the GS will connect to the STEREO GSE via an Ethernet interface. The GSE will consist of Instrument GSE, Guidance & Control GSE,

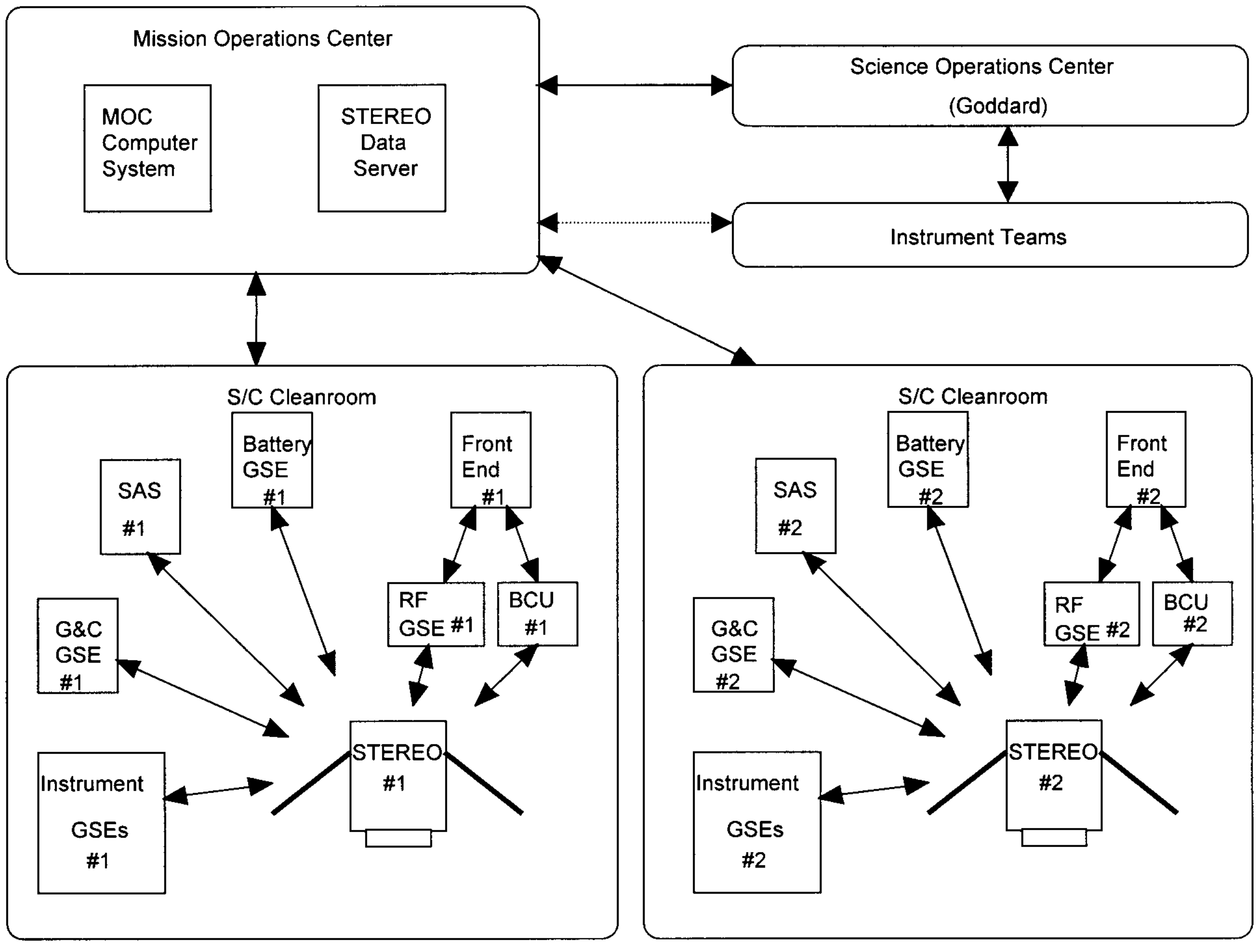


Figure 6-4 STEREO Ground System Block Diagram for Integration and Test

Solar Array Simulator, Battery GSE, RF GSE, Blockhouse Control Unit (BCU), and a Front End Processor for each of the STEREO Spacecraft. The STEREO GS will be able to command and receive telemetry from both Spacecraft simultaneously. All I&T commanding will be possible using the STEREO GS.

6.4.3 Field Operations

Figure 6-5 is a block diagram of the STEREO GS configuration during field operations. This phase of the mission includes environmental testing at the Goddard Space Flight Center (GSFC) and Launch Operations at the Kennedy Space Center (KSC). The testing and operations will be performed like Integration and Test in that the commands will be sent from the STEREO GS at APL and telemetry will be received and analyzed at the MOC. All instrument teams will now be interfacing directly with the SOC. The MOC connection to the STEREO Spacecraft will be through a NASCOM-like interface via the GSFC. The link must support the flow of telemetry and commands for both spacecraft simultaneously. This interface will most likely be a TCP protocol interface. This interface will also require several voice circuits for communication between the STEREO MOC and the field operations personnel.

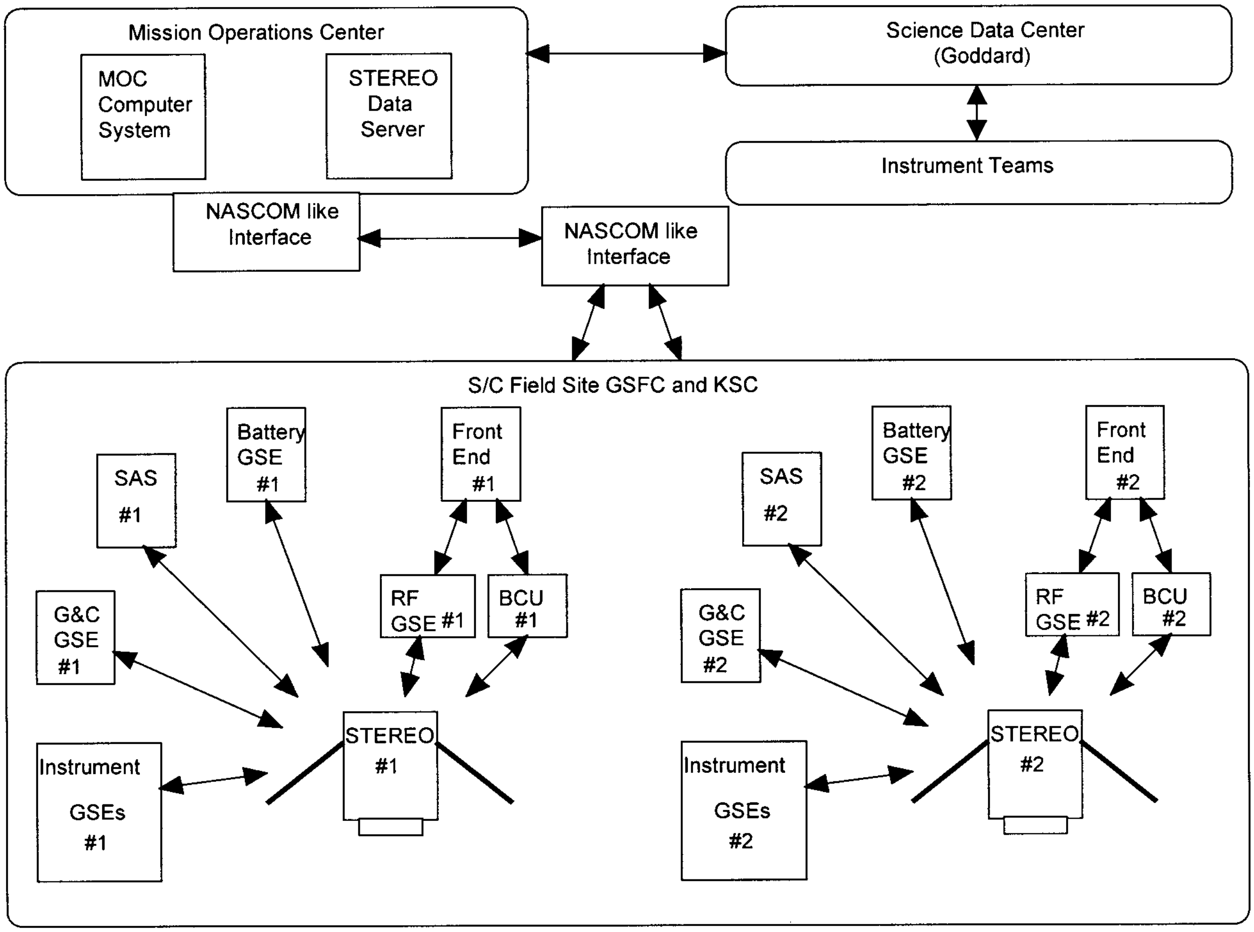


Figure 6-5 STEREO Ground System Block Diagram for Field Operations

APPENDIX A

ACRONYM DEFINITIONS

A/D	Analog to Digital
ACE	Advanced Composition Explorer
ADC	analog to digital converter
AFC	Attitude Flight Computer
AIE	Attitude Interface Electronics
AO	Announcement of Opportunity
AOS	Acquisition of Signal
APL	Applied Physics Laboratory
ARW	angular random walk
AU	Astronomical Unit
BCU	Blockhouse Control Unit
BER	Bit Error Rate
BW	bandwidth
BWG	Beam Wave Guide
C&DH	Command and Data Handling
C&T	Command and Telemetry
CASE	Computer Aided Software Engineering
CCB	Configuration Control Board
CCD	Charged Coupled Device
CCSDS	Consultative Committee for Space Data Systems
CDR	Critical Design Review
Cg	center of gravity
CLTU	Command Link Transmission Unit
CME	Coronal Mass Ejection
CONOPS	Concept of Operations
CONTOUR	COmet Nucleus TOUR
CORE	Orbiter/Cargo Standard Interfaces
COTS	Commercial off-the-Shelf
CP	center of pressure
CPU	Central Processing Unit
CTT	Compatibility Test Trailer

DB	database
DET	direct energy transfer
DP	data processing
DPAF	Dual Payload Adapter Fitting
DPU	Data Processing Unit
DS	Data Server
DSAD	Digital Solar Attitude Detector
DSN	Deep Space Network
DTF	DSN Transponder Facility
ECR	Engineering Change Request
EEPROM	Electrically Erasable Programmable Read-Only Memory
EIRP	Effective Isotropic Radiated Power
ELV	expendable launch vehicle
EMI	Electro-Magnetic Interference
EOL	End of Life
EPD	Energetic Particle Detector
ESD	electrostatic discharge
ETR	Eastern Test Range
FOG	Fiber-optic gyros
FOV	Field of View
FPGA	Floating Point Gate Array
FPGAs	field programmable gate arrays
FSS	Flight Support System
FST	Factor of Safety for Test
FSU	Factor of Safety for Ultimate Strength Design
FSY	Factor of Safety for Yield Strength Design
G&C	Guidance and Control
G/T	Gain/Temp
GCC	Guidance and Control Computer
GEVS	General Environmental Verification Specification
GIIS	General Instrument Interface Specification
GPS	Global Positioning System

GRO	Gamma Ray Observatory
GS	Ground System
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
GTDS	Goddard Trajectory Determination System
HEF	High Efficiency Front-end
HENA	High Energy Neutral Atom
HGA	High Gain Antenna
HI	Heliospheric Imager
HRG	Hemispherical Resonator Gyros
I&T	Integration and Test
I/O	Input/Output
I^2C	Inter-Integrated Circuit
ICD	Interface Control Document
ICE	International Cometary Explorer
ID	Identifier
IDT	Instrument Design Team
IEM	Integrated Electronics Module
IMAGE	Imager for Magnetopause-to-Aurora Global Exploration
IMU	Inertial Measurement Unit
IRU	Inertial Reference Unit
ISEE	International Sun Earth Explorer
ISI	Integral Systems, Inc.
ITO	Indium Tin Oxide
JHU/APL	The Johns Hopkins University Applied Physics Laboratory
JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
LEO	Low Earth Orbit
LGA	Low Gain Antenna
LMLV	Lockheed Martin Launch Vehicle
LOS	Loss of Sun
LOW	Launch Opportunity Window

LV	launch vehicle
LV/SV	Launch Vehicle/Space Vehicle
LVS	Low Voltage Sense
MA	Multiple Access
MAG	Magnetometer
MAP	Microwave Anisotropy Probe
MCD	Maximum Liklehood Convolutional Decoder
ME	Maintenance Event
MET	Mission Elapsed Time
MGA	Medium Gain Antenna
MIL	Merritt Island Launch
MIPS	million instructions per second
MIRT	Mission Integration Readiness Test
MLI	Multi-Layer Insulation
MOC	Mission Operations Center
MOGS	Mission Operations Ground Segment
MOS	Mission Operations System
MOT	Mission Operations Team
MPE	Maximum Expected Environment
MPT	Mission Planning Team
MSU	Margin of Safety on Ultimate Strength
MSY	Margin of Safety on Yield Strength
M_v	Magnitude-Variable
N/A	not applicable
NASA	National Aeronautics and Space Administration
NAV	Navigation
NEAR	Near Earth Asteroid Rendezvous
NISN	NASA Integrated Service Network
NOAA	National Oceanographic and Atmospheric Administration
NOCC	Network Operations Control Center
NRL	Naval Research Laboratory

NSTS	NASA SpaceTransportation System
OAP	Orbit Average Power
OASPL	overall sound pressure level/overall sound power level
OCEAN	Orbit/Covariance Estimation and Analysis
OSC	Ultra Stable Oscillator
OSU	Ohio State University
PB	Playback
PCI	Peripheral Component Interconnect
PDR	Preliminary Design Review
PI	Principal Investigator
POC	Payload Operations Center
POCC	Payload Operations Control Center
PPT	Peak Power Tracker
PR	Problem Report
PROM	Programmable Read Only Memory
P_S	probability of success
PSD	Power Spectral Density
PSE	power system electronics
PWM	Pulse Width Modulation
QARL	Quality Assurance Requirement Level
RBT	Radio Burst Tracker
RF	Radio Frequency
RIU	Remote Interface Unit
RLG	ring-laser gyros
RMDC	Radiometric Data Center
RMS	Root-Mean-Square
ROM	Read Only Memory
RPM	rotations per minute
RS	Reed Solomon
RT	Real-time
RTW	Real-Time-Workshop™
RWA	Reaction Wheel Assembly

S/C	Spacecraft
S/W	Software
SA	Solar Array
SBET	Spacecraft Bus Engineering Team
SCIP	Solar Corona Imaging Package
SDS	STEREO Data Server
SDT	Science Definition Team
SEL	Software Engineering Laboratory
SIIS	Specific Instrument Interface Specification
SMEX	Small Explorer
SOC	Science Operations Center
SOHO	Solar and Heliospheric Observatory
SOMO	Space Operation Management Office
SPE	Sun-Probe-Earth
SQAE	Software Quality Assurance Engineer
SRAM	Static Random Access Memory
SSA	Solid State Amplifier
SSR	Solid State Recorder
STEREO	Solar TErrestrial RElations Observatory
STF	Supplemented Telemetry Frame
STP	Supplemented Telemetry Packet
STS	Space Transportation System
SWAS	Sub-millimeter Wave Astronomy Satellite
SWPA	Solar Wind Plasma Analyzer
TBD	to be determined
TDM	Time Division Multiplex
TF	Telemetry Frame
TIMED	Thermosphere, Ionosphere, Mesosphere, Energetics and Dynamics
TLM/CMD	Telemetry/Command
TOGS	Test Operations Ground Segment
TP	Telemetry Packet
TRACE	Transitional Region and Coronal Explorer

TWTA	Travelling Wave Tube Assembly
UHF	Ultra-High Frequency
USA	United Space Alliance
USO	Ultra-Stable Oscillator
UT	Universal Time
UTC	Coordinated Universal Time
UTMC	United Technologies Microelectronics Center
V	Volts
VHF	Very-High Frequency
VLBI	Very Long Baseline Interferometry
W	Watts
WIRE	Wide-Field Infrared Explorer
WWW	World Wide Web
XB	X-Band
XMIT	Transmit

APPENDIX B

Software Development
and
Management Plan

Table of Contents

1.0 Introduction

This document describes the management approach for all software developed or procured by JHU/APL for the Solar TErrestrial RElations Observatory (STEREO) mission. Cost constraint is the primary challenge facing this mission. Therefore, the goal of this plan is to define a management approach and development guidelines that are sufficient to monitor software development and ensure the use of good engineering principles, yet not unduly burden developers with more formal reviews and documentation than are required to meet mission objectives. This plan establishes guidelines for the management, engineering, and quality assurance for the providers of STEREO software.

1.1 Scope

This management plan addresses all operational software to be developed or procured by JHU/APL for STEREO. JHU/APL has responsibility for two primary elements: the Spacecraft Bus and the Mission Operations Center. Science instruments, the Science Operations Center, and their corresponding flight and ground software are the responsibility of the Goddard Space Flight Center (GSFC).

The plan defines policy regarding providers of all software purchased, contractually acquired, developed or maintained by JHU/APL or its subcontractors for the STEREO project. It describes management mechanisms, the technical approach to the software development life cycle, required reviews and documentation, and configuration management.

1.2 Applicable Documents

(1) SRS-yy-nnn *Space Department Software Quality Assurance Guidelines 2 (Currently in Draft Form)*
(2) CLO-9805 *The Johns Hopkins University Applied Physics Laboratory Quality Assurance Plan*, August 1998
(3) APL-SDO-9989 *APL Space Department Software Quality Assurance Guidelines*; October 22, 1992.
(4) *GSFC Software Engineering Laboratory Software Development Process Guidelines*
(5) STEREO Software Quality Assurance Plan
(6) *UML Distilled*, Martin Fowler with Kendall Scott, 1997

1.3 Acronyms and Abbreviations

C&DH	Command and Data Handling
CASE	Computer Aided Software Engineering
CCB	Configuration Control Board
CDR	Critical Design Review
COTS	Commercial off-the-Shelf
DSN	Deep Space Network
ECR	Engineering Change Request
G&C	Guidance and Control
GSE	Ground Support Equipment

GSFC	Goddard Space Flight Center
I&T	Integration and Test
JHU/APL	The Johns Hopkins University Applied Physics Laboratory
MOC	Mission Operations Center
NASA	National Aeronautics and Space Administration
PDR	Preliminary Design Review
PR	Problem Report
QARL	Quality Assurance Requirement Level
SEL	Software Engineering Laboratory
SOC	Science Operations Center
SQAE	Software Quality Assurance Engineer
STEREO	Solar TErrestrial RElations Observatory
TBD	To Be Determined

2.0 Overview of STEREO Mission Components

Figure 2-1 is a block diagram showing the primary STEREO mission components.

The flight components consist of the Command and Data Handling (C&DH) system, Guidance and Control (G&C) system, and instruments for two identical spacecraft that will study solar phenomena. JHU/APL is responsible for the C&DH and G&C systems, while GSFC will select teams that will be responsible for developing the instruments.

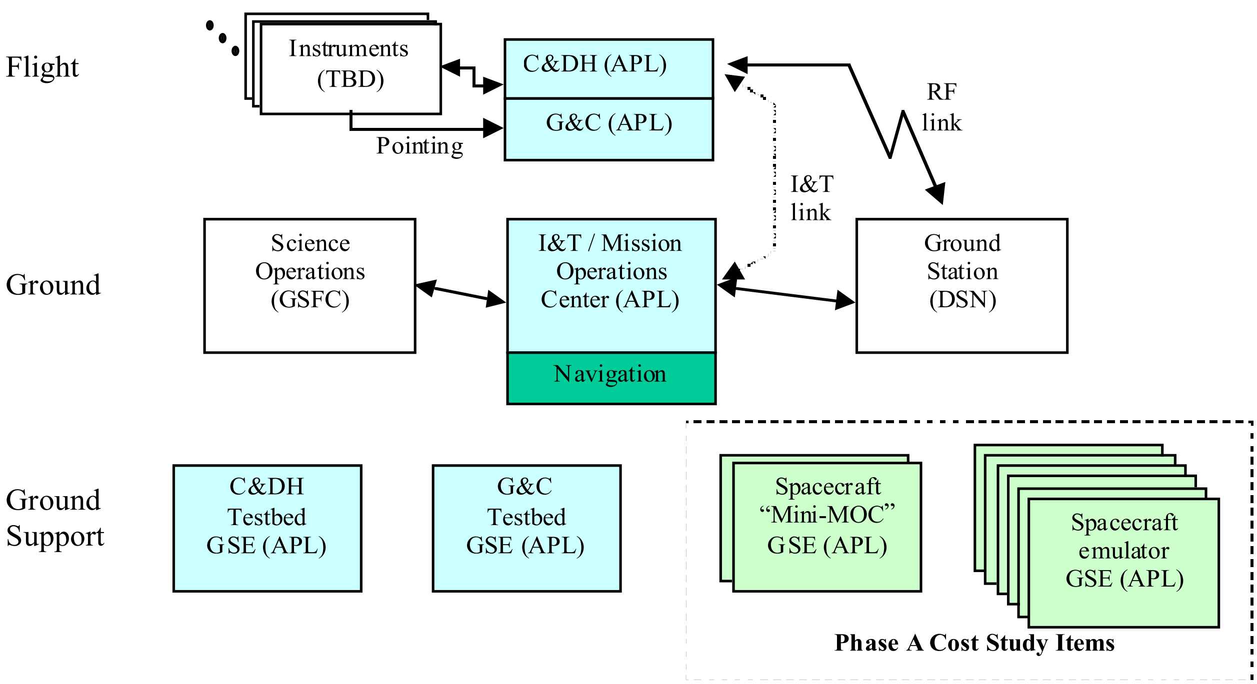


Figure 2-1. STEREO Mission Components

The major ground components consist of the Science Operations Center (SOC) to be developed and run by GSFC, the Mission Operations Center (MOC) to be developed and run by JHU/APL during spacecraft Integration and Test and flight operations, and NASA's Deep Space Network (DSN) ground stations. In addition to performing mission command, control, and monitoring functions, the MOC also includes a real time simulator for testing scenarios. JHU/APL also has a navigation team that works closely with the MOC and DSN to perform orbit determination, predictions, and updates for the mission.

In addition to the primary flight and ground system deliverables, a number of Ground Support Equipment (GSE) subsystems are required to support development and testing of deliverable hardware and software. These items, which include simulators, stimulators and bench test equipment, are generally software-based subsystems in their own right.

3.0 Software Development Process

The STEREO software development process will conform to the JHU/APL Space Department's software quality assurance guidelines described in Reference 1. The JHU/APL Space Department's Software Process Engineering Team is developing this document. The document is currently in draft form. This document will establish guidelines for software developed by the Space Department, based on its predecessor (Reference 2) and recommendations from GSFC's Software Engineering Laboratory (Reference 3).

The document outlines a process that emphasizes early coordinated analysis of the flight, ground, and GSE requirements, development of an end-to-end data architecture to meet those requirements, and an iterative design, coding, and test cycle for each subsystem. Each iteration of this cycle "builds production-quality software, tested and integrated, that satisfies a subset of the requirements". (Reference 6, p. 15) The test environment and scripts are developed in parallel with the deliverable system.

Reference 1 defines four Quality Assurance Requirement Levels (QARLs), and requires each program deliverable to be assigned a level according to its intended use (e.g., criticality). QARL 1 is the most stringent, while QARL 4 is least. All flight software and MOC software with the functions of generating and transmitting commands to the spacecraft will use QARL level 2. All other deliverable software will use QARL level 3. Overall, the STEREO program is designated to use QARL level 2.

The draft of Reference 1 that is currently available refers to enclosures that have yet to be delivered. They will provide guidance for tailoring the SEL software development process to individual programs. The final version of this software development plan will follow those guidelines as closely as possible within the cost and schedule constraints of the STEREO program.

4.0 Software Development

The software development process for STEREO will be uniform for all deliverable software components, whether they are defined as QARL 2 or 3. The primary differences will be in the formality of the documentation and the reviews required. Cost and schedule can both benefit from a standard process that emphasizes architecture and design, an adequate level of documentation, and a comprehensive review and testing program. Rather than cutting important steps from the process in an attempt to reduce costs, STEREO will instead emphasize the use of appropriate tools to reduce the effort required in carrying out those steps.

The final version of this document will contain the identification of individual deliverable software components, their associated QARLs, and the specific documents, reviews, and tests that will be required for each.

4.1 Software Development Environment

The STEREO software development environment for flight, ground, and test software will be based on a set of support tools that assists STEREO system engineers, software architects, designers, developers and testers, and program managers in performing their tasks related to STEREO software.

The tool set will be centered on a common electronic information repository that will contain (at minimum) the following:

- a requirements database that tracks fundamental and derived requirements for all subsystems, as well as requirement dependencies between subsystems.
- current versions of software documents, including requirements specifications, architectural design documents, interface control documents, detailed design documents, user manuals, etc.
- a configuration management system and database that contains current versions and revision histories of all software, with the ability to track and recreate previous versions of full builds
- software design aids/CASE tools that help software engineers to focus on design more than coding, and to produce design documentation easily in a standard format that both developers and reviewers can understand.
- tracking and planning information such as build schedules, bug reports, etc.

The primary purposes of the software development tool set are to assist the software teams in developing quality software and to promote good communication within each subsystem team, amongst the teams, and between the software teams and program management. It is important to realize that tools require training as well as some support effort themselves, and that use of a common tool set by many development teams in different hardware environments can impose unwanted constraints on the teams. For these reasons, selection of an appropriate tool set that minimizes cost/benefit ratios will be an important part of the software-planning phase of the project.

5.0 Configuration Management and Control

Configuration management is a process for tracking software versions, for controlling access to files in a multi-user development environment, and for documenting software development progress throughout the program. Configuration control is a process for reviewing proposed changes, assessing the costs and benefits of implementing them, approving or disapproving implementation of the changes, and documenting the decisions. The following subsections give an overview of the configuration management and configuration control mechanisms that the STEREO software team will utilize.

5.1 Configuration Management

The configuration management system/tools described in Section 4.0 will be in place before any deliverable software is developed. Software developers will receive training on using the system during the early phases of the program. Configuration management of all deliverable software files will begin at the time of creation of each file and will continue throughout the mission.

5.2 Configuration Control

The STEREO Configuration Control Board (CCB) has responsibility for managing both hardware and software configuration items. An item that has been placed under configuration control may not be modified nor granted a requirements waiver without explicit approval of the CCB. Members of the STEREO CCB include: the Mission system engineer, Mission software lead, Software I&T lead, Software quality assurance engineer and Project science representative.

STEREO software items will be placed under configuration control as follows:

Documents: Required formal documents will be placed under configuration control at the time of their sign-off. Software Requirements Specifications and software sections of Interface Control Documents will be signed off at or prior to the subsystem software PDR. Software Test Plans will be signed off before the beginning of the acceptance test phase, although nearly-complete drafts should be in place by the time of the subsystem software CDR. Documents that have been updated to reflect approved ECRs will carry a new revision letter when reissued, and will indicate the changes.

Software. Software, including COTS components, will be placed under configuration control at the start of subsystem acceptance testing, which precedes delivery of the final subsystem build to I&T. Some subsystems will include stored macros, table-driven functions and/or scriptable functions can be modified without technically making a change to application code. Any such scripts or data structures, when loaded by default upon boot or startup of the application code, shall be treated as part of the code image and will be under configuration control. The final version of this document will define the build/version-numbering scheme that provides for tracking of approved software modifications.

5.2.1 Engineering Change Request

An Engineering Change Request (ECR) may be submitted to the CCB to request a change to a configuration item. For software, the ECR includes fields to describe the requested change, the reason for the change, the scope of the change (requirements, interfaces, design, code, test), and an impact assessment including possible workarounds and testing that would be required to re-qualify the modified code. The CCB has the authority to approve or reject the ECR. The STEREO Software Quality Assurance Engineer will track the status and final disposition of all ECRs.

5.2.2 Problem Reports

A Problem Report (PR) form will be used to track apparent errors encountered during and subsequent to the I&T. For software, the PR will include fields that lists the originator, describes the problem, the environment/circumstances in which the problem arose, and identification of the subsystem(s) involved. The PR will be routed to the cognizant engineer(s) to who will complete the form by writing a problem analysis and a recommended course of action. These courses of action include:

- none—procedural error
- fix—problem is due to software error (a description of the corrective action will be included when a “fix” is recommended)
- waiver—use operational workaround or impact is insufficient to warrant change

The PR may thus result in an ECR being written to authorize a change to a configuration item. PRs will not be considered closed out until signed off by the designated lead engineer and CCB representatives.

6.0 Software Maintenance

The CCB will continue to provide software configuration control throughout the life of the STEREO mission. Updates or modifications to any ground or flight software will be permitted only with an approved ECR. This includes updates or modifications to COTS components such as commercial databases or planning tools. Before it may be used operationally, any approved software change must undergo re-qualification with the pertinent acceptance test procedures and simulations/GSE. The Mission Operations Center will maintain a real time spacecraft simulator which will provide a means of testing modifications to flight software prior to upload to the spacecraft.

APPENDIX C

Data Processing Unit Conceptual Design

A conceptual design for an instrument Data Processing Unit (DPU) was completed as part of the STEREO Pre-Phase A study. Although the current spacecraft baseline does not include a DPU, a design has been developed for use should there be instruments proposed that could benefit from one. A DPU would be beneficial in two cases. In the first case, the proposed instrument is an existing design that is incompatible with the STEREO spacecraft bus. A DPU would be used to adapt the instrument to the STEREO spacecraft and minimize instrument redesign costs. In the second case, the proposed instrument is a new design with modest data processing requirements. A DPU would be shared among multiple instruments to reduce total spacecraft cost.

The DPU would act as an interface unit between the instruments and the fixed interfaces of the STEREO spacecraft. Four of the instruments defined in the STEREO Instrument Announcement of Opportunity (AO) are considered candidates for utilizing an instrument DPU. These are the Magnetometer, Radio Burst Tracker, Energetic Particle Detector and the Solar Wind Plasma Analyzer. These instruments have relatively low data bandwidths and data processing requirements making them suitable candidates for sharing a common DPU.

The interfaces between the DPU and the STEREO spacecraft are limited to one primary power circuit at a nominal +28 VDC and a MIL-STD-1553 data bus for commands, telemetry and time distribution. The interfaces between the DPU and the instruments will be defined only after final instrument selections have been made. For the purposes of this study, it has been assumed that the instruments receive formatted digital commands and output formatted digital telemetry. A science data path consisting of analog signals between an instrument and the DPU could be accommodated but is not recommended due to integration, calibration and noise issues.

The primary task of the DPU would be to reformat instrument commands and telemetry for compatibility with the STEREO spacecraft MIL-STD-1553 data bus. In addition, the DPU would provide data processing and data compression services (if necessary) to the instruments as DPU processing capability and bandwidth permit. The DPU would also time tag instrument data as required and accommodate a limited amount of voltage, current and temperature monitoring for the instruments.

The design of the STEREO Instrument DPU is a second-generation design based upon the instrument DPU designed for the Near Earth Asteroid Rendezous (NEAR) spacecraft and used on the Advanced Composition Explorer (ACE) and Cassini spacecraft. It is a modular computer system consisting of a core set of boards with custom instrument interface cards added as required. The core set of boards consists of a processor board, a MIL-STD-1553 interface board, a housekeeping voltage, current and temperature monitoring board and a DC/DC converter board. The instrument interface boards would be application specific and would be based upon field programmable gate arrays (FPGAs) with memory components added as needed to buffer instrument data. Second-generation enhancements to these boards include updated memory components, updated FPGAs and stacking inter-board connectors. The stacking inter-board connectors eliminate the system motherboard thereby reducing system size, mass and cost.

The modular nature of the design is enhanced with a modular electronics chassis similar to that used by the APL Command and Data Handling System in Your Palm project. Each four-inch by four inch electronic board, with stacking inter-board connector, is housed in a frame. When stacked and connected with the other electronic boards in the system the boards form the top, bottom, front and back walls of the electronic chassis. End plates are added and the entire assembly is through-bolted together to form a completed unit. This approach minimizes chassis design costs because only three piece parts need be designed: the 0.50 inch thick board frame used by most boards, the 0.75 inch thick board frame used by the DC/DC converter boards and the endplate. A system with any number of boards can be built from these piece parts. The board frames will be modified by machining openings for right angle micro-miniature “D” connectors as required for each board.

A block diagram of the DPU is shown in Figure C-1. Power and mass figures are given in Table C-1. The DPU consists of the four core boards and two application specific instrument interface boards. The processor board is based upon the 16 bit RTX2010RH processor operating at 6 million instructions per second (MIPS). It contains 64 Kbytes of Programmable Read Only Memory (PROM), 256 Kbytes of Electrically Erasable Programmable Read-Only Memory (EEPROM), 256 Kbytes of Static Random Access Memory (SRAM), an FPGA for processor support and interface functions and a customizable external interface connector. The MIL-STD-1553 interface board is based upon the United Technologies Microelectronics Center (UTMC) Summit DX protocol processor and data transceiver with an FPGA for interface functions and 64 Kbytes of buffer SRAM. The DC/DC converter board is based upon hybrid converter and filter modules. It incorporates in-rush current limiting circuitry, under-voltage lockout circuitry and output noise filtering. The housekeeping voltage, current and temperature monitoring board uses input multiplexers and signal conditioning operational amplifiers to feed a single analog to digital converter (ADC). An FPGA is used to control data acquisition and interfaces the system to the processor.

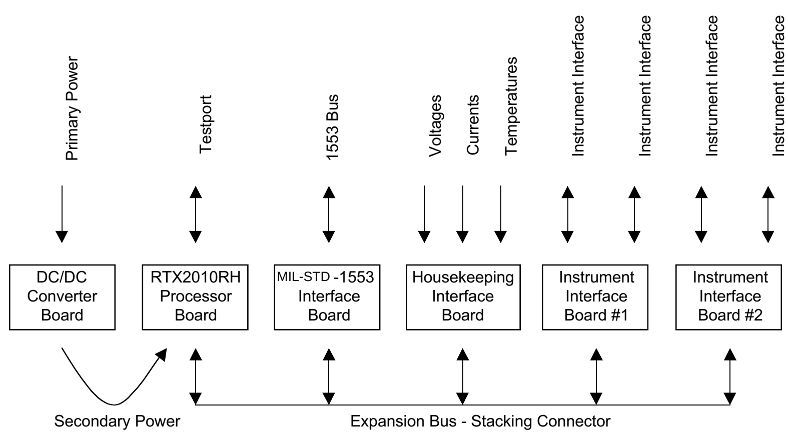


Figure C-1 DPU Block Diagram

Table C-1 DPU Power and Mass

Unit	Board Weight (grams)	Chassis Weight (grams)	Power (watts)	Thickness (inches)
Processor Board	125	25	1.00	0.50
MIL-STD-1533 Board	125	25	0.50	0.50
Housekeeping Board	125	25	0.50	0.50
DC/DC Converter Board	175	55	1.50	0.75
Instrument Interface Board	125	25	1.00	0.50
Instrument Interface Board	125	25	1.00	0.50
Chassis End Plate	0	60	0.00	0.06
Chassis End Plate	0	60	0.00	0.06
Total	800	300	5.50	3.37

Estimated Power: **5.50 watts**
Estimated Weight: **1.10 kg**
Estimated Dimensions: **4.2 x 4.2 x 3.4 inches** (not including mounting feet)

Three of the four core boards in the system are based upon existing designs. The processor board is very similar in design to the processor board used in the instrument DPUs that were flown on the NEAR, ACE and Cassini spacecraft. The updated memory components for this processor board are used in the High Energy Neutral Atom (HENA) instrument DPU currently qualified and awaiting launch. The MIL-STD-1553 interface board is similar in design to the interface board used on the NEAR spacecraft and would be software compatible with the existing unit. The DC/DC converter board uses the same components and circuit design as used in the HENA and GUVI instruments. The change in connectors and board size necessitates a new board layout. The housekeeping board would be a new design and could be tailored to the requirements of the selected instruments. Circuit design of this board would be based upon similar boards developed for NEAR, Cassini and HENA.

Area studies of the processor board, MIL-STD-1553 interface board, and the DC/DC converter board were completed to verify the feasibility of packaging these functions onto the four inch square board form-factor. A detailed printed circuit board routability study of the processor board was completed to insure that the parts packing assumptions used in the area studies were valid. Both studies indicated that the system is feasible and would not present packaging problems.

APPENDIX G

Preliminary Structural Design and Test Requirements

Table of Contents

1.0 Introduction

The purpose of this document is to define preliminary structural design and test requirements for the Solar Terrestrial Relations Observatory (STEREO) and its components. The baselined launch vehicle for the STEREO program is the Athena II, however, since Shuttle is being carried as an alternative, this document addresses those requirements as well.

2.0 References

The following documents are referred to within this document and were used as guidance in the development of design load factors and environmental test requirements.

(1) *LMLV Mission Planner's Guide*, Lockheed Martin Astronautics, Denver Colorado, Initial Release, September 1997.
(2) *Athena Environments Update to the Athena Mission Planner's Guide* (as of 9/98), Lockheed Martin Astronautics, Denver Colorado.
(3) *Flight Support System (FSS) User 's Guide for Space Shuttle*, STE-35, Baseline Issue, July 17, 1992, United Space Alliance.
(4) *Shuttle Orbiter/Cargo Standard Interfaces (CORE)* ICD-2-19001, Revision L, CPN-68, United Space Alliance, January 15, 1998.
(5) *Payload Verification Requirements, Space Shuttle Program,* NASA Space Transportation System (NSTS) 14046 Revision C, NASA, April 1994.
(6) GEVS-SE REV A, *General Environmental Verification Specification for STS & ELV Payloads, Subsystems, and Components*, NASA Goddard Space Flight Center, Greenbelt, Maryland, June 1996.
(7) *Trans-Lunar Injection Timer*, Lockheed Martin document Number P402S002, 20 May 1996.

3.0 General Approach

The design and test requirements are based on the assumption that the STEREO Spacecraft will be launched on either a Lockheed Martin Athena II or a Space Shuttle with Flight Support System (FSS) cradles. The Athena II is currently being carried as the baseline launch vehicle. The Athena II configuration consists of a Model 92 payload fairing, a Model 47 payload adapter, and Thiokol STAR-37FM motor injection system. The Space Shuttle configuration uses a Thiokol STAR 48V motor injection system. An attempt has been made to envelop the Athena and Shuttle load cases without being overly conservative. The testing philosophy is based on the proto-flight approach, where hardware is tested to design qualification levels for flight acceptance durations.

4.0 Spacecraft Requirements

This section addresses design loads and environments for the spacecraft.

4.1 Design Load Factors

The primary structure shall be designed to the following limit loads (maximum expected loads) multiplied by the appropriate factor of safety. Limit load factors for the Athena II are provided in References 1 and 2. Limit load factors for the Shuttle are provided in Reference 3.

PRELIMINARY

Preliminary Athena II Spacecraft Center of Gravity (Cg) Load Factors

Lockheed-Martin Proprietary Data

Preliminary Shuttle Spacecraft Center of Gravity (Cg) Load Factors

	Load Factors (g)		
Shuttle Flight Event	**Nx (thrust)**	**Ny (lateral)**	**Nz (lateral)**
Liftoff	+6.4	+2.0	+5.0
Landing	–3.6	+4.0	–8.4

Note:
(1) Load factors are considered to be “yield” load factors.
(2) Axial and lateral load factors should be applied simultaneously for each load case.
(3) Load factors are to be applied at the center of gravity of the payload.
(4) Payload is defined as the STEREO spacecraft with the STAR 48V motor assembly.
(5) Load factors contain a spacecraft dynamic uncertainty factor of 2.0.

4.2 Factors and Magins of Safety

The following factors of safety shall be used for design of the spacecraft primary and secondary structures:

FST = Factor of Safety for Test = 1.25
FSY= Factor of Safety for Yield Strength Design = 1.25
FSU = Factor of Safety for Ultimate Strength Design = 1.4

Margins of Safety:

$$\text{MSY} = \text{Margin of Safety on Yield Strength} = \frac{\text{Material Yield Strength}}{\text{FSY x Applied Stress}} - 1.0 \geqq 0.10$$

$$\text{MSU} = \text{Margin of Safety on Ultimate Strength} = \frac{\text{Material Ultimate Strength}}{\text{FSY x Applied Stress}} - 1.0 \geqq 0.10$$

4.3 Stiffness Requirements

4.3.1 Athena II Requirements

Lockheed-Martin Proprietary Data

4.3.2 Shuttle Requirements

To avoid dynamic coupling of spacecraft and Shuttle Flight Support Structure modes, the primary spacecraft structure should be designed to have fundamental frequencies above 10 Hz. (Reference 3).

4.4 Sine Vibration Testing

A spacecraft level sine burst or sinusoidal vibration test will be designed to cover both the structural strength test and the simulated flight environment. The final vibration levels will depend on results of the STEREO/launch vehicle dynamic coupling analysis, but preliminary levels are presented below:

Preliminary Athena II Sine Vibration Levels
Sweep rate = 4 octaves/min

Lockheed-Martin Proprietary Data

4.5 Acoustic Testing – Athena II and Shuttle

To verify the ability of the spacecraft to survive the launch acoustic environment, a spacecraft level acoustic test shall be performed in a reverberant sound pressure field. The spacecraft shall be tested to the following proto-flight (maximum expected levels +3 dB) acoustic levels:

4.6 Shock Testing

The primary source of shock for the STEREO spacecraft is the separation nut ordnance used to separate the STEREO Injection Stage from the STEREO Spacecraft. Actual ordnance shall be fired to test the separation of the spacecraft from the STEREO Injection Stage. To account for the scatter associated with the actuation of the same device, this test shall be performed twice (Reference 6). The separation system is assumed to be similar to that for the Lunar Prospector Mission. The expected shock environment is shown below (Reference 7).

PRELIMINARY

Athena I & II Internal Acoustical Levels (Reference 1)
1/3 OCTAVE SPL, dB

Lockheed-Martin Proprietary Data

Shuttle Orbiter Cargo Bay Internal Acoustic Environment (Reference 4)

1/3 Oct. Band Center Frequency (Hz)	Sound Pressure Level (dB) Ref. 2×10^{-5} N/m2 — MEI/Lift-off — 5 Seconds per Mission* Payload Diameter < 160 Inches	Sound Pressure Level (dB) Ref. 2×10^{-5} N/m2 — Max Q/Transonic — 10 Seconds per Mission* Payload Diameter ≤ 180 Inches	Sound Power Level (dB) Ref. 10^{-12} Watts — Payload Bay Vents ** — Lift-off 5 Seconds per Mission	Sound Power Level (dB) Ref. 10^{-12} Watts — Payload Bay Vents ** — Max Q/Transonic 35 Seconds per Mission
31.5	122.0	112.0	119.0	110.0
40	124.0	114.0	121.0	114.0
50	125.5	116.0	122.0	114.0
63	127.0	118.0	126.0	115.0
80	128.0	120.0	128.0	118.0
100	128.5	121.0	130.0	120.0
125	129.0	122.5	126.0	125.0
160	129.0	123.5	130.0	130.0
200	128.5	124.5	129.0	125.0
250	127.0	125.0	132.0	121.0
315	126.0	125.0	130.0	124.0
400	125.0	124.0	127.0	118.0
500	123.0	121.5	130.0	119.0
630	121.5	119.5	122.0	117.0
800	120.0	117.5	123.0	115.0
1000	117.5	116.0	122.0	114.0
1250	116.0	114.0	121.0	115.0
1600	114.0	112.5	118.0	109.0
2000	112.0	110.5	121.0	108.0
2500	110.0	108.5	123.0	110.0
OASPL	138.0	133.5	140.0	134.0

* Time per mission does not include a scatter factor

** The payload bay vents act as individual noise sources for the payload bay
The noise radiated from any one vent is described below

The pyrotechnic devices used to release booms, solar arrays, protective covers, etc. produce a local source of shock. Shock testing shall be conducted at the spacecraft level by firing the ordnance and allowing the release of the boom, solar array, cover, etc. These tests shall be performed twice.

4.7 Mass Properties

Spacecraft mass properties shall be determined by analysis and measured to the following accuracy:

Lunar Prospector Separation Shock Environment

Frequency (Hz)	Shock Level (g's)
100	35.2
315	59.3
1250	1125
2500	3741
3150	4942
8000	5983
10000	5420

STEREO Spacecraft Mass Property and Spin Balance Accuracy

	Launch Configuration	Orbital Configuration
Weight	±TBD%	±TBD%
Center of Gravity	±TBD% inches w/respect to separation plane at geometric center.	Derived analytically from launch configuration.
Moment of Inertia	Measured in three mutually perpendicular axes within ±TBD%. have a goal of TBD times lateral (y, z) moment of inertia.	Derived analytically from launch configuration. Thrust (x) moment of inertia shall
Spin Balance	Angle between spacecraft geometric thrust (x) axis and spacecraft principal thrust axis must not exceed TBD.	Derived analytically from launch configuration.

5.0 Component, Subsystem and Instruments Requirements

5.1 Design Load Factors

The following design load factors, multiplied by the appropriate factor of safety, shall be applied to components, subsystems, and instruments. Thrust and lateral load factors are to be applied simultaneously for each load case.

5.2 Factors and Margins of Safety

The following factors and margins of safety will be used for the development of component, subsystem and instrument structures:

Instrument and Component Design Load Factors for Athena II

Load Case	Thrust (x) Load Factor (g's)	Lateral (y, z) Load Factor (g's)
1 (Due to 2nd Stage Castor Ignition)	±2 0	±2.5
2 (Due to 2nd Stage Motor Resonance and Gust)	±14.5	±4

Instrument and Component Design Load Factors for Shuttle

Load Case	Thrust (x) Load Factor (g's)	Lateral (y) Load Factor (g's)	Lateral (z) Load Factor (g's)
1 (Due to lift-of f)	±9.5	±3.0	±7.5
2 (Due to landing)	±5.4	±6.0	±12.6

Factor s of Safety:

FST = F actor of Saf ety for Test = 1.25
FSY= Factor of Saf ety for Yield Str ength Design = 1.25
FSU = Factor of Saf ety for Ultima te Str ength Design = 1.4

Mar gins of Saf ety:

$$MSY = \text{Margin of Safety on Yield Strength} = \frac{\text{Material Yield Strength}}{\text{FSY x Applied Stress}} - 1.0 \geqq 0.10$$

$$MSU = \text{Margin of Safety on Ultimate Strength} = \frac{\text{Material Ultimate Strength}}{\text{FSY x Applied Stress}} - 1.0 \geqq 0.10$$

5.3 Stiffness Requirements

Components, subsystems, and instr uments shall be designed to ha ve pr imar y str uctur al vibr ation modes a bove 100 Hz in the thr ust axis and 50 Hz in the la ter al ax es.

5.4 Pressure Requirements

Components, subsystems, and instruments shall be designed to withstand a maximum pressure rate chang e of TBD psi/sec. Pr essur e pr ofile testing is consider ed optional.

5.5 Shock Requirements

Separation shock must be considered in the design of components, subsystems, and instruments located near the spacecr aft/STEREO injection system se par ation plane . Expected shoc k levels shall be addressed on a case by case basis.

Self-induced shock shall be considered in the design of components, subsystems, or instruments and shall be tested a t the component le vel. Self-induced shoc k can r esult fr om the actua tion of pyrotechnic or pneumatic devices to release booms, solar arrays, protective covers, etc. "End of travel" impact of deployable devices can also produce a self-induced shock. Self-induced shock shall be tested by actuation of the device, allowing release of the boom, cover, etc. This test shall be performed twice.

5.6 Sine Vibration Testing

Components, subsystems, and instruments shall be subjected to sinusoidal vibration along each of the three orthogonal axes. Proto-flight levels are shown in the following tables.

Preliminary Athena II Component Sine Vibration Levels Sweep rate = 4 octaves/min

Axial		Lateral	
Frequency (Hz)	Acceleration (g's, zero to peak)	Frequency (Hz)	Acceleration (g's, zero to peak)
5 to 30	0.4	5 to 30	0.3
30 to 35	20.0	30 to 35	12.0
35 to 70	0.6	35 to 60	0.3
70 to 100	0.45	60 to 75	0.7
		75 to 100	0.85

Preliminary Shuttle Component Sine Vibration Levels Sweep rate = 4 octaves/min

Axial		Lateral	
Frequency (Hz)	Acceleration (g's, zero to peak)	Frequency (Hz)	Acceleration (g's, zero to peak)
5 to 30	0.5	5 to 30	0.5
30 to 35	23.0	30 to 35	19.0
35 to 100	0.5	35 to 100	0.5

5.7 Random Vibration Testing

Components, subsystems, and instruments shall be subjected to random vibration along each of the three orthogonal axes, one of which is parallel to the thrust axis. The proto-flight levels, obtained from Reference 6, are as follows:

Preliminary Component Random Vibration Levels
Duration = 60 seconds

Frequency (Hz)	PSD Level
20	0.026 g^2/Hz
20–50	+6 dB/Oct
50–800	0.16 g^2/Hz
800–2000	−6 DB/Oct
2000	0.026 g^2/Hz
Overall	**14.1 G rms**

5.8 Mass Properties

Each component, subsystem, and instrument shall be weighed to an accuracy of 1% or 1 pound, whichever is less. The accuracy of the center of gravity and moment of inertia calculations shall have an accuracy goal of 10%.

APPENDIX H

Concept of Operations (CONOPs)

DRAFT

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Acknowledgement

Many of the novel operational concepts contained within this document were developed and refined for the NASA TIMED Mission. The STEREO Mission Operations Team would to thank the TIMED Mission Operations staff for their assistance in the development of the STEREO Concept of Operations.

1.0 General

1.1 Purpose

This purpose of this Concept of Operations (CONOPS) document is to describe, from a user's operational perspective, how the Mission Operations Team will conduct spacecraft on-orbit operations in response to the STEREO mission requirements.

1.2 Scope

This document focuses on those operations related to the spacecraft bus. Instrument (the spacecraft payload) operations are not specifically addressed due to the decoupled instrument approach. The Mission Operations Center (MOC) is emphasized while the other components of the ground system, like the Deep Space Network (DSN) and Science Operations Center (SOC) are only covered as far as their interface to the MOC is concerned.

1.3 Reference Documents

1. STEREO Requirements, A. Santo, October 8, 1998
2. MOC to Science Operations Center ICD, TBD
3. Program Service Level Agreement, January 29, 1999
4. Instrument ICDs, TBD
5. MOC Configuration Management Plan, TBD
6. Operational Constraints document, TBD
7. Operations Handbook, TBD
8. Contingency Plans, TBD
9. Early On-orbit Operations Plan, TBD

2.0 Operational Description

2.1 Mission

As part of NASA's Sun-Earth Connections program, the STEREO mission will provide a new perspective on solar eruptions and their consequences for Earth. To provide the images for a stereo reconstruction of solar eruptions, one spacecraft will lead the Earth in its orbit and one will lag. Each will carry a suite of instruments. When simultaneous telescopic images are combined with data from observatories on the ground or in low Earth orbit, the buildup of magnetic energy, lift off, and trajectory of Earthward-bound Coronal Mass Ejections (CMEs) can all be tracked in three dimensions. When a CME reaches Earth's orbit, magnetometers and plasma sensors on the STEREO spacecraft will sample the material and allow investigators to link the plasmas and magnetic fields unambiguously to their origins on the Sun.

The STEREO mission consists of two identical spacecraft in heliocentric elliptical orbits that are in the ecliptic plane at approximately 1 AU, one ahead of the Earth and the other behind it. The angular separation of the two spacecraft will be gradually increasing with a drift rate of 20°/year for the leading spacecraft and -28°/year for the lagging spacecraft.

The STEREO mission has a goal of two years with a possible extension of three years. There are four mission phases for the five years, which are determined by the angle between the two spacecraft (S/C) (a). Each angular separation phase has specific science objectives as listed below:

Phase	S/C Angular Separation	Science Objective
Launch to 400 days	$\alpha \leq 50°$	3D corona effects
400 to 800 days	$50° \leq \alpha \leq 100°$	CME physics
800 to 1100 days	$100° \leq \alpha \leq 200°$	Earth directed CMEs
After 1100 days	$\alpha > 180°$	Global solar evolution and space weather

2.2 Spacecraft Description

Each of the two STEREO spacecraft will be identical[1] with no redundancy. The spacecraft bus will be built by JHU/APL with NASA Goddard Space Flight Center (GSFC) procuring the instruments. The entire S/C will be integrated at JHU/APL.

The spacecraft bus consists of six operational subsystems supporting a payload suite of six instruments (Figure 2-1). The spacecraft bus is designed around an Integrated Electronics Module (IEM). The IEM is a single box that contains the Command & Data Handling (C&DH) and RF Communications subsystems on plug-in cards. The cards within the IEM communicate over a PCI parallel data bus. A MIL-STD-1553 bus is used for transferring command and telemetry data between the IEM, the instruments, the Guidance and Control (G&C) subsystem, and the Power subsystem. An RS-422 high-speed data bus is used for the science data interface between the IEM and the Solar Corona Imaging Package (SCIP) instrument.

The C&DH subsystem provides real-time, time tagged, macro, and autonomy command capabilities. It uses a Mongoose-V, 12 MHz, 32-bit processor that formats all telemetry into CCSDS compliant packets. A 7.5 Gbit RAM Solid State Recorder (SSR) is used for data storage of all science and engineering data. An Ultra Stable Oscillator (OSC) will be used for time reference. Time between the two spacecraft will be synchronized to within 0.1 seconds.

The RF Communications subsystem[2] will provide simultaneous X-Band (XB) uplink, downlink, and navigation data using one High Gain Antenna (HGA), two Medium Gain Antennas (MGA), and two Low Gain Antennas (LGA). The LGAs will provide communications from launch to 0.01 AU. The two MGAs are fanbeam antennas. The wide angle MGA will provide communications from 0.01 AU to 0.23 AU while the narrow angle MGA will be used for emergency communications when the S/C is in Safe Hold or Earth Acquisition modes. The HGA consists of a gimbaled, 1.1 meter, parabolic dish with a 115° gimbal travel. It will be used when the spacecraft range is greater than 0.23 AU. The HGA is steered autonomously.

There are two XB uplink rates, 125 bps for normal operations and 7 bps for emergency operations. Navigation data will be generated by an APL developed transceiver modified to obtain corrected two-way Doppler from uplink and downlink frequencies. The RF Communications subsystem is designed to use the DSN 34-meter Beam Wave Guide (BWG) antennas.

The G&C subsystem provides three-axis attitude control of the S/C and also controls the pointing of the HGA. Nominal orientation of the S/C will have the X-axis of the S/C pointed at the Sun

[1] The Energetic Particle Detector (EPD) alignment differs between leading and lagging spacecraft, hence the two S/C will not be truly identicl. Additionally, the SCIP occulation disks will be different between spacecraft.
[2] In order to accommodate changing requirements, the RF design for STEREO will most likely be changed during Phase A.

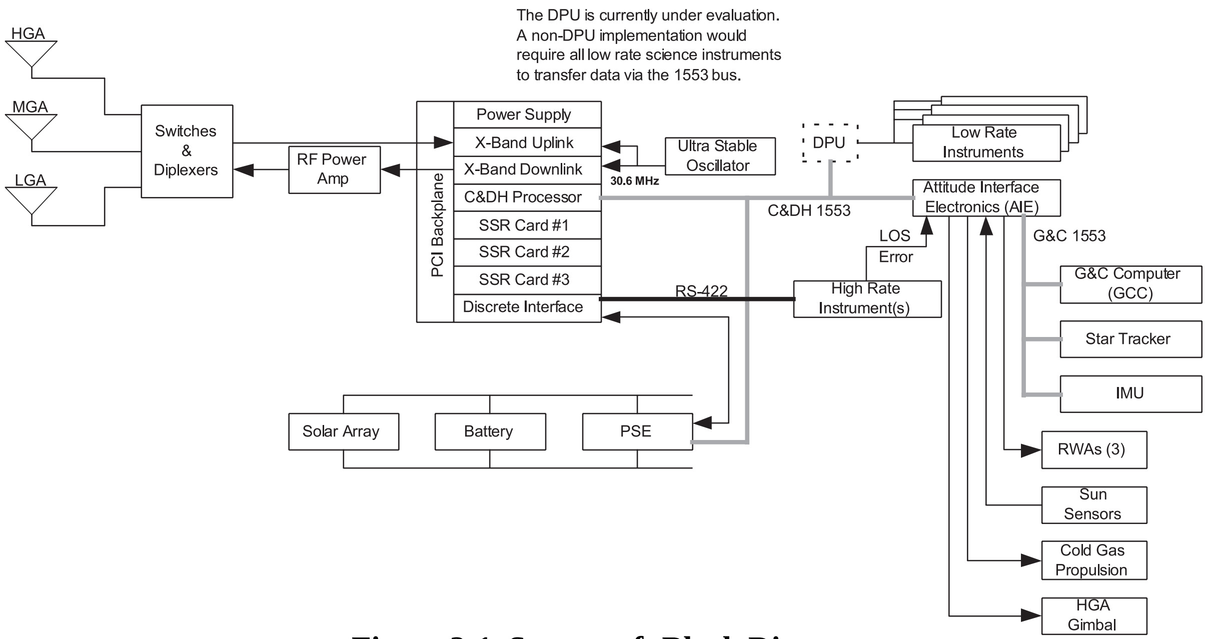

Figure 2-1 Spacecraft Block Diagram

within 0.1° (3 σ) and the HGA pointed at the Earth, also within 0.1° (3 σ). The G&C subsystem consists of the two processors, the Attitude Interface Electronics (AIE) and Attitude Flight Computer (AFC), three attitude sensors; an Inertial Measurement Unit (IMU), Star Tracker, and Digital Solar Aspect Detectors (DSAD), and two control components; Reaction Wheel Assemblies (RWA) and the Propulsion subsystem.

The AIE, using an RTX2010RH, 12 MHz processor, provides the interface between all attitude components and the C&DH subsystem. It also autonomously provides S/C attitude safing operations. A separate G&C MIL-STD-1553 bus provides communications between the AIE and the AFC, Star Tracker, and IMU. The AFC, using the same processor as the C&DH subsystem, implements the attitude control algorithm, thruster control, and HGA gimbal pointing.

The IMU provides S/C rate data using four hemispherical resonator gyro units. The Star Tracker autonomously identifies stars with brightness less than 7.5 Mv. There are five DSADs each with a ±640 field of view (FOV) to determine the Sun's location with an accuracy of 0.5°.

Three RWA provide pointing control. As S/C momentum builds in the RWAs it will be dumped autonomously by the G&C computer. This occurs, nominally, on every four day intervals, using thrusters in the Propulsion subsystem.

The Propulsion subsystem consists of a cold gas tank, two transducers, high-pressure latch valve, regulator, and four thrusters. The cold gas tank will contain approximately 1.7 kg (five liters) of GN2 at 5000 psia. This will be sufficient propellant to dump momentum for five years with a 10 percent leakage allowance.

The Power subsystem employs two fixed GaAs solar arrays (SA). Power is managed by a Peak Power Tracker (PPT) that will provide an unregulated 28V ±6V DC bus voltage. A 21 ampere-hour Super NiCd battery provides power from launch to SA deployment and for Low Voltage Sense (LVS) conditions.

The Thermal subsystem is a passive design using blankets, radiators, and thermostatically controlled heaters. All instruments will be thermally isolated from the S/C structure.

2.3 Spacecraft Operations

The spacecraft will be operated to collect data as the on board instrument suite observes the solar regions of scientific interest. The spacecraft bus is operated to support this data collection by providing nominal attitude, power, navigation data, thermal control, data storage, and rule-based autonomy. It is expected that the spacecraft will operate nearly autonomously, requiring only minimal ground support to uplink occasional command messages and to recover science and engineering data on a daily basis. Nominally, one contact, or track, with each spacecraft will be scheduled each day, for a duration of two or more hours. During the remainder of the time, the spacecraft will be on its own collecting data, measuring its own health and responding to any self-discovered anomalous operations. It will do this by carrying out a continuous performance assessment function that consists of observing its own telemetry and evaluating pre-stored autonomy rules related to performance and operation. The goal of the Mission Operations Team (MOT) will be to maintain a science data gathering capability though it is likely that some on board anomalies cannot be autonomously handled in a manner so as to preserve normal operations. In these cases the spacecraft may transition to a safe-hold or Earth acquisition mode (see Section 5.4) where science data collection is suspended and all non-essential instruments and subsystems are powered down.

The C&DH processor will receive ground-based command messages which provide initialization data, control instrument configurations, specify attitude configuration, allocate data storage reserves, and, in general, to ‘orchestrate’ the operation of the entire spacecraft. Command messages not specifically addressed to the C&DH processor will be conveyed to the addressed spacecraft bus subsystem and instrument destinations via a data bus. The C&DH subsystem will report in telemetry, the status of both the receipt (from the ground) and delivery (to the on board instrument or subsystem) of these commands.

State-of-health (engineering housekeeping) data throughout the spacecraft bus is sampled by the C&DH processor and is stored on the SSR. Science and engineering data produced by the instrument suite are also transferred to the SSR for storage. The SSR will hold about 7.5 Gbits of data which will be played back once per day. Real-time spacecraft engineering data will be interleaved (3%) with SSR playback data during a track. The amount of SSR data transmitted during a track will vary over the duration of the mission. Initially, the entire SSR will be played back each day. However, as the S/C to Earth range increases, the amount of SSR data played back will decrease. Eventually, during an eight-hour track, the entire SSR cannot be played back.

Spacecraft attitude will, nominally, be maintained with the X-axis at the Sun, with an accuracy of 0.1∞ (3 σ), and with the gimbaled HGA pointing at the Earth also with an accuracy of 0.1∞ (3 σ) continuously. This attitude will be controlled autonomously by the G&C subsystem. Attitude will be measured by star cameras, gyros, and DSADs, processed by the AFC and adjusted by controlling spacecraft momentum using the RWAs. Momentum build-up in the wheels will periodically be dumped using the cold gas thrusters. Attitude data will be generated and provided to the C&DH processor for broadcast to on board subsystems and instruments. The G&C subsystem produces, along with nominal engineering telemetry data, diagnostic data to be used for ground-based performance assessment. The contents of these buffers are transmitted to the MOC to support G&C subsystem performance assessment and possible anomaly investigations. If the spacecraft cannot maintain the preferred operational attitude, the G&C subsystem, via the AIE, will configure and maintain a safe-hold attitude with solar panels directed toward the Sun and the HGA pointed at the Earth with all non-essential loads powered down. Commands issued by the MOT will be required to transition out of this safe-hold mode.

Spacecraft power is provided by solar panels augmented by a storage battery. Typically, the battery will only be used during launch, anomalous operations and propulsion events. The solar panels are fixed along the Z-axis (Sun facing). A Peak Power Tracking feature controlled by a software algorithm resident in the C&DH processor maintains maximum power subsystem efficiency.

The C&DH subsystem continuously monitors telemetry parameters for violations of established operating rules. These rules are defined and uploaded by the MOT. Rule violations invoke command sequences that tend to overcome the cause of the violating condition. These autonomous operations attempt to maintain an operational spacecraft, but there maybe anomaly situations that the S/C autonomy rules do not cover. In these instances, the spacecraft will transition to safe-hold mode or earth-acquisition mode and await ground command response.

As part of the normally generated engineering telemetry, the spacecraft preserves a record of all commands executed on board. Included are real-time, time tagged, macro, and autonomous command execution. Such data is necessary to assess operational performance of the spacecraft bus.

2.3.1 Command Uplink

The spacecraft C&DH subsystem receives command uplinks from the ground system that may either be addressed to the Uplink Critical Command Decoder or the C&DH processor. Command packets are formatted into command transfer frames by the MOC. A packet, of variable length, may be embedded within a single transfer frame or may span over several transfer frames. The packet is addressed to a particular spacecraft subsystem or instrument and all packets contained within a transfer frame must be addressed to the same subsystem or instrument. The C&DH processor ingests and assembles the complete packet, then routes it, via the PCI or MIL-STD-1553 bus (depending on the addressed subsystem or instrument), to its destination. If the received packet is incomplete or otherwise unacceptable, based on error detection criteria, the packet is rejected in its entirety and a status message is transmitted to the ground system. However, the C&DH processor does not check the contents of the command packet. Once the designated subsystem or instrument has received a packet, it is the responsibility of that subsystem or instrument to evaluate the packet content for acceptability. The report of this evaluation must be conveyed via the engineering telemetry produced by that subsystem or instrument.

Besides the direct delivery method of commanding described above, the Uplink Critical Command Decoder may command certain spacecraft configuration states. This unit, which ‘parallels’ the C&DH processor, may issue only configuration commands which control the power switching relay states. These critical (relay) commands are formatted in packets, one command per packet, by the MOC, and then uplinked, in real-time to the spacecraft Uplink Critical Command Decoder. These commands are executed as they are received in real-time since there is no on board storage of these commands.

The Uplink Critical Command Decoder also handles all relay (power switching) commands and provides an emergency capability to configure the spacecraft (i.e., to power on and off instruments and subsystems). The normal method of commanding is through the C&DH processor. However, the C&DH processor may not always be powered on, hence the need for an alternate emergency command capability.

The S/C can handle two uplinks rates, 125 and 7 bps. Normally, 125 bps will be used. The lower 7 bps rate is for communications during anomalous operations.

2.3.2 Telemetry Downlink

There are three downlink rates available on the S/C; high-rate science, low-rate science, and low-rate engineering. Normally, the high-rate science downlink rate will be used. This rate employs the HGA with Reed-Solomon (RS) +6:1 convolutional coding. Real-time science and engineering telemetry has priority. That is, if real-time science and engineering data are selected for downlink, then any data generated in real-time by either the spacecraft bus or instruments is formatted into transfer frames and downlinked immediately. Typically this rate will be used to play back the SSR with three percent real-time data interleaved.

The low-rate science downlink rate uses RS +2:1 convolutional coding. This rate will only be used if the scheduled DSN station cannot support the RS +6:1 convolutional coding. Real-time science and engineering data as well as SSR playback data can be transmitted at this rate. Space weather data (broadcast mode) will also be transmitted at this rate at all times when the spacecraft is not communicating with DSN.

The low-rate engineering downlink rate does not use any encoding. It will only be for anomalous operations.

2.3.3. Early Operations

Early spacecraft operations is treated as a separately because of its criticality and singular use. The Early Operations phase extends from launch vehicle separation to the declaration, by the Mission Planning Team, that the spacecraft bus and all instruments are capable of supporting the mission objectives. This phase encompasses launch load preparation through on-orbit checkout and evaluation. The launch load consists of a command sequence that will control critical spacecraft bus operations immediately after separation from the boost motor. These commands will provide:

- Deployment of solar panels
- Powering on the IEM and other bus components
- Initiation of safe-hold mode attitude capture

The S/C separation sequence triggers the pre-stored launch load command sequence in the C&DH subsystem. This sequence will sustain spacecraft operations until the first scheduled DSN track occurs. The MOT, upon evaluation of the performance of the spacecraft bus at that time, may choose to alter the planned operational sequence or may elect to continue with the stored launch load sequence until the next scheduled track.

The primary activity during this Early Operations phase will be to checkout the subsystems and instruments on the S/C. The STEREO MOT will depart from the normal operations staffing plans to provide a more or less continuous 24-hour/day support. For the first few days, more than one DSN track per day will be requested. The Early Operations phase will be supported by combined MOT, SBET, and instrument teams located at the MOC and the SOC. Launch operations teams may also be utilized as required.

2.4 Instrument Operations

During Normal Operations, all instruments will be powered and time tagged command capability will be enabled. For this mission, all instrument operations will be decoupled from the spacecraft bus operations (per Reference (1)). The goal is to support a concept of decoupled operations, which reduces system complexity and cost. With only a few exceptions, the spacecraft bus and the instruments will be operated independently of each other (Section 2.5). The same can be said about the ground elements (the SOC and the MOC). This decoupling of instrument operations concept greatly simplifies the operations process, which traditionally requires these functions to be merged in a complex manner.

The SOC is responsible for the following operational tasks for all instruments:

- Planning, scheduling, and generating instrument commands
- Instrument health
- Calibration
- Synchronization of instrument operations between S/C

Instrument command loads will be assembled as packets and transferred to the MOC prior to a scheduled contact, or track. Separate command messages are required for each S/C. Included, as part of the command packet transfer to the MOC, will be certain identifying data to be used by the MOC to verify that an authorized source has generated and transferred the data. The SOC will attach data that specify both the earliest and latest times that the attached command packet may be uplinked to the instrument. The MOC will be responsible for the delivery of the content of the packet(s) to the addressed instrument but assumes no responsibility regarding the actual commands. On the S/C, a shared MIL-STD-1553 data bus provides all data interfaces between the spacecraft and the instruments . The spacecraft acts as the bus controller, and each instrument is a remote terminal. The C&DH subsystem will report in telemetry, the status of both the receipt (from the ground) and delivery (to the on board instrument or subsystem) of these commands. The SOC will be responsible for the verification and validation of instrument response based on the uplinked command load.

Due to the relatively low command uplink rate, instrument teams should design their instrument to require a small number of commands per day. Nominally, an hour of instrument command uplink time for each track has been set. This translates into approximately 450 kbits of instrument command data for all instruments per track.

The spacecraft will support storage of command packets for distribution to the instruments at a later time. The aggregate size of memory available to all instruments for stored commands is enough to hold approximately 400 commands. Stored command packets may be individually time tagged with one second precision, or may be part of a macro sequence.

An unpacketized broadcast message will be distributed to all instruments once per second.

This message will contain:

- Time
- Warning flags:
  - Sun keep-in violation
  - Thruster firing

  - Instrument power off
  - Indication that the next housekeeping data set will be downlinked or recorded
- Spacecraft housekeeping data required for instruments

The spacecraft will poll the instruments for data via the MIL-STD-1553 bus (and science packets from the dedicated RS-422 high speed link for the SCIP only) according to a fixed schedule. It will either downlink the data in real-time or record it on the SSR.

The spacecraft will not process instrument data before recording or downlink; any processing or data compression is the responsibility of the instruments. The instruments will have no direct access to the SSR and will not be able to retrieve data stored on it.

The instruments will generate each science data packet according to the full CCSDS telemetry packet format, including primary and secondary headers, checksums, etc. The maximum aggregate data collection rate for science packets from all instruments will be 408 kbps.

Housekeeping data from each instrument will be collected every second. The spacecraft will perform very rudimentary monitoring of this data strictly for fault protection. For example, one bit in the packet will be designated as a request by the instrument for the spacecraft to turn off its power. Other than this monitoring, the instruments cannot depend on the spacecraft to perform any processing of their housekeeping data. Each instrument will include housekeeping data in its science data packets if needed for science evaluation.

A small amount of unpacketized "space weather" data from each instrument will be collected every second.

2.5 Data Flow

Figure 2-2 illustrates conceptual flow of command and telemetry data between the ground-based spacecraft bus and instrument operations elements and the on-orbit STEREO spacecraft. The 'outer-loop' depicts instrument operations. Using a decoupled instrument operations approach, all instruments will be operated by the instrument operations team at the Science Operations Center (SOC). In Figure 2-2, begin at the SOC Planning on the far right, where instrument commands are produced. These command messages, which will be packetized along with some additional information needed by the MOC, are transmitted to the MOC via the Internet. At the MOC (MOC Authorize and Route) there is some checking performed, then these commands are queued for eventual uplink to the instrument. Along with the command packets, the SOC appends timing information which indicates the time span (earliest and latest times) over which the command packet may be uplinked to the instrument. Real-time command packets, when uplinked to the spacecraft, are immediately routed by the spacecraft bus C&DH processor (the C&DH Routing Service) to the appropriate instrument and time tagged and macro command packets are stored in the instrument's allocated storage locations in the C&DH processor. Conceptually, the command packet goes 'directly' from the SOC to the instrument, since the MOC, ground station and spacecraft bus are merely the delivery system. This delivery system notifies the SOC of the delivery status of the command message.

Whereas the SOC produce instrument commands, the MOC produces spacecraft bus commands. This is depicted in the 'inner-loop' on the Data Flow diagram (Figure 2-2). Starting at the MOC

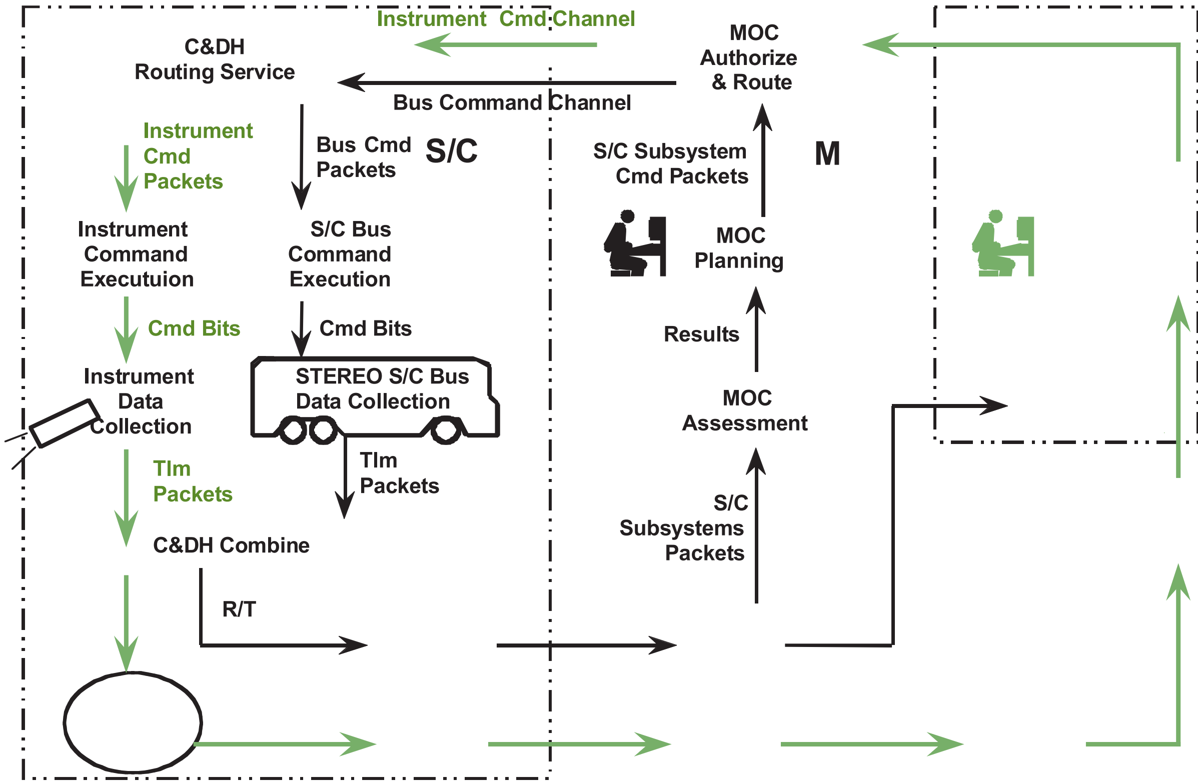

Figure 2-2 STEREO Mission Data Flow

Planning process, the Mission Operations Team (MOT) prepares command messages to the spacecraft bus to operate it during the next day. These command messages are queued for uplink (MOC Authorize and Route) just like the instrument commands, only they go to a different destination (via the C&DH Routing Service). The C&DH processor immediately routs real-time commands, to the appropriate spacecraft subsystem and time tagged and macro commands are stored in the C&DH processor. The MOC receives delivery status of the command packets just as the SOC does.

Once these command messages have been executed (Instrument Command Execution and S/C Bus Command Execution) on the spacecraft, they integrate its operation. The instruments produce science and engineering data (Instrument Data Collection) in response to the uplinked command messages. The data produced by the instruments is sent to the spacecraft data system in the form of CCSDS telemetry packets. Similarly, engineering data produced by the spacecraft bus is also formatted into CCSDS packets. The packets, produced by the instruments and the spacecraft bus and conveyed to the spacecraft data system (C&DH Combine), are stored on the SSR within the spacecraft data system (C&DH Recording). During a track with the S/C, the contents of the SSR are transmitted to the MOC (C&DH Frame Packaging).

On the ground (Ground System Telemetry Routing), real-time data is forwarded to the MOC and to the STEREO Data Server (SDS), while all recorded data is sent to the server facility (SDS Clean and Merge). All instrument data will be sent to the SOC for processing and analysis. The cycle repeats, with the SOC preparing instrument commands for still another day in space. Spacecraft bus data is routed to the MOC (MOC Assessment) where an assessment function is performed. The MOC spacecraft bus planning process then repeats.

Of significance is that the instruments and spacecraft bus are operated (almost) entirely independent of each other. The same can be said about the ground elements (the SOC and the MOC). This decoupling of instrument operations concept greatly simplifies the operations process, which traditionally requires these functions to be merged in a complicated manner.

2.5.1 Command Data Flow

The MOC produces spacecraft bus commands, processes SOC-generated instrument commands and provides a command gateway for all commands to the spacecraft. All instrument command messages from the SOC, along with some additional identifying and timing data are sent to the MOC (Figure 2-3). At the MOC, the FTP Server and Authenticator validates the proper sender [the SOC provides an encrypted source identifier and an application command packet (destination instrument) identifier]. An authentication receipt is returned to the SOC.

Once a command has authenticated, it enters a Staging Queue. There is one such queue for each instrument on each S/C. Appended to the command message (the command packet and header) is a start and expiration time specification. These represent the earliest and latest times that the command

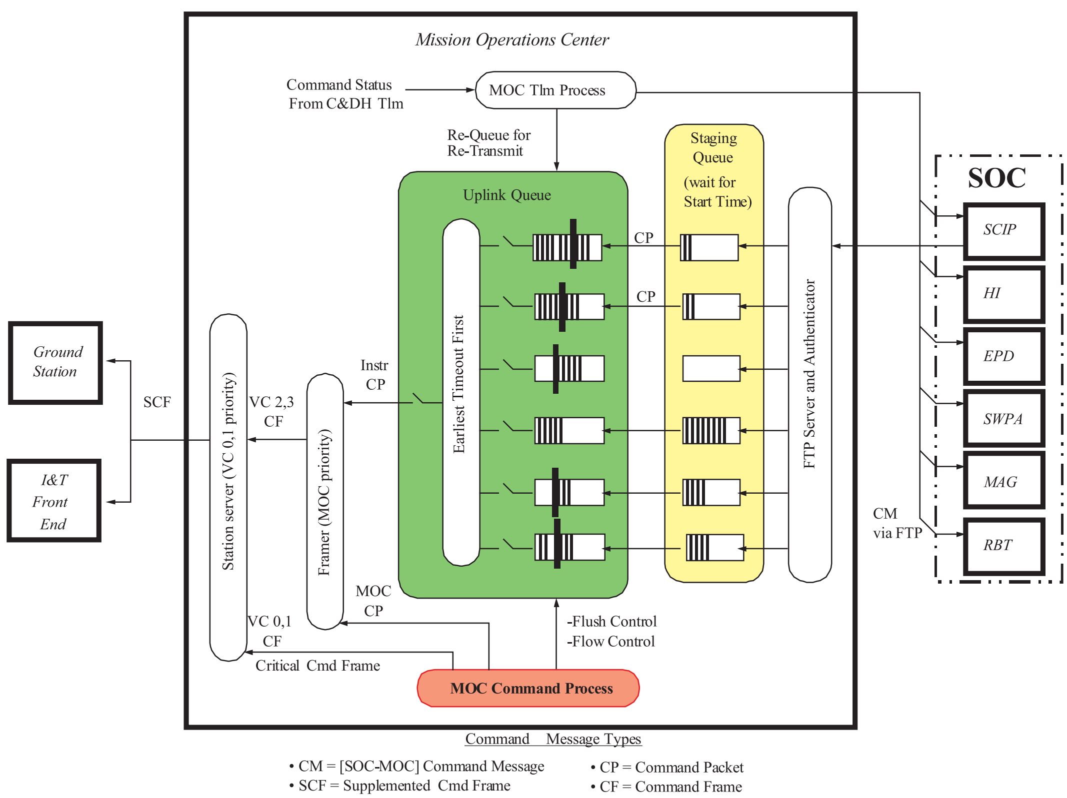

Figure 2-3
Command Data Flow

packet may be uploaded to the instrument as specified by the SOC. Once past the start time, the command packet is transferred to the Uplink Queue (again there is such a queue for each instrument on each S/C). Command packets in the Uplink Queue are ordered by expiration date, with those packets marked by the earliest expiration date ordered so as to be uplinked first. The MOT may examine the contents of the Uplink Queue to determine the number and size of the packets stored there. All stored commands will be uplinked beginning at the next ground station track, as long as time permits (the track is of sufficient duration) unless the MOT places a ‘grocery bar’ separating command packets within the queue. All commands to the left of the bar are uplinked, those to the right of the bar are prevented from being uplinked. This mechanism affords the MOT some degree of control of the uplink command packet traffic to the spacecraft.

The switch at the output (left side) of the Uplink Queue will either enable, when closed, or disable, when opened, instrument packet command flow to the spacecraft. The queues can be flushed by MOT control. If the SOC desires to replace the content of an instrument uplink queue, either entirely or in part, the entire queue is flushed and must be reloaded in its entirety. The MOC will issue notification (receipt) to the SOC of either an uplinked packet or a flushed queue.

Commands prepared for spacecraft bus operation are merged with instrument commands. Normal C&DH command packets are merged at the Framer. Here, spacecraft bus commands have priority, i.e., spacecraft bus commands are uploaded first. Commands destined to the spacecraft bus Uplink Critical Command Decoder (Virtual Channel 0 or 1) are merged at the Station Server with Uplink Critical Command Decoder command packets assigned a higher priority than spacecraft bus C&DH processor or instrument command packets (Virtual Channel 2 or 3).

The status of command packet delivery to the C&DH processor is provided via C&DH telemetry. Additional status of command packet delivery to an instrument or subsystem is also provided via C&DH telemetry. This status is forwarded to the SOC for instrument command packet delivery as a return receipt. Thus the SOC are informed of the delivery, but not the verification of actual command content. This must be provided by instrument telemetry, which can be processed only by the SOC.

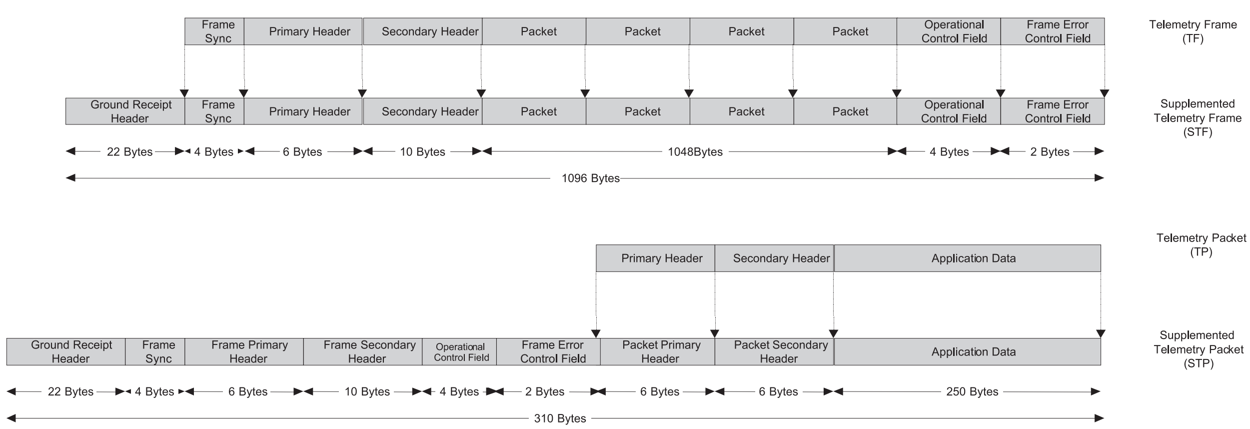

Figure 2-4 Telemetry Data Formats

The Integration and Test (I&T) Front-End, which is only utilized during spacecraft ground-based testing, is shown in the Figure 2-3. This may be utilized to provide MOC-generated command data to the spacecraft simulator during the on-orbit phase.

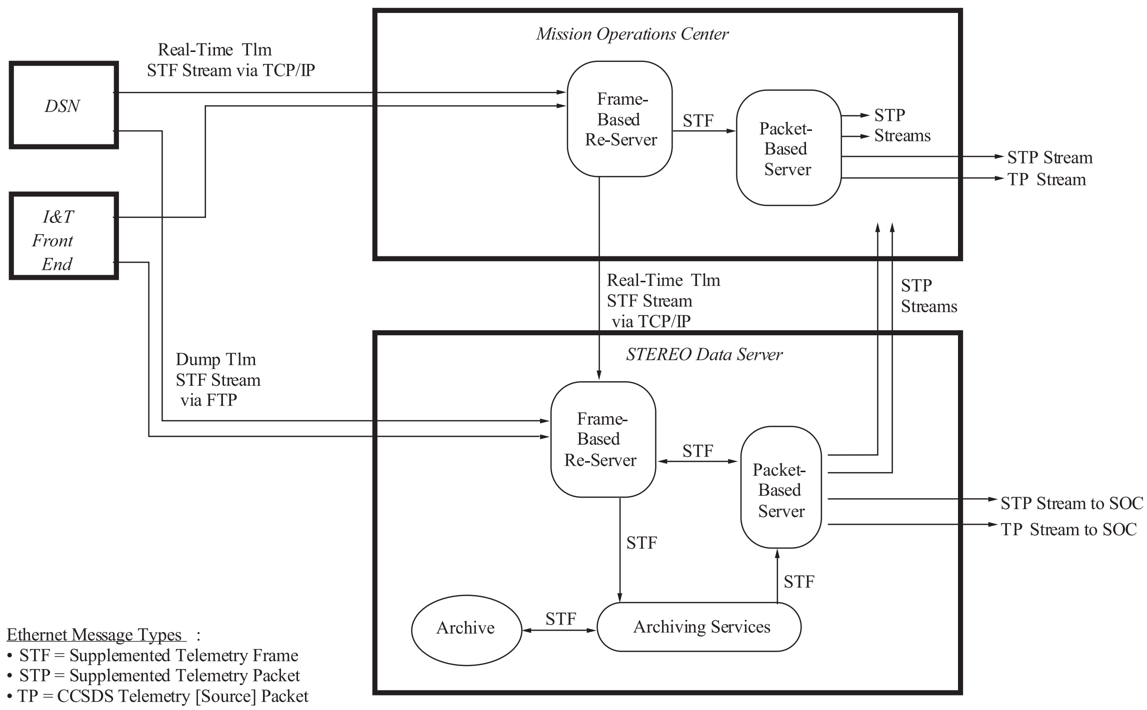

Figure 2-5 Telemetry Data Flow

2.5.2 Telemetry Data Flow

The C&DH processor generates CCSDS formatted Telemetry Frames (TF) (Figure 2-4). A ground receipt header, containing the ground receipt time, RS encoding type, and other necessary information, is added to each TF by DSN and modified by the MOC. A Supplemented Telemetry Frame (STF) is created by the addition of the ground receipt header to the TF. The STF are the form of data that is flowing in the MOC and is stored in the SDS.

Each S/C bus and instrument generates CCSDS formatted Telemetry Packets (TP). With the addition of the ground receipt header and TF header data to the TP forms the Supplemented Telemetry Packet (STP) (see Figure 2-4).

Figure 2-5 illustrates the flow of real-time and SSR playback telemetry data between DSN and MOC and SDS. Real-time telemetry data is flowed from DSN to the MOC as STF. Playback telemetry data is flowed from DSN to the SDS, again as STF. In both real-time and playback data flows, the received packets are re-served to a packet-based server where telemetry packets are extracted and output as STP and as CCSDS telemetry source (as the data was generated by the spacecraft) packets (TP). In both cases, packet streams, a flow of packets placed end-to-end in time order as they were received, are produced. The MOC provides the real-time telemetry it receives to the SDS for storage. Telemetry packets stored by the SDS, including the spacecraft playback bus engineering telemetry, is provided to the MOC. For the SOC, real-time telemetry are 'streamed' while playback data are file transferred.

The I&T front-end, which is only utilized during spacecraft ground-based testing, is shown also. This may be utilized to provide spacecraft simulator generated data to the MOC and SDS during the on-orbit phase.

3.0 Operational Requirements

The operational requirements for the Mission Operations System (MOS), as delineated by Reference (1), are as follows:

3.1 The MOS must be designed to support launch, early orbit checkout, and the first 800 days of science life mission phases.
3.2 Instrument operations will be decoupled from the S/C bus operations, i.e., instrument commanding and assessment will be done by the SOC.
3.3 There will be one track/day/vehicle.
3.4 The MOS must support operations for two concurrent S/C.
3.5 The MOS must support near real-time, bent-pipe instrument commanding and provide bent-pipe telemetry to SOC for each track.
3.6 The MOS must support wheel desaturation maneuvers.
3.7 Playback SSR data on each track.
3.8 Do not overwrite SSR data.
3.9 Maintain the Mission Elapsed Time (MET) correlation to Coordinated Universal Time (UTC) within 0.5 seconds and provide correlation data to the SOC.
3.10 Ground System Requirements
3.10.1 Provide a near real-time and time tagged commanding interface to the S/C for the SOC up to 1 hour (~450 kbits) of instrument commands/track.
3.10.2 Provide a telemetry interface for a near real-time access from a file-based system for the following: housekeeping, science, attitude history, time correlation, real-time broadcast, and navigation data.
3.10.3 Provide C&DH command storage space of 400 command packets/instrument for instrument time tagged and macro commands.
3.10.4 Provide the capability to identify dropped packets from an SSR playback within 1 hour of receipt.
3.10.5 Create a real-time S/C simulator after launch by assembling C&DH and G&C brassboards to G&C environmental simulator.

4.0 Mission Operations System Overview

The STEREO MOS consists of the two spacecraft, DSN ground stations, MOC, and SOC (Figure 4-1) and their respective operational teams. The STEREO spacecrafts will be operated by JHU/APL utilizing DSN for vehicle communications. The spacecraft bus and the instrument suite will be operated in a decoupled fashion; the MOC will provide all support of spacecraft bus operations, the SOC will operate all instruments on both S/C although communication between the SOC and the spacecraft will necessarily flow through the MOC. All spacecraft servicing, including

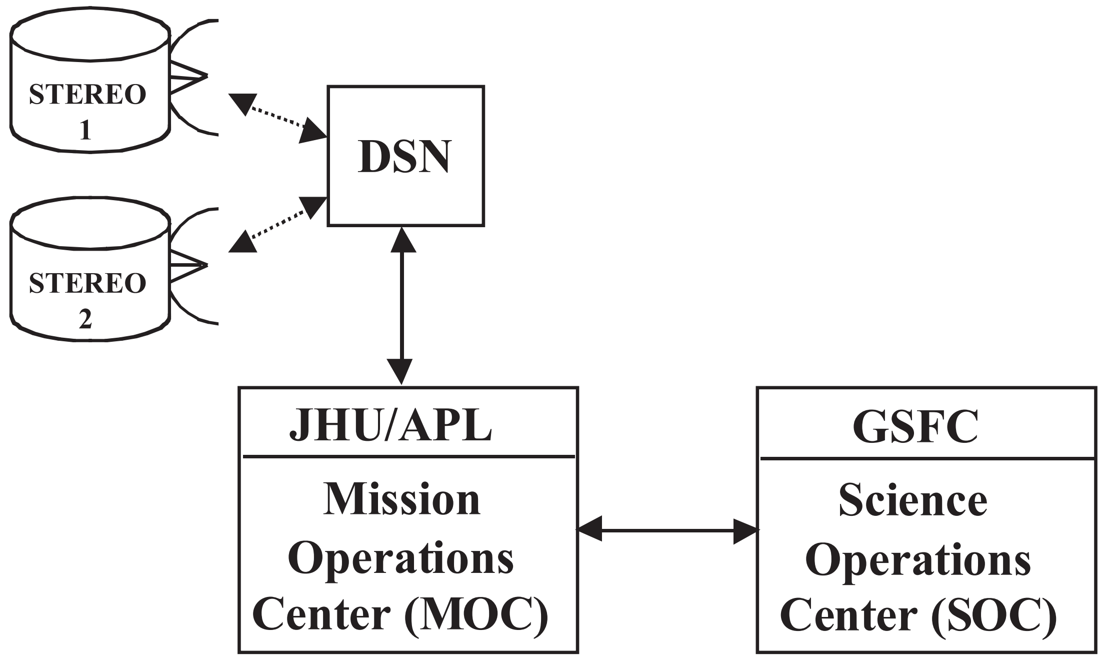

Figure 4-1 STEREO MOS Architecture

commanding and data recovery will occur during a single (nominal) ground track each day. This track will extend over a four to eight hour window, depending on the vehicle range from Earth. Spacecraft command messages will be uploaded and real-time engineering data will be received and evaluated to assess spacecraft health. The SSR will be played back on each track and all science data flowed to the SOC in near real-time.

A descriptive overview of the spacecraft and its operation are discussed in Sections 2.2, 2.3, and 2.4.

4.1 Mission Operations Center (MOC)

The MOC has the primary responsibility of management of the spacecraft bus including the development of command messages and the uplink to the spacecraft by way of DSN. Recovery of spacecraft bus engineering (state-of-health) telemetry and the performance analysis based on this telemetry is also performed at the MOC. The MOC receives instrument command messages (packets) from the SOC and, after verification that the SOC has prepared the commands, queues these for uplink to the spacecraft based on start and expiration times appended to the command messages by the SOC.

The MOC is located at JHU/APL in Laurel, MD. It is operated by the MOT and is nominally staffed during business hours, five days per week.

Figure 4-2 illustrates the MOC and includes interfaces to other MOS elements. NASCOM communication lines connect the MOC to DSN. Communications to the SOC are via the Internet,

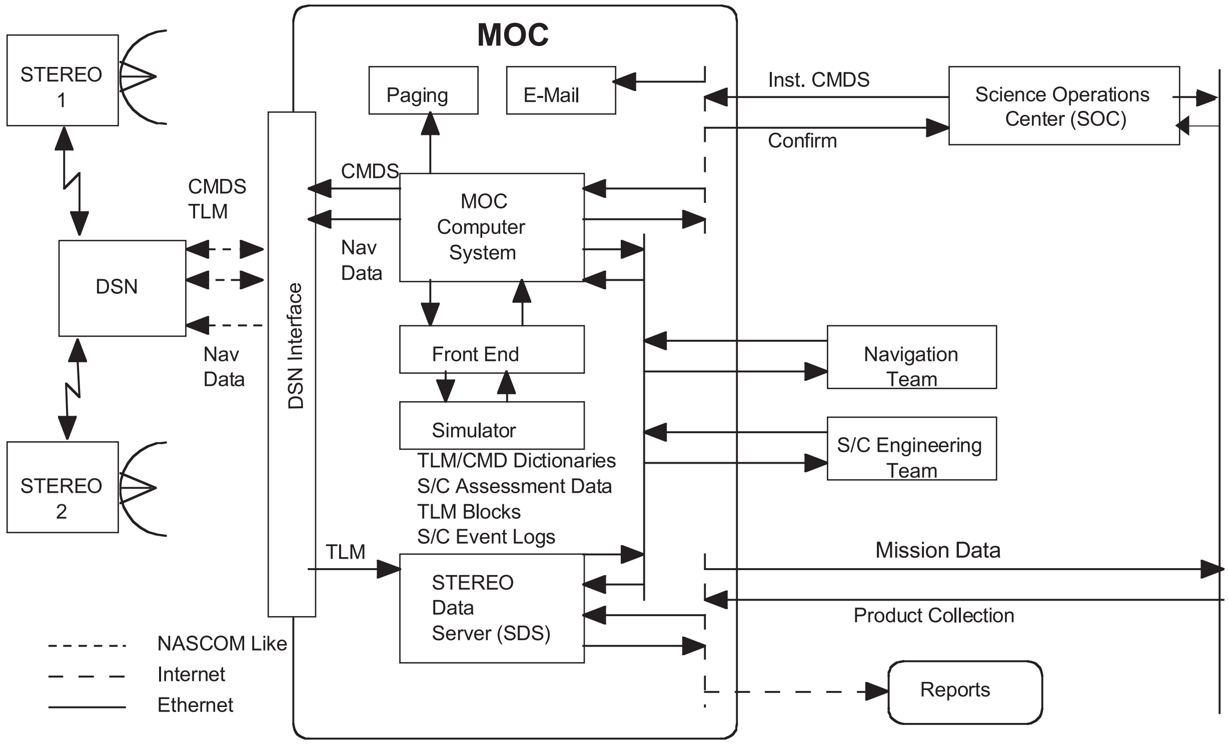

Figure 4-2 MOC Interfaces

with a modem backup. Within the MOC are workstations to support spacecraft commanding, spacecraft bus monitoring and analysis, primary and backup databases, and user files. The primary and backup command workstations are isolated from the rest of the MOC by a router/firewall. Commands to the spacecraft may *only* be issued from these workstations. Real-time telemetry is flowed through the firewall to the remaining workstations. Received telemetry (both real-time and SSR playback retrieved from the SDS) may be processed and displayed on these workstations. Main data paths are at least 100 Mbps Ethernet, with distribution within the MOC to some workstations and printers on 10 Mbps Ethernet. Unix based workstations are provided for MOT and spacecraft bus engineering team use. The SDS and the Spacecraft Simulator are also contained in the MOC.

4.1.1 STEREO Data Server

The SDS is located at JHU/APL in the MOC and functions as the central repository of spacecraft bus engineering telemetry, command files, mission planning data, ground system telemetry and status, and external correlative measurement data for the MOC. The SDS may be accessed, continuously (24 hours/day), via standard Ethernet/Internet-type communication lines.

A detailed diagram showing the data flow in and out of the SDS is illustrated in Figure 4-3. The SDS receives raw spacecraft data (SSR playback science and engineering) directly from the DSN (or a simulator). The MOC may transfer real-time telemetry and command data to the SDS. These data enter the SDS at the Telemetry/Command (TLM/CMD) Ingest process. A tape backup is

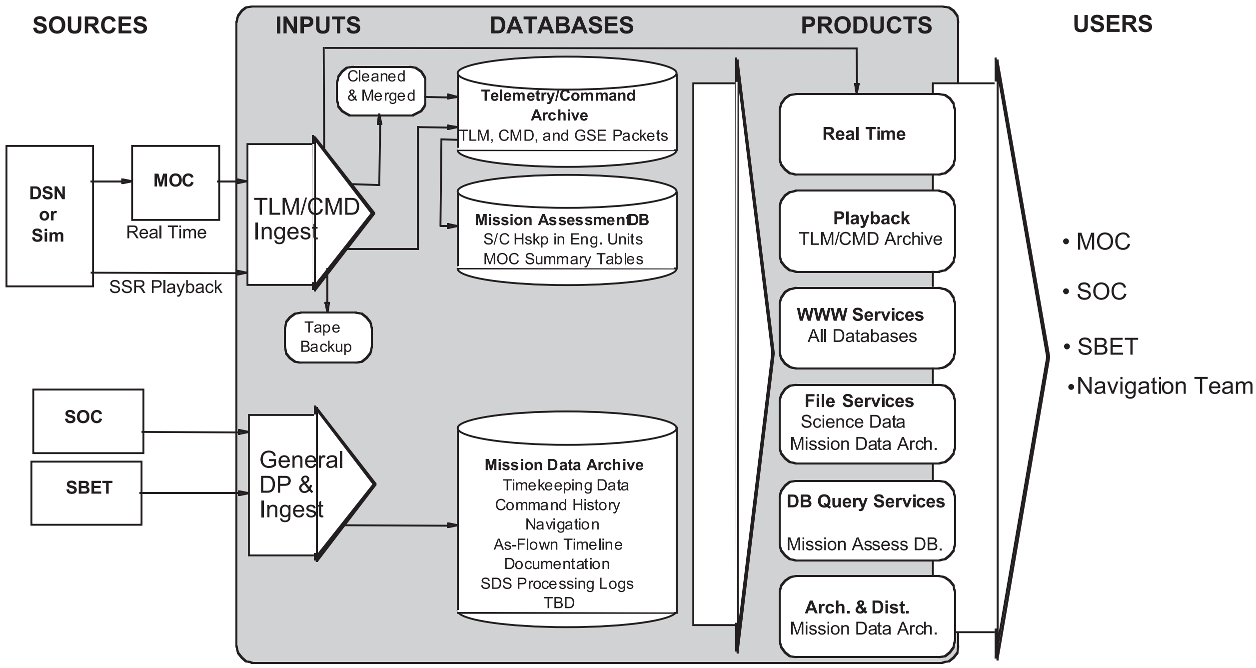

Figure 4-3 STEREO Data Server and Interfaces

provided here. This process may also serve real-time data (science and engineering telemetry) to users. Ingested data are cleaned and merged, then stored in the Telemetry/Command Archive database as packets. Engineering telemetry and command data, as well as ground system, data are stored here. Some data are stored in a Mission Assessment database (DB). Included are converted (to engineering units) spacecraft bus engineering telemetry and MOC processed data. The Mission Data Archive database including timekeeping data, navigation and orbit data, timelines, processing logs, command history, and documentation. These are inputted and processed by the General DP (Data Processing) and Ingest function.

The databases maintain the data products required by the users. These products, as shown, include real-time engineering telemetry, playback engineering telemetry and command files, World Wide Web (WWW) access to all databases, files services to access archived data, mission assessment database, and archival and distribution services. The users will include the SOC, MOC, and spacecraft bus engineering team.

On a daily basis, the SDS stores all command packets produced by the SOC and the MOC as well as all spacecraft engineering data and ground system engineering data. The real-time data acquired by the MOC is flowed to the SDS as it is received (the MOC also retains this data to support real-time operations and assessment). The SOC may access this real-time data from the SDS, with only a short time delay incurred. The SSR playback data will be transferred to the SDS and will be distributed to both the MOC (spacecraft bus engineering data) and the SOC (instrument science and engineering data). Spacecraft telemetry data are referenced (stored) by virtual channel number at the SDS and the data are accessed as ‘streams’ of data (that is the unprocessed telemetry is flowed in time order from the SDS to the user). The user must accept all data of the specified virtual channel and over the time interval specified. In the case of real-time data transfer, this ‘spigot’ is either open or closed. When open, all real-time data is flowed to the user as the SDS ingests the data.

4.1.2 Spacecraft Simulator

A S/C Simulator will provide real-time verification of command sequences, software loads, and expected telemetry. It also will be used for training new Mission Operations staff.

It will comprised of equipment used during the I&T of the S/C.

4.2 Deep Space Network (DSN)

The DSN will be used to provide communications to both spacecraft from launch to end of life (EOL). The use of all three DSN antenna facilities, Goldstone, Madrid, and Canberra, are required to determine the elevation component for the navigation of each spacecraft. Nominally, one track per day per spacecraft will be conducted using the 34-meter BWG antennas.

The MOC is connected to the DSN via NASCOM links. Orbit data for each spacecraft will be provided periodically to DSN.

4.3 Science Operations Center

The SOC has the responsibility for the operation and assessment of all instruments on the spacecraft. This includes the following instrument operational tasks:

- Planning, scheduling, and generating instrument commands
- Instrument health
- Calibration
- Synchronization of instrument operations between S/C

Instrument command loads will be assembled as packets and transferred to the MOC prior to a scheduled spacecraft track. Separate command messages are required for each S/C. Included, as part of the command packet transfer to the MOC, will be certain identifying data to be used by the MOC to verify that an authorized source has generated and transferred the data. The SOC will attach data that specify both the earliest and latest times that the attached command packet may be uplinked to the instrument. The SOC will be responsible for the verification and validation of instrument response based on the uplinked command load. The MOC will be responsible for the delivery of the content of the packets to the addressed instrument but assumes no responsibility regarding the actual commands.

Similarly, the processing of science and engineering data pertaining to all instruments is the responsibility of the SOC. Recorded science and engineering data will be available to the SOC via the SDS.

The SOC also has the responsibility for archiving all science data for the duration of the mission for both S/C. The MOC will provide, via the SDS, the following data products in file format to the SOC:

- Science
- Real-time space weather
- S/C bus and instrument engineering

- Attitude history
- Time correlation
- Navigation

The MOC is connected to the SOC via a commercial Internet connection with a modem backup and will utilize standard TCP/IP protocols. The SOC is located at the GSFC in Greenbelt, MD.

Real-time command and telemetry monitoring operations are expected to be available although such operations are considered only as contingency operations (including the possibility of initial early on-orbit operations).

The SOC will be provided ground system planning information by the MOT. Included are the schedules for DSN tracks schedules, track plans, orbit data, S/C bus health, etc. All such information will be provided via the SDS.

4.4 Mission Operations Team

The following four teams will work to support the STEREO mission.

- Mission Planning Team (MPT)
- APL Mission Operations Team (MOT)
- DSN
- Science Operations Team

This section will only discuss the APL MOT.

The MOC is staffed and operated principally by the MOT. The Spacecraft Bus Engineering Team (SBET) and instrument teams (when the Test SOC is installed) will provide staffing to support specific operations.

The MOT is responsible for all spacecraft and commanding, the recovery of all spacecraft telemetry, the assessment of spacecraft bus performance, and the control, monitor and performance assessment of all ground components necessary to support these functions. During the Normal Operations mission phase, the MOT staff will be comprised of the following:

- Flight Operations Manager
- Spacecraft Specialists (two/vehicle)
- DSN Scheduler
- System Maintenance Engineer

The *Flight Operations Manager* is responsible for the following functions:

- Overall conduct of operations
- Providing a central point of contact between the MOT and the external MOS
- Manage any adjustment to the operational schedules
- Interface between science personnel and the MOC
- Occupy the position of a DSN Scheduler, when necessary

The *Spacecraft Specialists* are responsible for the following functions:

- Spacecraft bus operational planning and command generation
- On-line spacecraft control and readiness tests
- Spacecraft bus and ground system performance assessment
- Lead spacecraft and ground system troubleshooting activities
- Operation of the S/C Simulator

The *DSN Scheduler* is responsible for the following:

- Advance, weekly, and daily scheduling of DSN antennas for both S/C
- DSN liaison

The *System Maintenance Engineer* is responsible for the following:

- Normal maintenance and calibration of the MOC components
- MOC communication connections
- MOC software upgrades

Each Spacecraft Specialist will have detailed operational knowledge of the commands, telemetry, and constraints of the each S/C. This will be acquired by assisting the SBET and I&T teams with S/C integration and test. Each Spacecraft Specialist will be able to carry out all on-orbit operational activities for each S/C, i.e., planning, control, and assessment.

During the Early Operations mission phase, the MOC will be staffed 24 hours/day and seven days/week. During the Normal Operations mission phase, the MOT staff will transition to business hours, five days/week. This will require the validation of many automated MOC procedures and autonomy rules on the S/C. Occasional off-business hours scheduling is likely to occur during some special operations including contingency activities.

4.4.1 Spacecraft Bus Engineering Team

Although the MOT will be entirely capable of operating the spacecraft bus and detecting and responding to anomalies, the Spacecraft Bus Engineering Team (SBET) is considered an essential and integral adjunct. The SBET consists of the spacecraft subsystem development teams, along with the MOT. Together, they maintain the complete technical knowledge base regarding the operation and performance of the spacecraft bus. During the subsystem level testing and the follow-on spacecraft bus I&T, the MOT, working side-by-side with the SBET, will acquire the knowledge necessary to operate the spacecraft bus on-orbit.

After launch, when operational anomalies are uncovered by the MOT, an immediate assessment will be made to ascertain the ability of the MOT, with the accrued knowledge and existing contingency procedures, to correct the anomalous operation within a reasonable time frame. If the anomaly is not clearly understood or if the recovery action is uncertain, the appropriate SBET will be notified and will support the MOT during the recovery process.

To maintain the SBET in a continuous state of readiness, the MOT will provide periodic performance reports to the SBET. The SBET will have access to all engineering telemetry stored on the SDS and will be able to access this from their office personal computers.

MOC workstations will be available to the SBET when direct support of on-orbit operations is necessary. Certainly this will occur during the Early Operations mission phase and most probably during periods of anomaly investigations.

4.4.2 Operations Planning

Operations planning will consist of the following activities necessary to support a scheduled track:

- Track scheduling
- Maintenance activity scheduling
- Managing the uplinking of instrument commands
- SSR management
- Timekeeping management
- Wheel desaturation management
- Navigation management
- Track Plan Generation

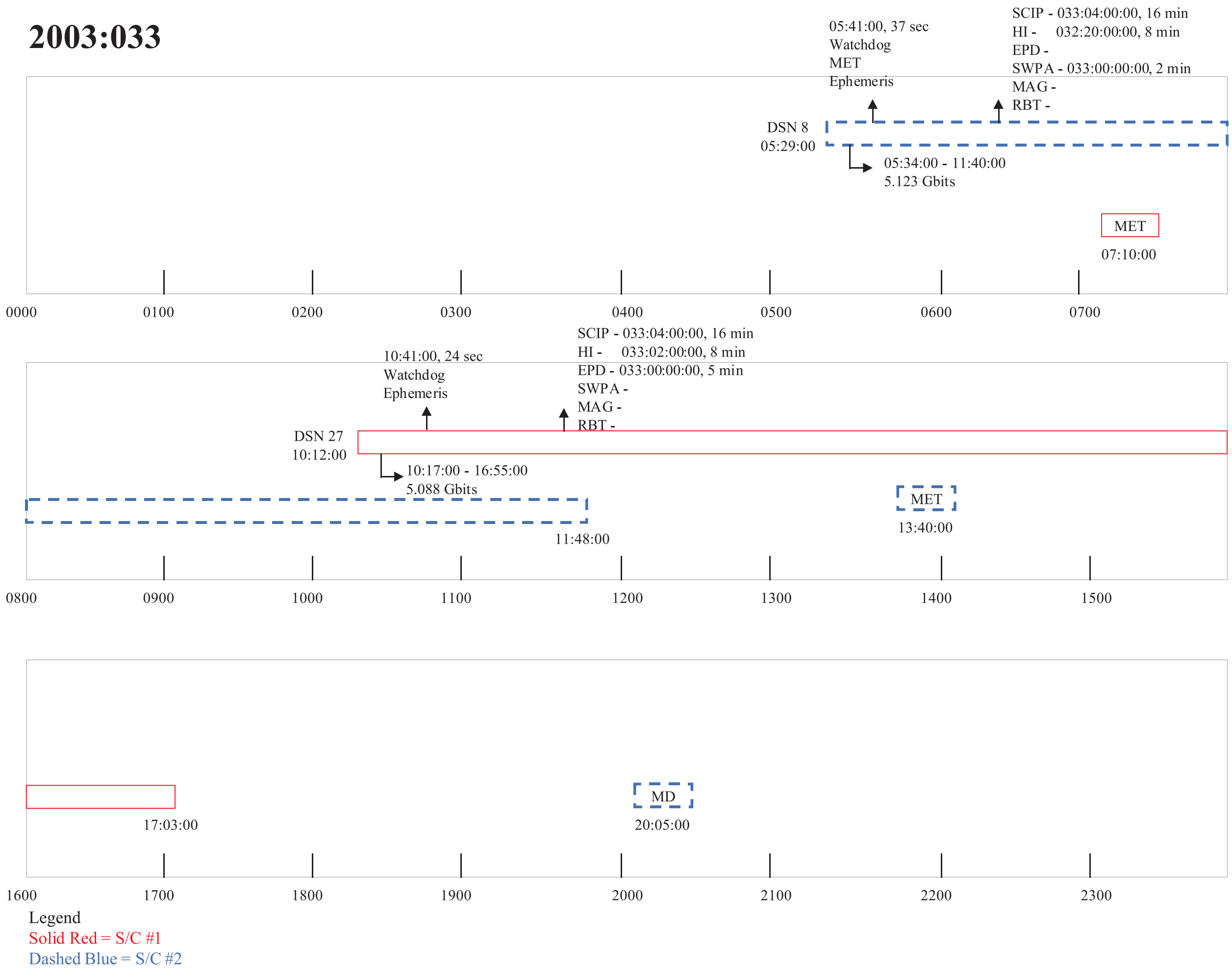

Figure 4-4 Daily Timeline

The STEREO operations' planning consists of planning a week of tracks in advance. The STEREO planning week starts on Monday. For example, on Monday of Week – 1, the MOT will be planning the following week. The MOT will determine the operational requirements of the spacecraft bus over the next week and will prepare the necessary command packets to satisfy these requirements.

Each day, the track schedule for the next day will be reviewed to ensure that it is up to date. A Daily Timeline (Figure 4-4) will be generated graphically depicting the significant operations on both vehicles for the next day. The final daily operations planning task is to generate a track plan.

4.4.2.1 Track Scheduling

The DSN track requirements for each spacecraft will be scheduled well in advance. Unlike most planetary missions, there is no encounter phase for the STEREO mission. Since the prime science phase is continuous, starting shortly after S/C checkout and continuing through to EOL with every track having the same priority.

Nominally, there will be one DSN track per day per spacecraft.

Planned DSN track schedules for both S/C will be stored on the SDS.

4.4.2.2 Maintenance Activity Scheduling

A Maintenance Event (ME) is an activity scheduled on the S/C and executed via commands for the purpose of maintaining the health of any spacecraft bus subsystem or managing the resources of the spacecraft. The MOT is responsible for the evaluation of the spacecraft bus and will plan and schedule all MEs. With the assistance of SBET, the health of the spacecraft will be managed by evaluating component performance and generating command sequences as Maintenance Events.

Maintenance activities for the instruments are the responsibility of the SOC and are not addressed in this document.

There are two categories of MEs, routine and sporadic. A routine ME has a set execution frequency, i.e., every track, daily, weekly, etc. A sporadic ME has no set execution frequency. A sporadic ME will be initiated based on evaluation of the telemetry data. A ME may consist of the following:

- Watchdog timer resets
- Updating the S/C ephemeris
- Updating MET
- Wheel desaturation
- S/C engineering buffer dumps
- Maintaining spacecraft bus macros
- Maintaining autonomy rules
- PPT voltage and temperature adjustments
- Software changes

4.4.2.3 Managing the Uplinking of Instrument Commands

Instrument command packets will be uplinked during each track. These instrument command packets will be prepared, in advance by the SOC for all instruments and will arrive at the MOC no later than two hours prior to the scheduled primary track. In general, however, instrument command packets should arrive well in advance of this deadline. Instrument teams are encouraged to forward command packets only once a week that includes packets designated for uplink each day of the week. Scheduling data, to accompany the command packets, will indicate when to actually uplink these packets. Until then, they will be stored at the MOC. The means by which these command packets are scheduled has been described previously (see Section 2.5.1). Briefly, the command packets will be augmented with two time tags. One time tag will specify the earliest time that the command packet can be uplinked to the instrument and the other will indicate the latest time that this can happen (the expiration time). Those command packets that have a start time exceeded by the next track start time will be queued for uplink at that track to that S/C. Each instrument on each S/C will have its own set of storage buffers, staging and uplink queues, in the MOC.

The MOT has the responsibility of managing the command uplink for all spacecraft commands, to the bus and to the instruments. Accordingly, the MOT must have some knowledge of just what is contained in the instrument uplink queues. The actual instrument commands is unimportant, but the quantity of data to be uplinked is important to the MOT. The scheduled track will offer limited uplink capacity. Nominally, an hour of instrument command uplink time for each track has been set. This translates into approximately 450 kbits of instrument command data for all instruments per track per S/C. Commanding during this interval is not continuous. Rather, the command packets, packed into transfer frames, are uplinked and verified, transfer frame by transfer frame. The verification process can slow the uplink down some. Also, any retransmissions will add additional delays. Therefore, instrument teams are encouraged to operate their instruments as command efficiently as possible.

The MOT will examine each instrument uplink command queues, sorted chronologically by expiration time, to assess the transmission time necessary to uplink the content. MOC software will provide status of the content of the queue including expiration time. The status provided will also indicate estimated transmission time of all queued commands. Further, the time to transmit all the commands in the uplink queues of all instruments will be provided. In Section 2.5.1, the concept of a ‘grocery bar’ was introduced. The purpose of this is to separate, in the active command queue, those packets which will be uploaded during the upcoming track and those which are being purposely held back for a later track. The manipulation of this ‘grocery bar’ by the MOT will provide the necessary traffic control of instrument commands so that the required instrument command packets to be transmitted, along with the real-time and time tagged spacecraft bus command packets, will meet the track uplinking allocation time.

Manipulation of this ‘grocery bar’ is the only control of the queued instrument command packets, except flushing (removing all command packets from) an instrument’s queues altogether. No editing control is available to the MOT. Flushing will only be permitted when authorized by the instrument team associated with that particular queue.

Spacecraft bus commands will receive priority during uplink operations and these can interrupt instrument command uploads. The MOT will actually have control over this process and can withhold spacecraft bus command uplink when necessary. The track plan, prepared by the MOT, will specify precisely what command packets, instrument and spacecraft bus, real-time and time tagged, are to be uplinked during a given track period.

4.4.2.4 Solid State Recorder (SSR) Management

As a baseline, the SSR will be played back on each track. For the STEREO mission, manual MOT management of the SSR will be minimized. This will be done by implementing software on the S/C and in the MOC to monitor CCSDS transfer frames on each virtual channel. It will automatically detect and retransmit missing transfer frames during an SSR playback. The state of the playback reception at the end of a track will be saved for the next track.

Currently, the SSR recorded data is divided into the following four prioritized data streams:

1. G&C Anomaly
2. Engineering Anomaly
3. Science
4. Nominal Engineering

At the beginning of an SSR playback, all G&C and engineering anomaly data is downlinked first, followed by science data, and then nominal engineering data. The amount of SSR data transmitted during a track will vary over the duration of the mission. Initially, the entire SSR will be played back each day. However, as the S/C to Earth range increases, the amount of SSR data played back will decrease. Eventually, during an eight-hour track, the entire SSR cannot be played back.

At the beginning of the mission, the SSR playbacks will be controlled by using real-time commands. As the mission progresses, to save communication time and increase the data downlinked, as a goal, the control of SSR playbacks will transition from uplinked commands to an on board autonomy rule.

An SSR Log will be maintained on the Mission Data Archive in the SDS to provide a chronological history of all science data recovered to date.

4.4.2.5 Timekeeping Management

The Mission Elapsed Time (MET), as generated by the C&DH, will be maintained to the required 0.5 seconds of UTC for each S/C. This will be accomplished by a software process in the MOC that will periodically estimate the time offset, based on the S/C oscillator drift rate along with known system time delays (one-way light time and internal S/C time delays). This time offset or update will be periodically uplinked to the S/C, as a time tagged maintenance event, to maintain the MET requirement.A history of the time updates for each S/C will be maintained in the Mission Data Archive on the SDS.

4.4.2.6 Wheel Desaturation Management

As S/C momentum builds in the RWAs it will need to be dumped periodically using the thrusters in the Propulsion subsystem. The spacecraft will have the capability to autonomously dump momentum. This will occur at four day intervals or greater. In the event that the function is turned off, the MOT will monitor the RWA speeds and send commands, as a time tagged maintenance event, to desaturate the wheels on each S/C. So as not to interfere with the science data collection of the instruments, time will be allotted on a daily basis for possible momentum dumps. If a momentum dump is to occur, instruments will be warned over the 1553 bus.

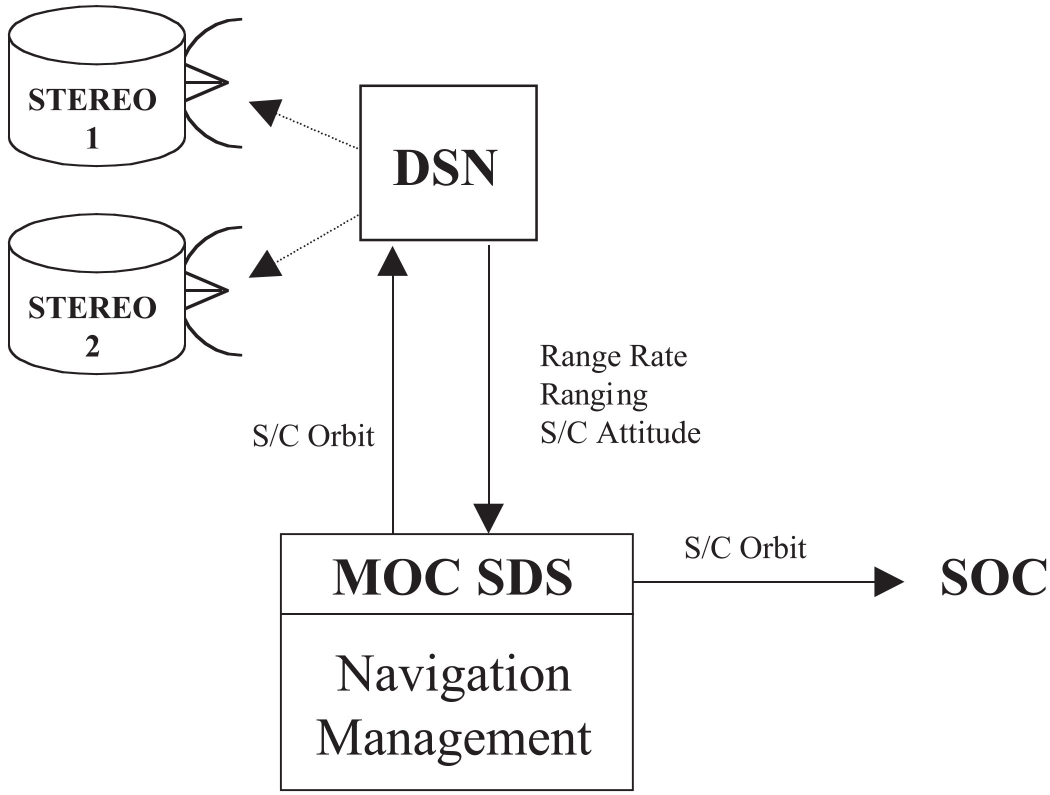

Figure 4-5 Navigation Data Flow

4.4.2.7 Navigation Management

The G&C engineering team at JHU/APL will conduct the navigation management for each S/C. Using the Doppler range rate, ranging, and S/C attitude data, the S/C orbit will be determined using the Goddard Trajectory Determination System (GTDS) (Figure 4-5). Required navigation data will be stored in the Mission Data Archive on the SDS. S/C orbit data will uplinked to the S/C AFC and will be provided to DSN at required intervals for proper antenna pointing.

4.4.2.8 Track Plan Generation

The track plan is the data product that is used by the MOT to conduct a track. After all activities for the track have been scheduled, a track plan will be generated. It will list the following information chronologically:

- S/C ID
- Track ID
- S/C range
- Expected downlink bit rate
- DSN station
- Acquisition of Signal (AOS) time

- SSR Playback start time
- SSR Playback total bits downlinked
- Spacecraft bus command uplink start time
- Spacecraft bus command total bits uplinked
- UT maintenance events (i.e., MET update, momentum dump, etc.)
- Real-time maintenance events
- Instrument command uplink start time
- Instrument command packets earliest uplink time & bit size (for each instrument)
- Instrument command start time total bits uplinked
- SSR Playback stop time
- LOS time

4.4.3 Operations Control

Operations control will consist of those activities immediately prior to and following a scheduled track and will include a pre-pass readiness test and track operation. A pre-pass readiness test of the ground facilities will include the following:

- Data circuits to DSN
- Voice circuits to DSN
- MOC (elements required to support real-time operations)

This testing will assure that the necessary elements of the ground system are functional and properly interconnected as required to support the prepared track plan.

At the conclusion of pre-pass testing, the respective operating teams will make any final configuration adjustments necessary to support the upcoming track.

Spacecraft commanding may only be initiated at the MOC command workstation. Three MOC workstations, protected by the security firewall, will be designated as command workstations, two of these may be activated to support concurrent S/C commanding during the track, the other will be ready as a ‘hot’ backup in case one of the active workstation fails. All MOC workstations may display processed telemetry data from both the spacecraft and the ground system. In general, the active command workstation will be dedicated to only commanding and the verification thereof, and to assure that the track plan is properly executed. The remaining workstations will be used to monitor spacecraft and ground system performance via processed telemetry.

After a track has been completed, an As-Run Track Plan will be generated. It will essentially be a marked-up or as-flown Track Plan and will also list the following:

- Amount of real-time data received
- Amount of SSR data received
- Available SSR space
- Status of all uplinks

The STEREO MOT will transition to unattended tracks after the Early Operations Checkout phase.

4.4.4 Performance Assessment

The objective of the Performance Assessment function is to maintain the health of the spacecraft bus subsystems and evaluate its performance to collect data. Subsystem performance assessment consists of routinely determining the status, configuration, command verification, and performance of each spacecraft bus subsystem. The following assessment tasks will be performed by the MOT:

- Alarmed telemetry processing
- Command verification
- Trend analysis
- Providing data to the SBET

Trend analysis will be conducted on critical subsystem components and on components that are known to degrade with time, e.g.:

- Battery voltage, pressure, temperature, state of charge
- SA temperatures and currents
- Operational and survival heater currents
- Propulsion tank pressure
- Other critical temperatures

These analyses will be conducted on a daily, weekly, monthly, quarterly, and annual basis.

The MOT will maintain a history of the changes to each S/C after launch in a configuration log. This will be used to maintain the S/C during processor resets and also for anomaly investigations. A PC spreadsheet implementation of the configuration logs will be sufficient.

The performance assessment function will be augmented by in-depth analysis of subsystem performance by the SBET. The SBET will have direct access to the engineering telemetry database stored at the SDS so any data may be accessed and processed to the satisfaction of the responsible engineer.

All of the performance assessment processing will be automated. Alarms processing, command verification, and trend analysis plotting will be done automatically. Each day, the MOT will review the output of these assessment processes. This will allow the MOT to minimize the daily time required to determine the health and performance of each spacecraft bus.

A Performance Assessment report for each S/C will be available on the Mission Data Archive in the SDS.

Performance assessment for the instruments is the responsibility of the SOC and is not addressed in this document.

4.4.5 Anomaly Investigations and Resolutions

The MOT is responsible for the safety and health of both S/C buses. They will lead and coordinate investigations with the SBET into all S/C bus anomalies. Anomalies identified both during a track and during performance assessment will be investigated. A cumulative database of all S/C bus anomalies for each S/C bus, from I&T through EOL, will be maintained in the Mission Data Archive on the SDS.

To assist the MOT during unattended tracks, an automated alarm notification system will be implemented. It will consist of an automated paging system that will notify the on-duty MOT staff of any alarms received during a track. Additional supporting engineering data will be emailed to an offsite (e.g., at home) MOT staff for further analysis of the problem. The notified MOT staff can then determine the appropriate action to take, i.e., request additional tracks, download additional engineering data, consult the respective SBET, etc.

4.4.6 Training

Staffing of the MOT will begin early during the development phase of the program. Every MOT staff member will have a detailed knowledge of the operation and constraints of both STEREO spacecraft and the MOS. The MOT will be assigned functional responsibilities necessary to provide both an education and essential tasks in support of the SBET as well as the I&T Team. The MOT will support the spacecraft subsystem engineering teams during the testing of these subsystems prior to delivery to the I&T Team. Components of the actual MOC will be employed to support subsystem testing, development of databases, display formats, data processing, and command sequences, etc., produced to support subsystem tests, may all be brought forward to the spacecraft system level support effort.

The MOT will develop procedures and will support the conduct of acceptance testing of the MOC hardware/software system.

During I&T phase, the MOT will be part of the I&T Team. They will define and produce the necessary system level tests to support the conduct of mission simulation tests. The S/C will be tested in the same manner as it will eventually be operated on-orbit. During the conduct of tests, the MOT Spacecraft Specialists will provide direct support to the Test Conductor as members of I&T Team. During this time, the function of the MOT will be to provide an assessment of the performance of the spacecraft subsystem under test. The MOT will assume the role of the Test Conductor during certain times within the I&T phase.

On-orbit mission simulations, where the spacecraft is operated as if it were on-orbit, will be conducted during the I&T phase. These tests will be conducted by the MOT just as they will during the actual on-orbit phase of the mission. All external operations supporting organizations and facilities (DSN and SOC) will be invited to support these tests. These tests will become the rehearsals of the MOT and the entire MOS.

The S/C Simulator will also be used provide training to new MOT staff after launch.

4.5 Mission Planning Team

The objectives of the MPT are to determine mission priorities, data recording and playback priorities, and manage the consumable resources (i.e., SSR storage, propellant, battery life, flash/EEPROM usage, power and thermal margins) on the S/C. The MOT reports directly to the MPT. It will consist of representatives from the following:

- Sponsor - NASA GSFC
- Lead Scientists
- APL Program Management
- Mission Operations

4.6 Mission Operations Working Groups

Prior to launch, at appropriate times as the development phase progresses, working groups will be organized by the MOT for the purposes of joint-preparation of essential procedures and related documentation that will establish the basis of the spacecraft and ground systems operating procedures.

It is the intent of the MOT to actually prepare the necessary documentation, the content however will require a collaborative effort with the spacecraft subsystems and instruments and all ground system development teams. Working groups will be formed to address the following:

4.6.1 Early Operations Plan

This working group, comprised of the MOT and SBET will establish the operational procedures necessary to conduct spacecraft bus operations between launch vehicle separation and until the spacecraft is declared operational.

4.6.2 Spacecraft Operating Rules and Constraints

This working group, comprised of the MOT and SBET, will identify rules and constraints to be imposed upon the spacecraft bus users and the MOT.

4.6.3 Spacecraft Autonomy Rules and Procedures

This working group, comprised of the MOT and SBET, will specify autonomy rules and the on board command sequences for the safe and efficient operation of the spacecraft bus.

4.6.4 Spacecraft and Ground System Contingency Plans and Procedures

This working group, comprised of the MOT, SBET and ground system development teams, will identify potential contingency situations for both S/C and the MOC.

4.6.5 Mission Operations System Processes and Interfaces

This working group, comprised of representatives of all elements of the total STEREO on-orbit operations support system (DSN, MOC and SOC) will address issues related to the overall ground system, hardware, software, teams, and procedures.

5.0 Operational Scenarios

5.1 Normal Operations

The Normal Operations scenario depicts the STEREO mission operations after both S/C have been checked out on-orbit. It is assumed that the range to both S/C is such that both SSRs can be played back in their entirety during their respective tracks. It does not consider operations during an anomaly on the S/C and/or the ground system.

The normal operational mode of the S/C consists of all instruments powered on, time tagged command capability enabled, the X-axis pointing at the Sun, and the HGA pointing at the Earth.

Upon arrival in the morning, the S/C Specialists will review all results from routine processes. This includes routine performance assessment plots, S/C alarm processing, As-Run Track Plans, instrument command queues, timekeeping management, SSR management, wheel desaturation management, and MOC and SDS equipment status. All S/C and MOC anomalies will be investigated and sporadic maintenance activities will be generated. Daily DSN scheduling activities, including track schedule generation, teleconferences, etc. will be performed.

During a spacecraft track, the MOT manages the flow of uplink command packets to the entire spacecraft. The flow of these commands is monitored to assure successful delivery to the destination. Automatic retransmission may be initiated if delivery is unsuccessful.

The downlink of real-time and SSR playback data are initiated under the control of the MOC. Real-time spacecraft bus engineering telemetry is converted to engineering units and tested against prestored acceptable limits and selectively displayed on visual media. Engineering telemetry that is played back is stored on the SDS. All science data and engineering telemetry from the playback are sent in a file format to the SOC via the SDS in near real-time.

5.2 Prelaunch

To be determined.

5.3 Early Operations

The Early Operations phase extends from launch vehicle separation to the declaration, by the Mission Planning Team, that the spacecraft bus and all instruments are capable of supporting the mission objectives. This phase encompasses launch load preparation through on-orbit checkout and evaluation. The launch load consists of a command sequence that will control, chronologically, critical spacecraft bus operations immediately after separation from the boost motor. These commands will provide:

- Deployment of solar panels
- Initiation of safe-hold mode attitude capture
- Powering on the IEM
- Initiation of the first track

The S/C separation sequence triggers the pre-stored launch load command sequence in the C&DH subsystem. This sequence will sustain spacecraft operations until the first scheduled DSN track occurs. The MOT, upon evaluation of the performance of the spacecraft bus at that time, may choose to alter the planned operational sequence or may elect to continue with the stored launch load sequence until the next scheduled track.

The primary activity during this Early Operations phase will be to checkout the subsystems and instruments on the S/C. The STEREO MOT will depart from the normal operations staffing plans to provide a more or less continuous 24-hour/day support. For the first few days, more than one DSN track per day will be requested. The Early Operations phase will be supported by combined MOT, SBET, and instrument teams all located at the MOC. Launch operations teams may also be utilized as required. Using the Space Shuttle for launch may increase the MOT staffing slightly for the first few weeks due to simultaneous checkout of both S/C.

5.4 Operations During Anomalies

The top priority for the MOT will be to maintain the S/C. When an anomaly is detected on the S/C, all necessary resources will be employed to resolve the anomaly with minimal mission impact.

There are three operational modes of each STEREO spacecraft; Operational, Safe Hold, and Earth Acquisition. Should an anomaly occur that degrades the Operational mode of the S/C, the S/C will autonomously transition to Safe Hold mode. This mode is designed to conserve power and configure the S/C for communications with ground system. For Safe Hold mode, the roll axis is controlled to point the X-axis within 1^0 of the Sun and the narrow angle (emergency) MGA within 1^0 of the Earth. All instruments will be powered down autonomously and time tagged commanding will be disabled.

If an anomaly occurs that degrades the Operational or Safe Hold modes of the S/C so that the roll axis is not known by the G&C subsystem, the S/C will autonomously transition to the Earth Acquisition mode. This mode is designed to conserve as much electrical power as possible and configure the S/C for communications with ground system. For Earth Acquistion mode, the AIE will use measured coarse sensor data to point the X-axis to within 1^0 of the Sun and rotate, or roll, the S/C at 1^0 per minute. This slow rotation will allow the MOT to re-establish communications with the S/C using the narrow angle (emergency) MGA once every three hours. The MOT can then stop the S/C rotation and download the necessary SSR data to determine the cause of the anomaly. All instruments will be powered down autonomously in this mode and time tagged commanding will be disabled.

The SOC will be notified via email of all anomalies that effect the mission.

5.5 End of Life (EOL)

The EOL for each S/C will be determined by the MPT. At that time, the S/C will be configured to conserve remaining resources. The necessary documentation, hardware, and software for re-establishing communications and operations of each S/C will be archived.

6.0 Acronyms and Abbreviations

AFC	Attitude Flight Computer
AIE	Attitude Interface Electronics
APL	Applied Physics Laboratory
BWG	Beam Wave Guide
C&DH	Command and Data Handling Subsystem
CCDSDS	Consultative Committee for Space Data Systems
CME	Coronal Mass Ejection
CONOPS	Concept of Operations
DSAD	Digital Solar Attittude Detector
DSN	Deep Space Network
EOL	End of Life
EPD	Energetic Particle Detector
FOV	Field of View
G&C	Guidance and Control Subsystem
GSFC	Goddard Space Flight Center
HGA	High Gain Antenna
HI	Heliospheric Imager
I&T	Integration and Test

ICD	Interface Control Document
IEM	Integrated Electronics Module
IMU	Inertial Measurement Unit
JHU	Johns Hopkins University
LGA	Low Gain Antenna
LVS	Low Voltage Sense
MAG	Magnetometer
ME	Maintenance Event
MET	Mission Elapsed Time
MGA	Medium Gain Antenna
MOC	Mission Operations Center
MOS	Mission Operations System
MOT	Mission Operations Team
MPT	Mission Planning Team
OSC	Ultra Stable Oscillator
PPT	Peak Power Tracker
RBT	Radio Burst Tracker
RS	Reed-Solomon
RWA	Reaction Wheel Assembly
S/C	Spacecraft
SA	Solar Array
SBET	Spacecraft Bus Engineering Team
SCIP	Solar Coronal Imaging Package
SOC	Science Operations Center
SDS	STEREO Data Server
SSR	Solid State Recorder
STEREO	Solar TErrestrial Relations Observatory
STF	Supplemented Telemetry Frame
STP	Supplemented Telemetry Packet
SWPA	Solar Wind Plasma Analyzer
TF	Telemetry Frame
TP	Telemetry Packet
UTC	Coordinated Universal Time
XB	X-Band