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 $\times$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

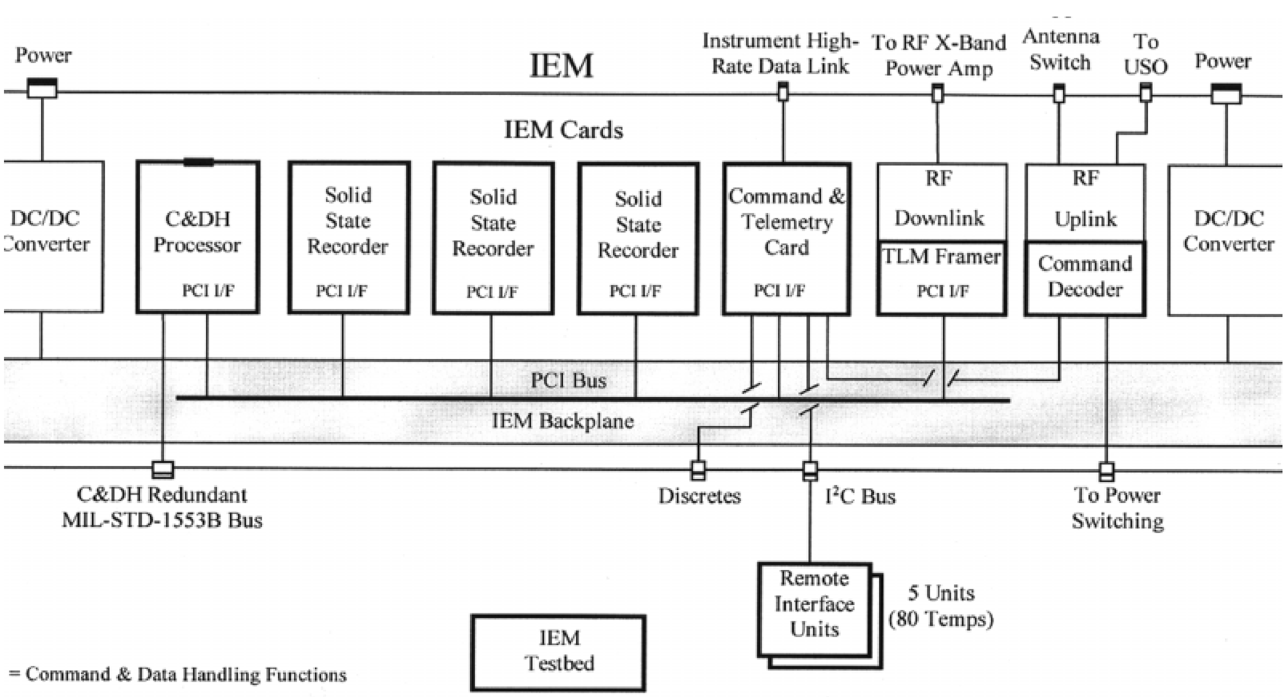


Figure 4-1 Integrated Electronics Module Configuration

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.

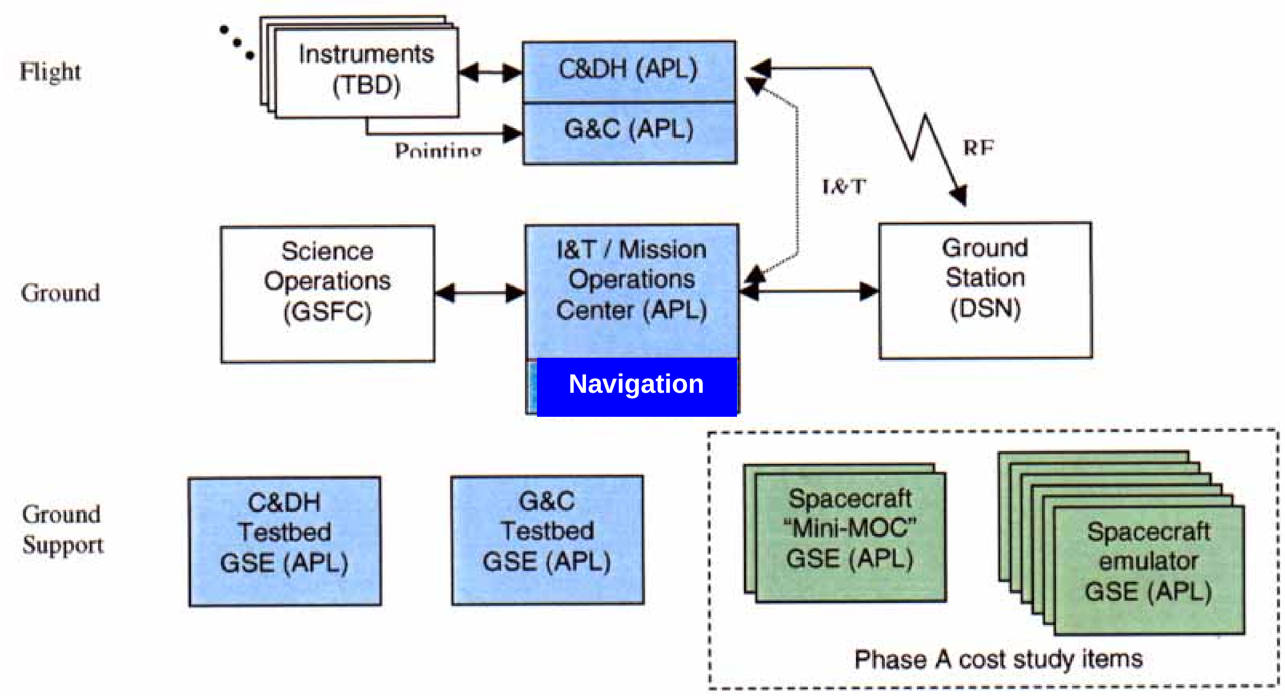


Figure 4-2 STEREO Mission Software Components

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

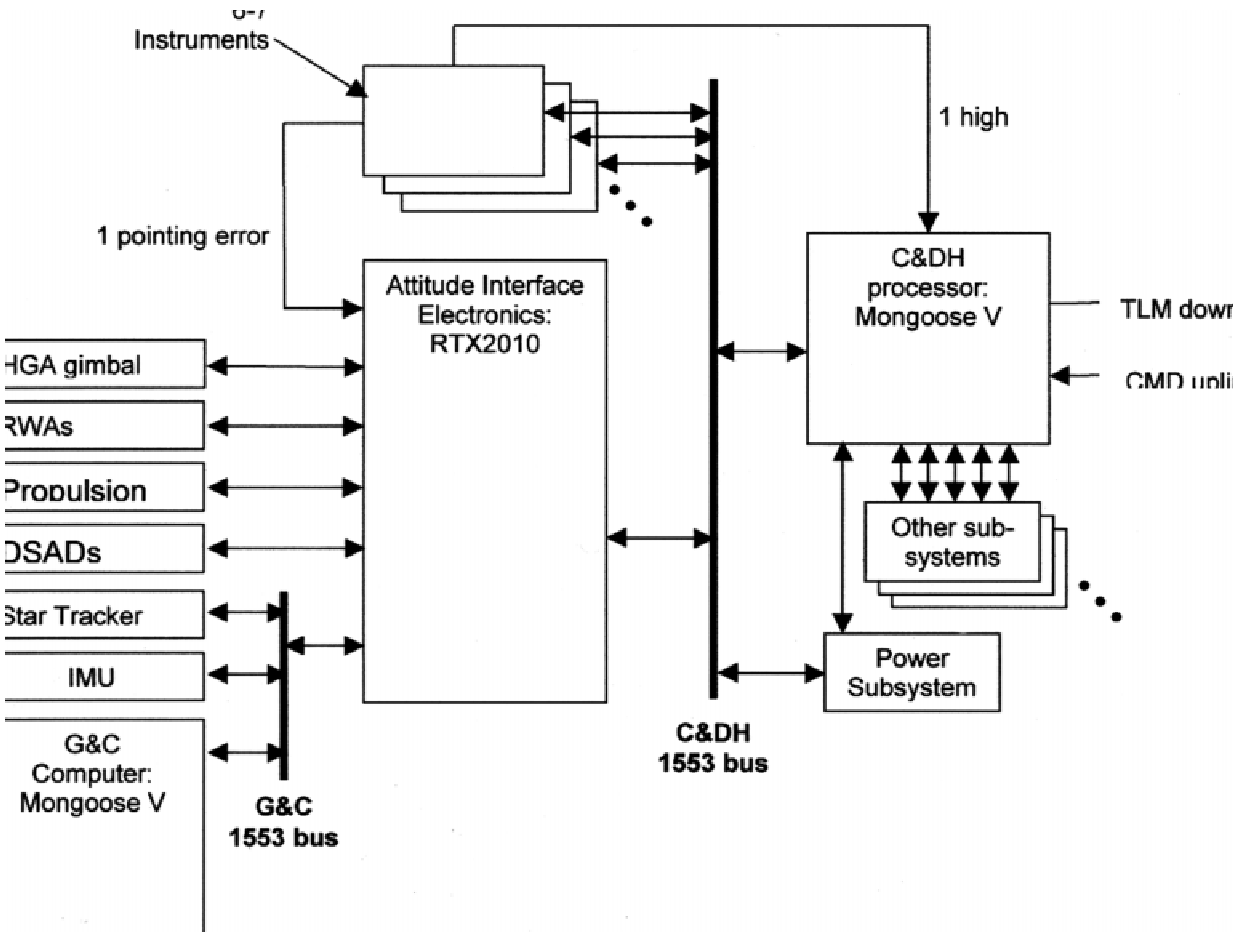


Figure 4-3 Baseline STEREO Data System Hardware Architecture

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™/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

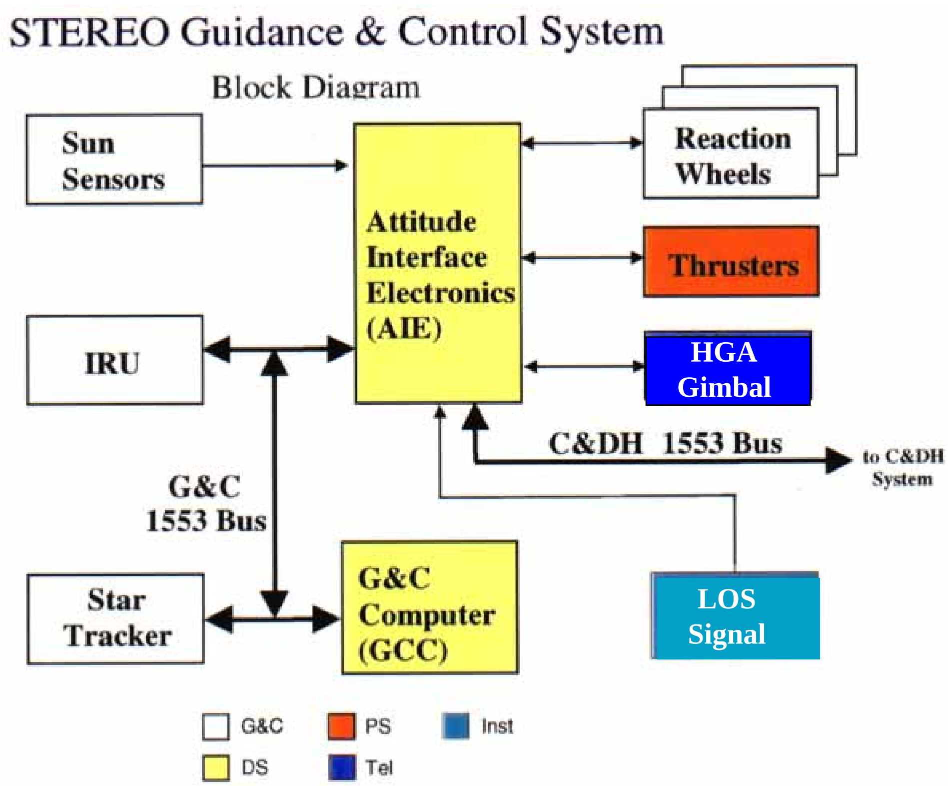


Figure 4-4 Guidance and Control Physical Block Diagram

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

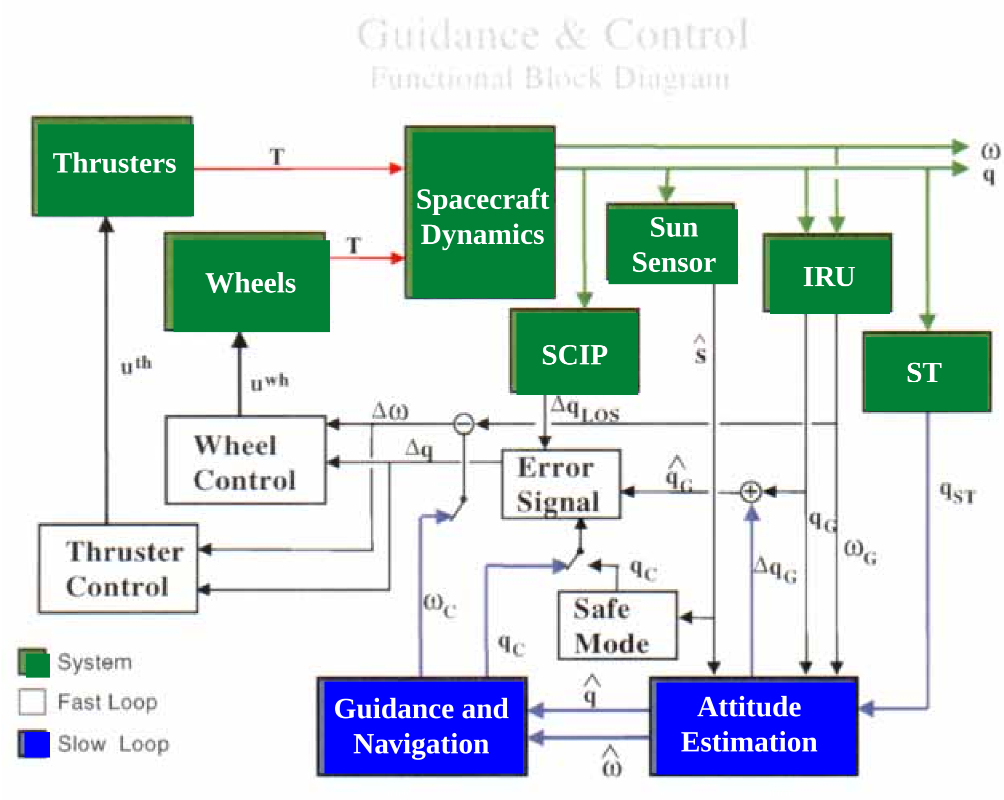


Figure 4-5 Guidance and Control Functional Block Diagram

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:	±20 arcsec	± 0.1 arcsec
Control:	±0.1°	±20 arcsec
Jitter:	30 arcsec RMS	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

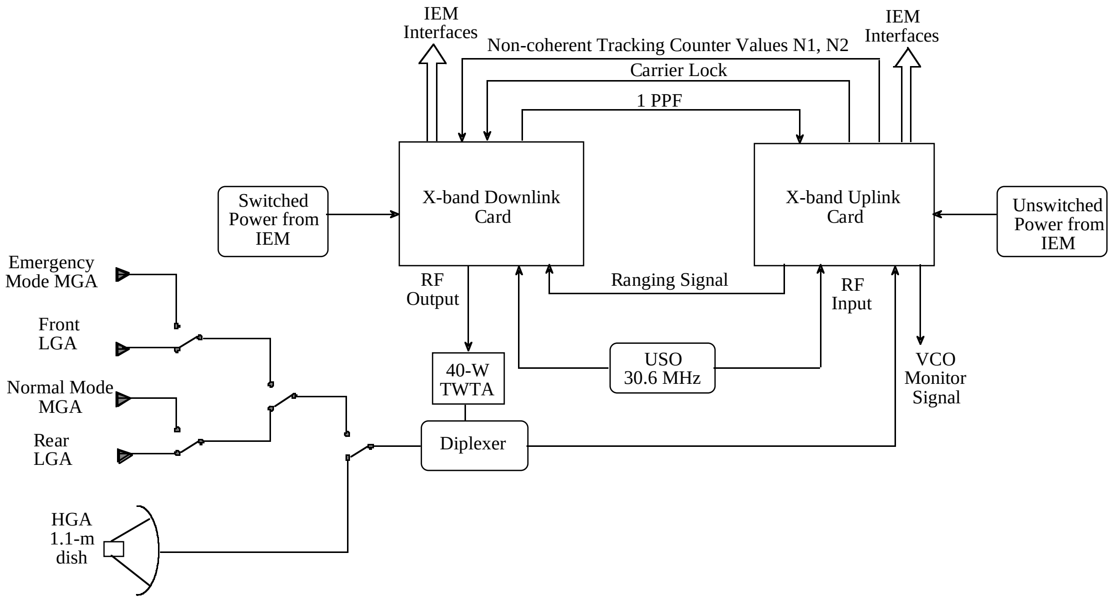


Figure 4-6 Preliminary STEREO X-band RF Telecommunications Block Diagram

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

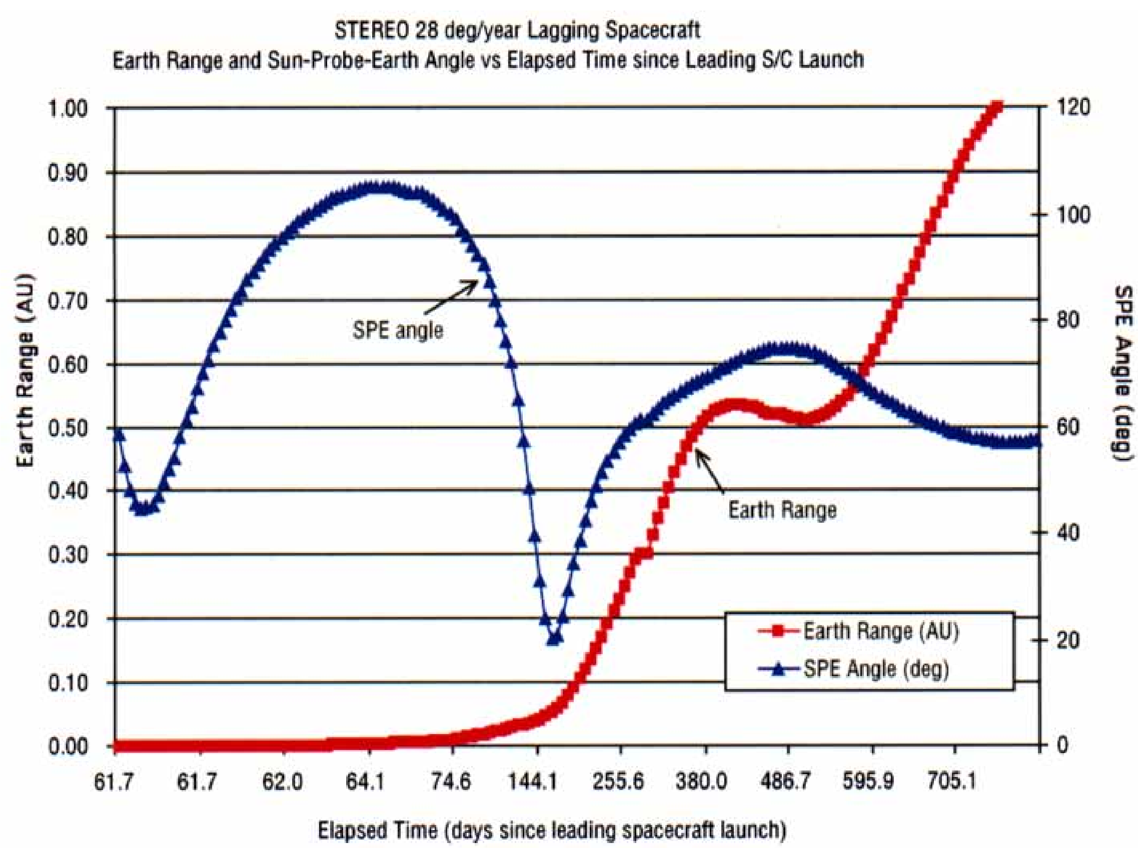


Figure 4-7 Lagging Spacecraft Earth Range and SPE Angle vs. Elapsed Time

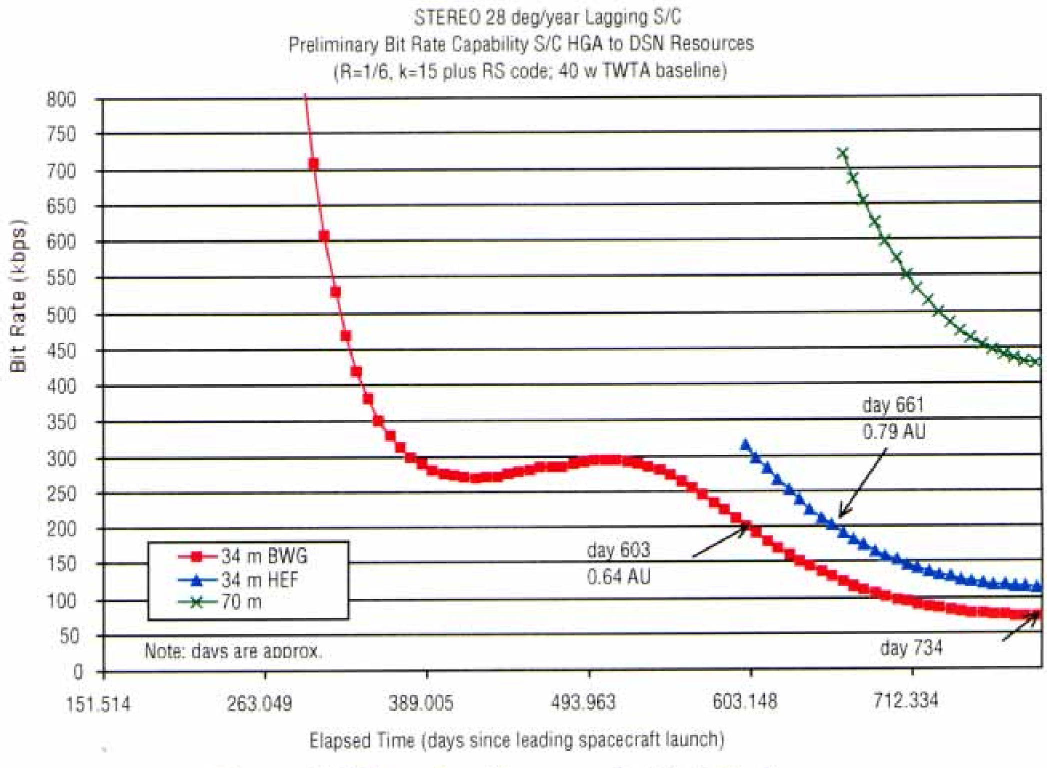


Figure 4-8 Lagging Spacecraft Link Performance

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

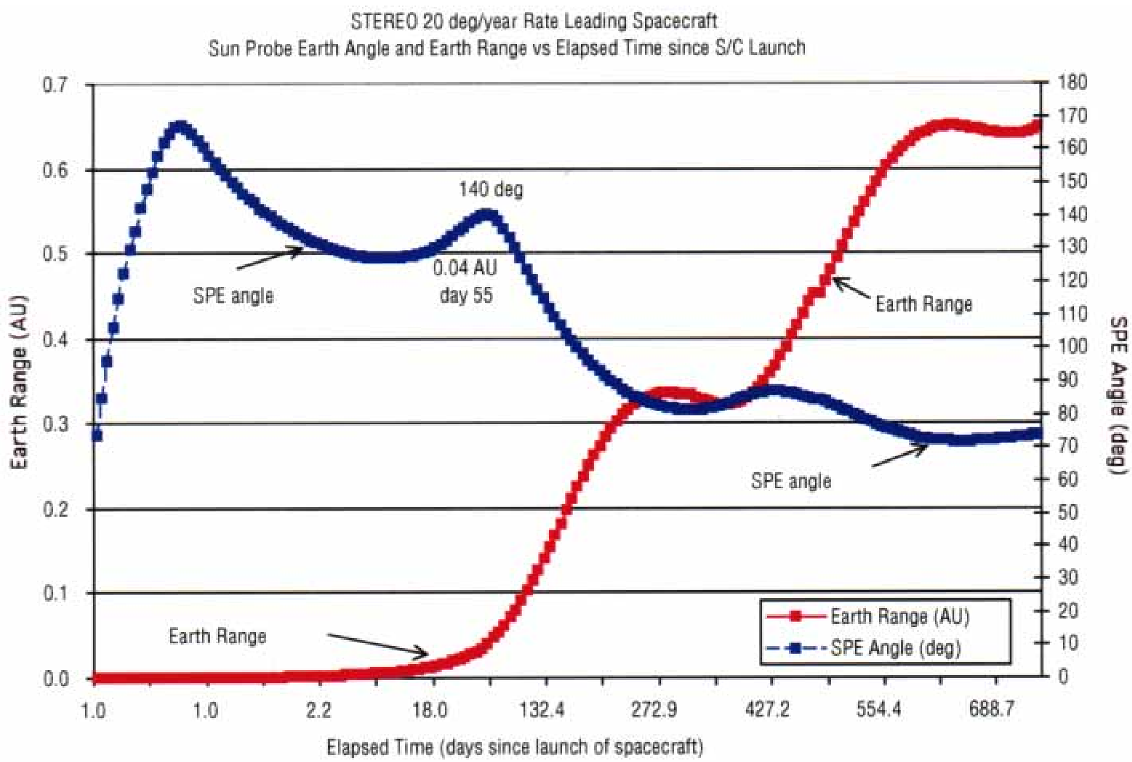


Figure 4-9 Leading Spacecraft Earth Range and SPE Angle vs. Elapsed Time

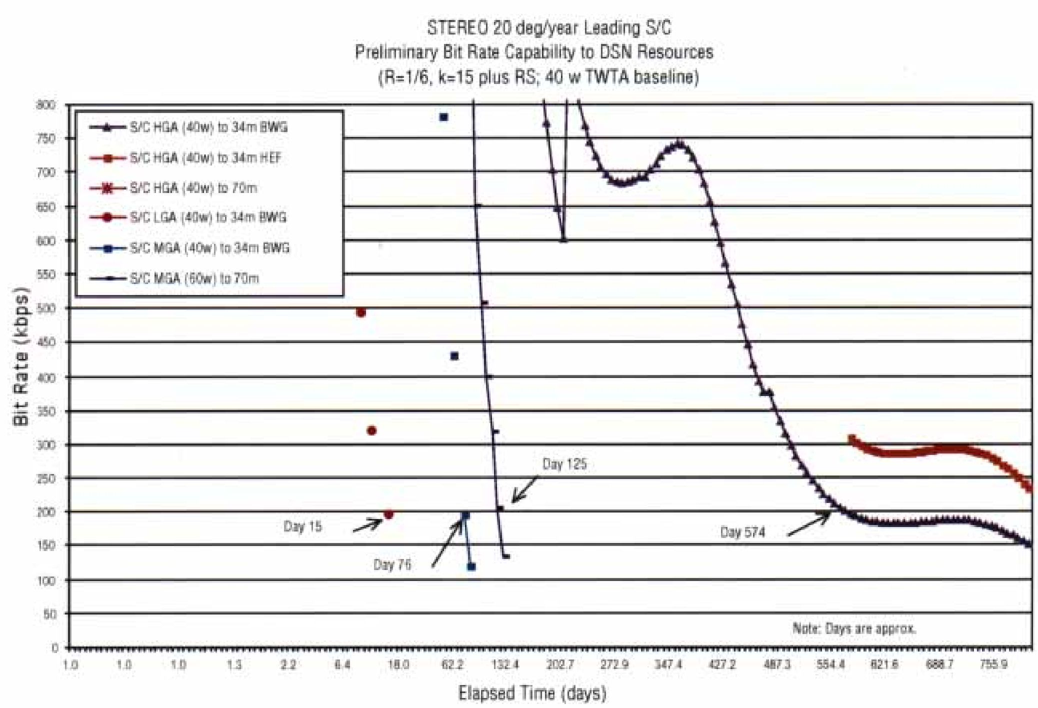


Figure 4-10 Bit Rate Performance for Leading Spacecraft

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	≤0.01	≥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	≥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	≥200
34m BWG	1.3m HGA	40	125-792	0.14-0.72	≥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

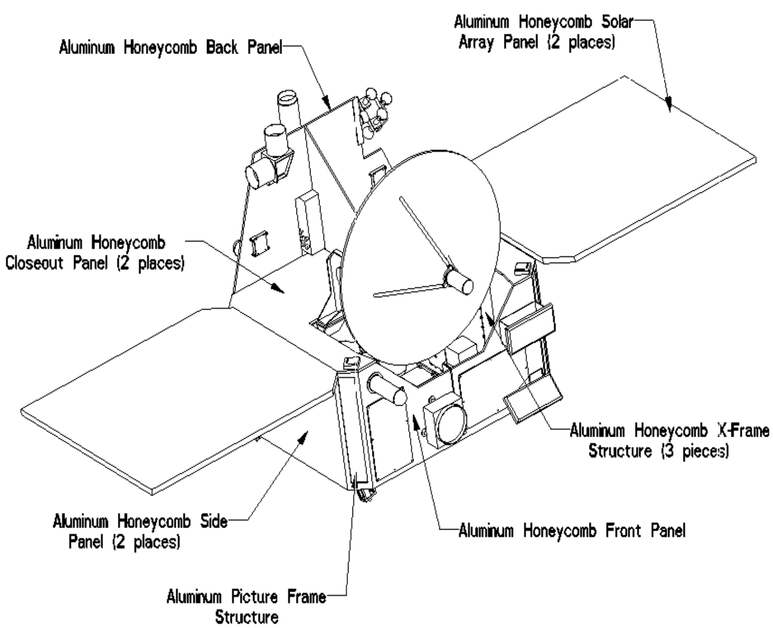


Figure 4-11 STEREO Spacecraft Structural Members

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)

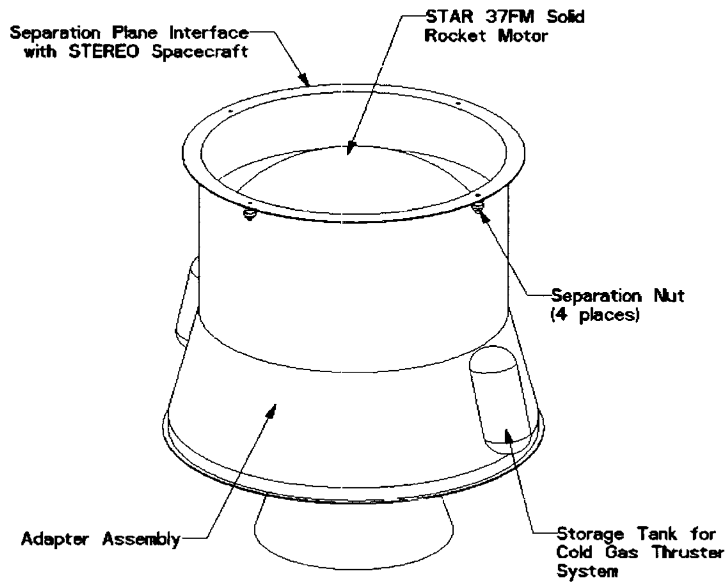


Figure 4-12 Orbit Injection Stage

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

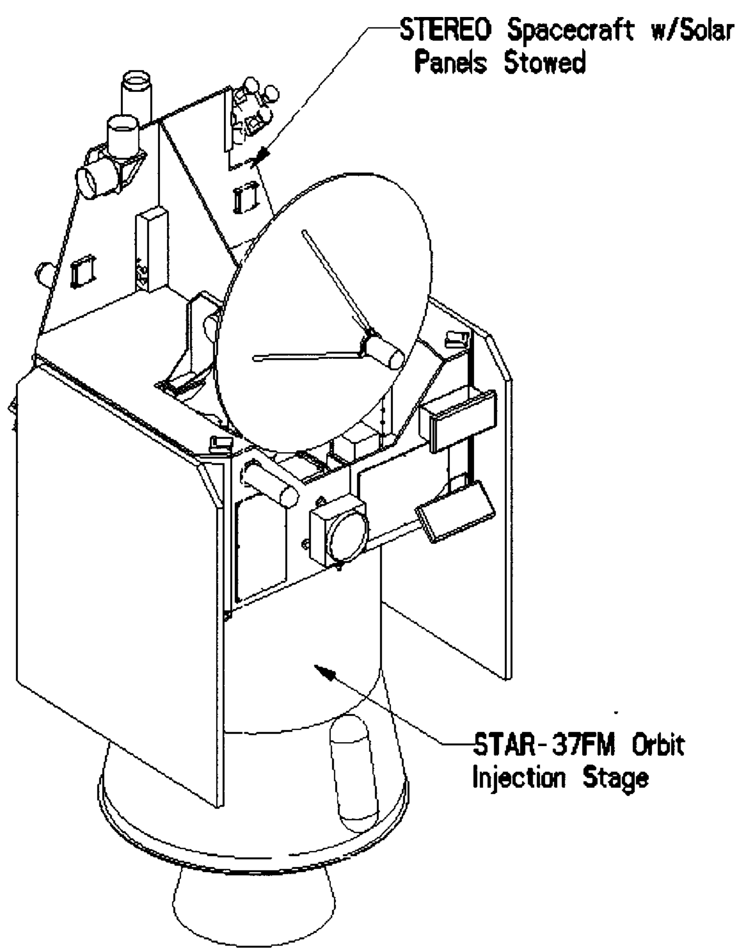


Figure 4-13 Launch Configuration

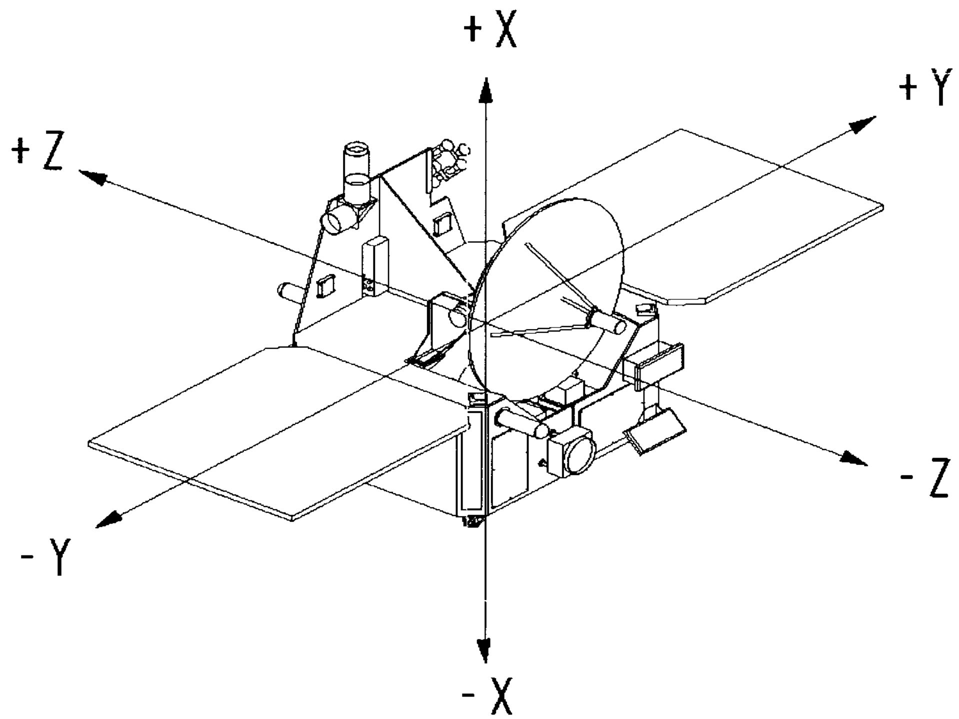


Figure 4-14 Spacecraft Axes

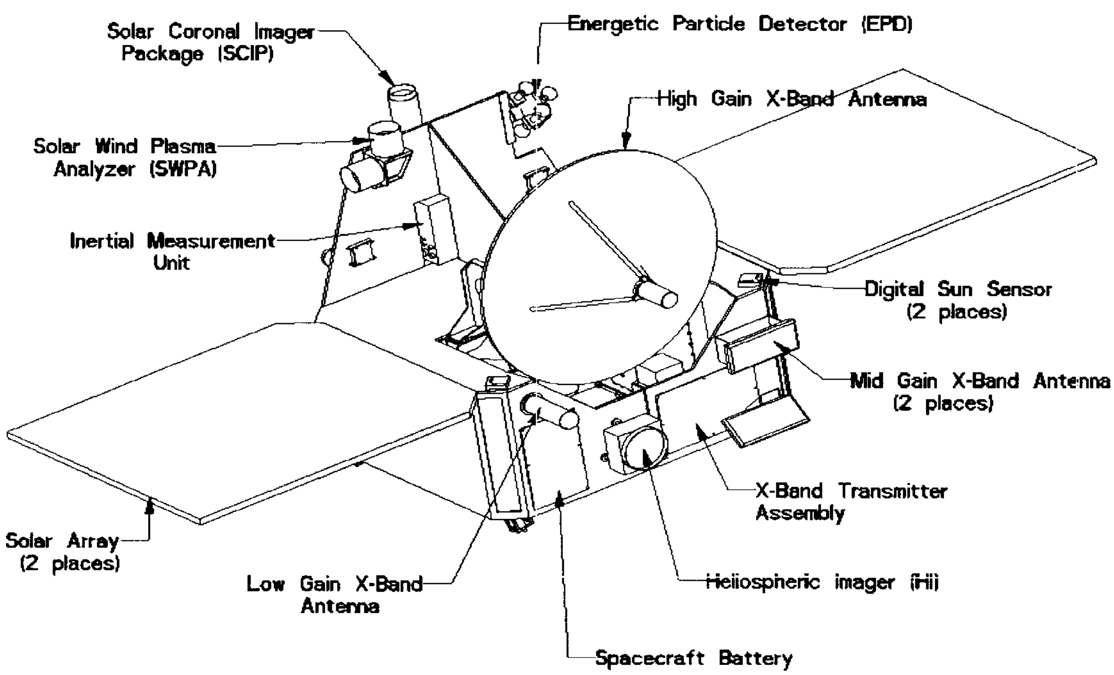


Figure 4-15

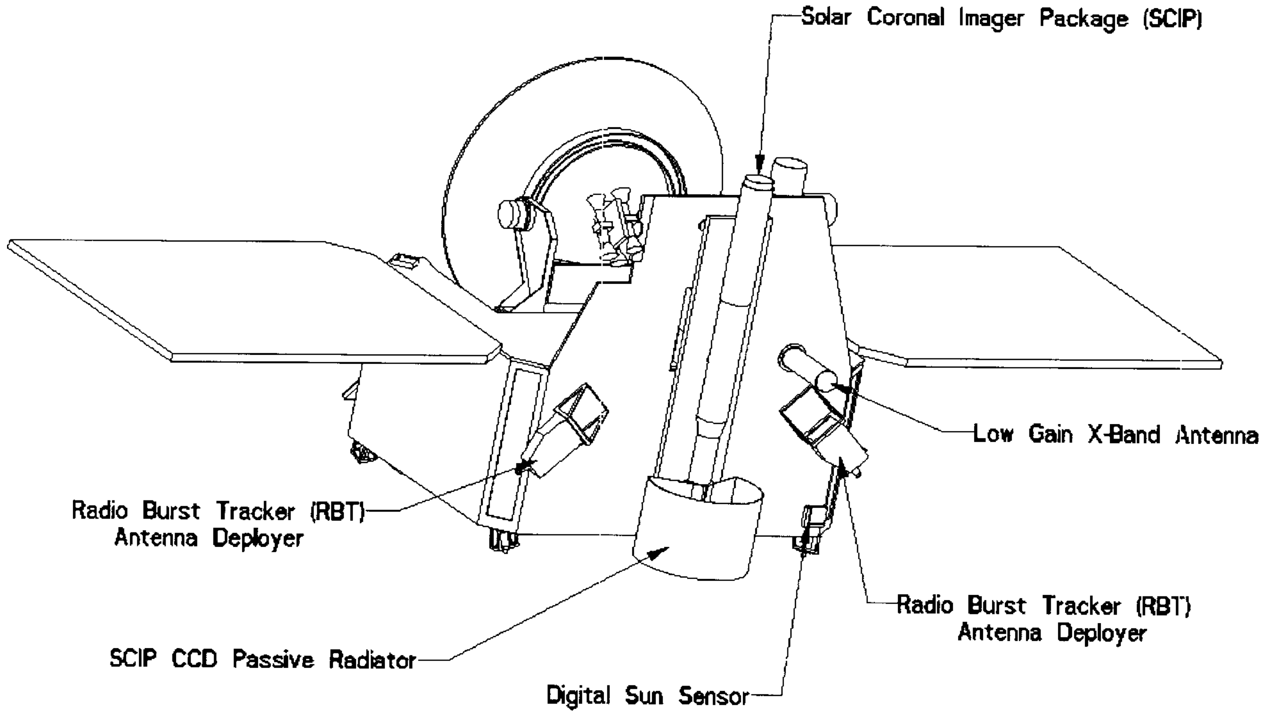


Figure 4-16 STEREO +Z Instruments

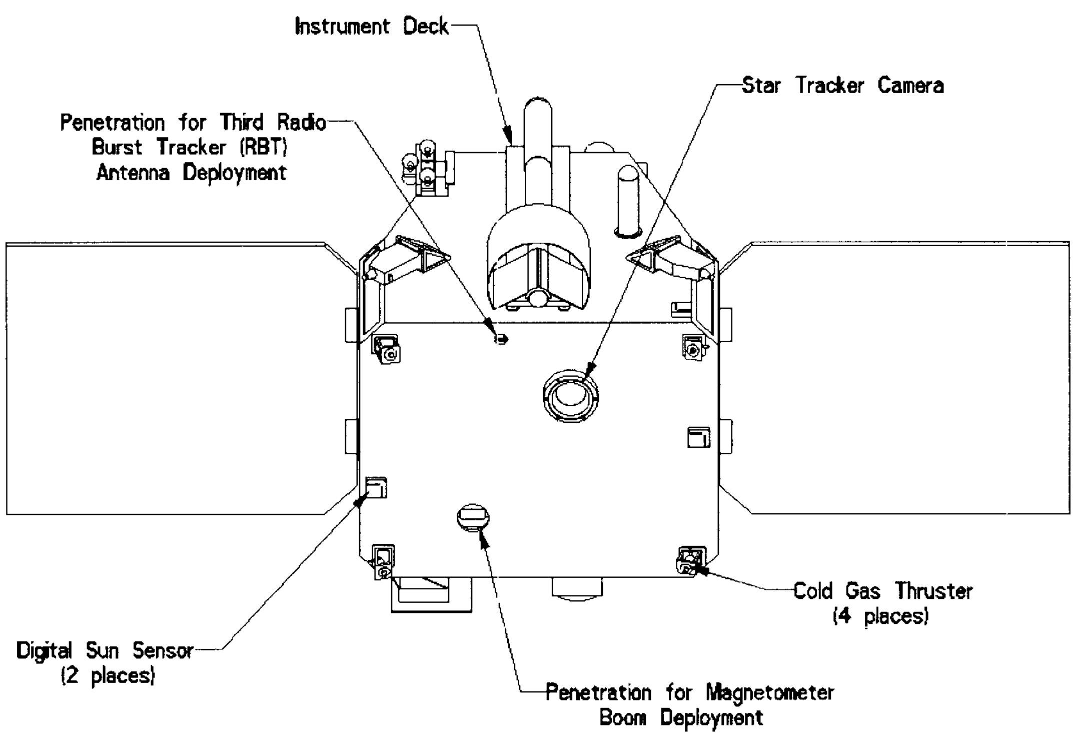


Figure 4-17 STEREO Spacecraft—X Axis

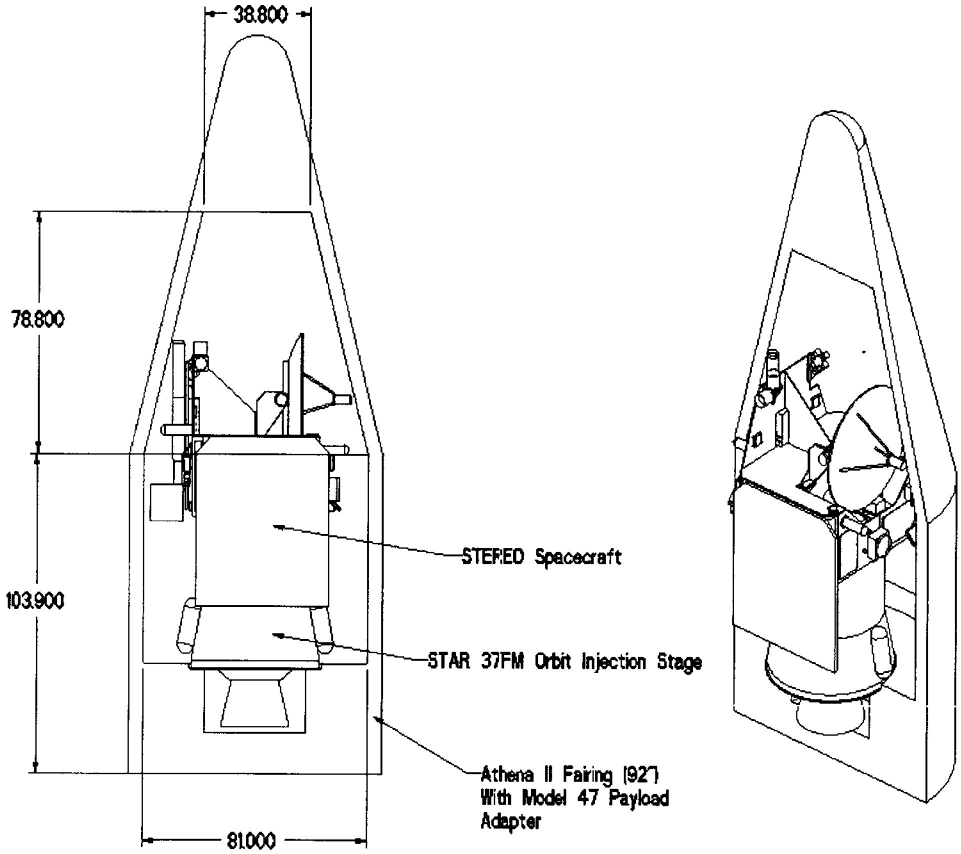


Figure 4-18 STEREO in Launch Shroud

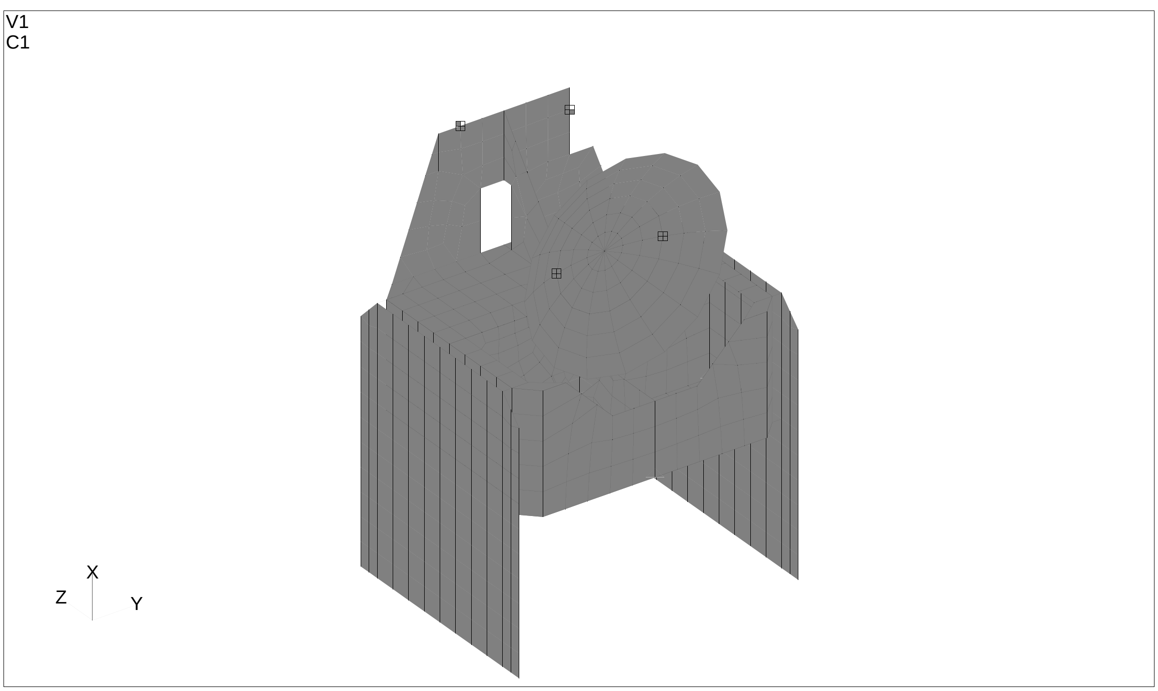


Figure 4-19 STEREO Finite Element Model (hidden lines removed)

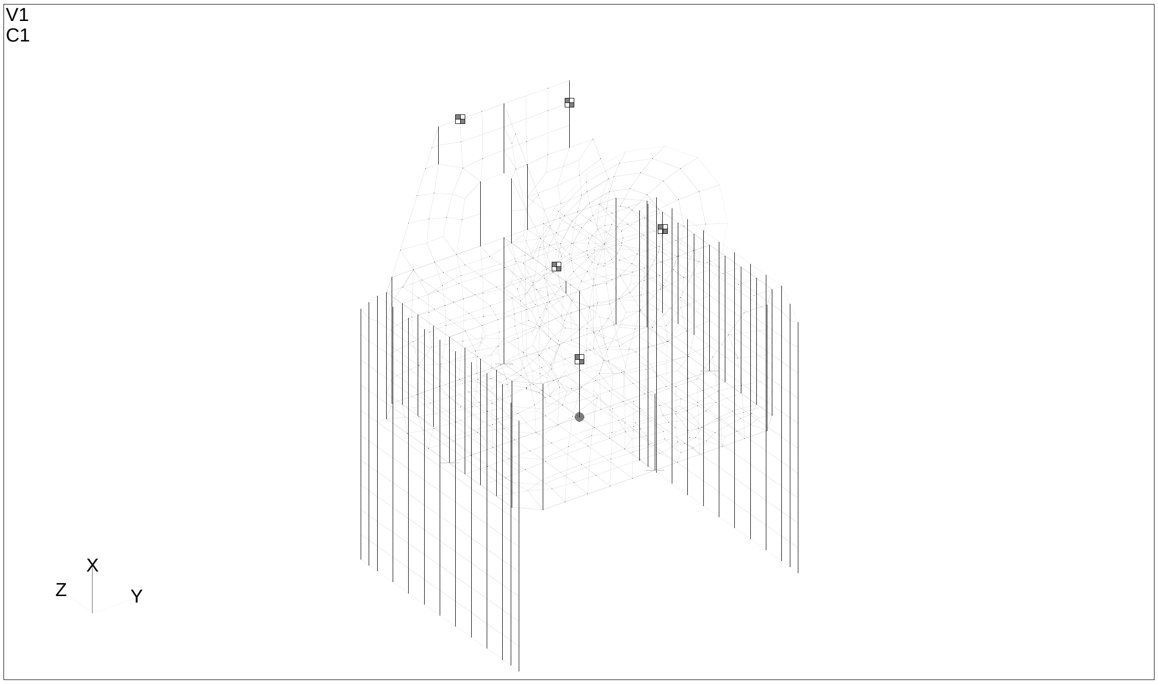


Figure 4-20 STEREO Finite Element Model

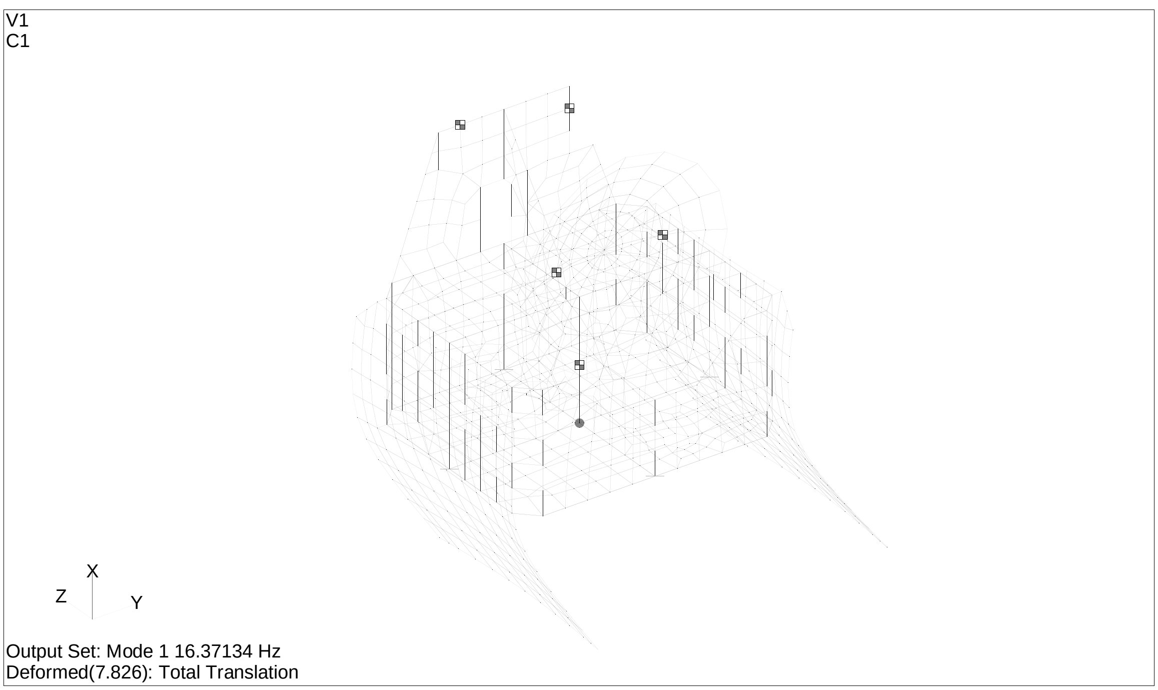


Figure 4-21 16.4 Hz, Solar Panel Flexure

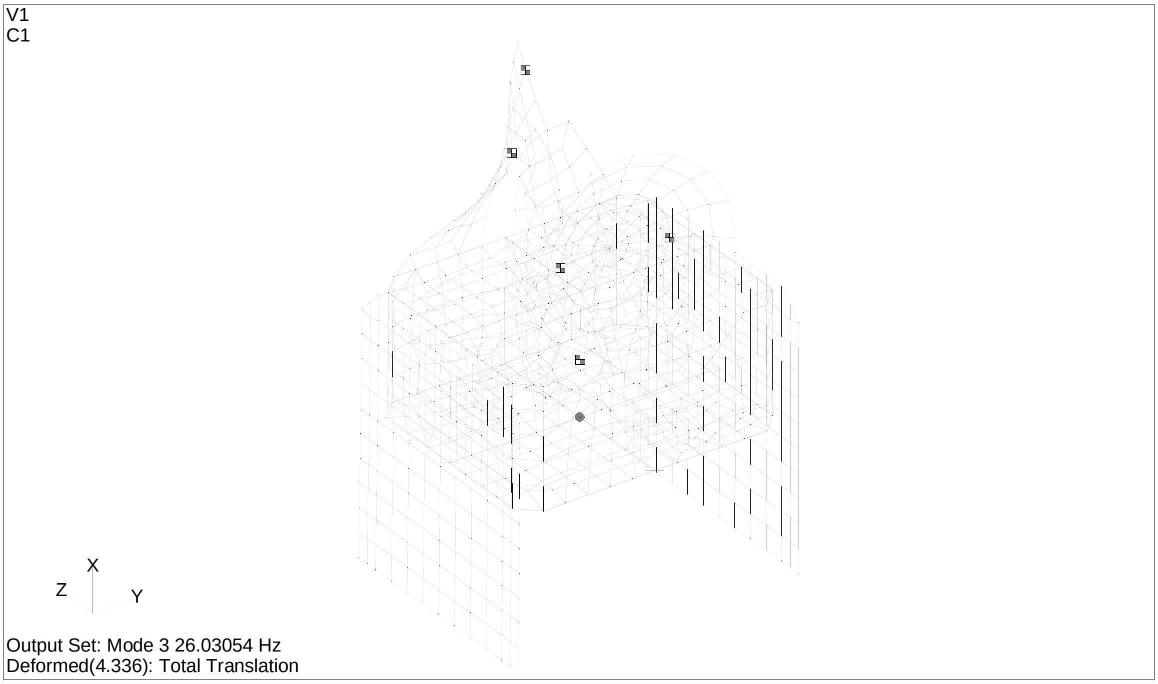


Figure 4-22 26.0 Hz, +Z Deck Flexure

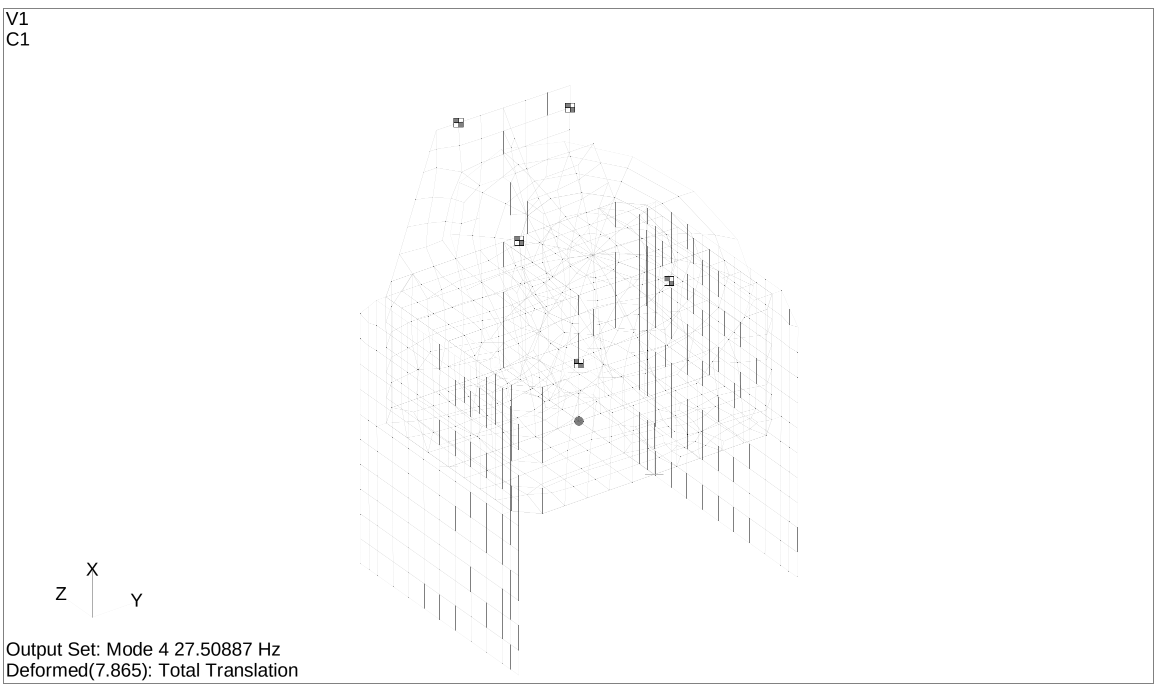


Figure 4-23 27.5 Hz, High Gain Antenna Support Flexure

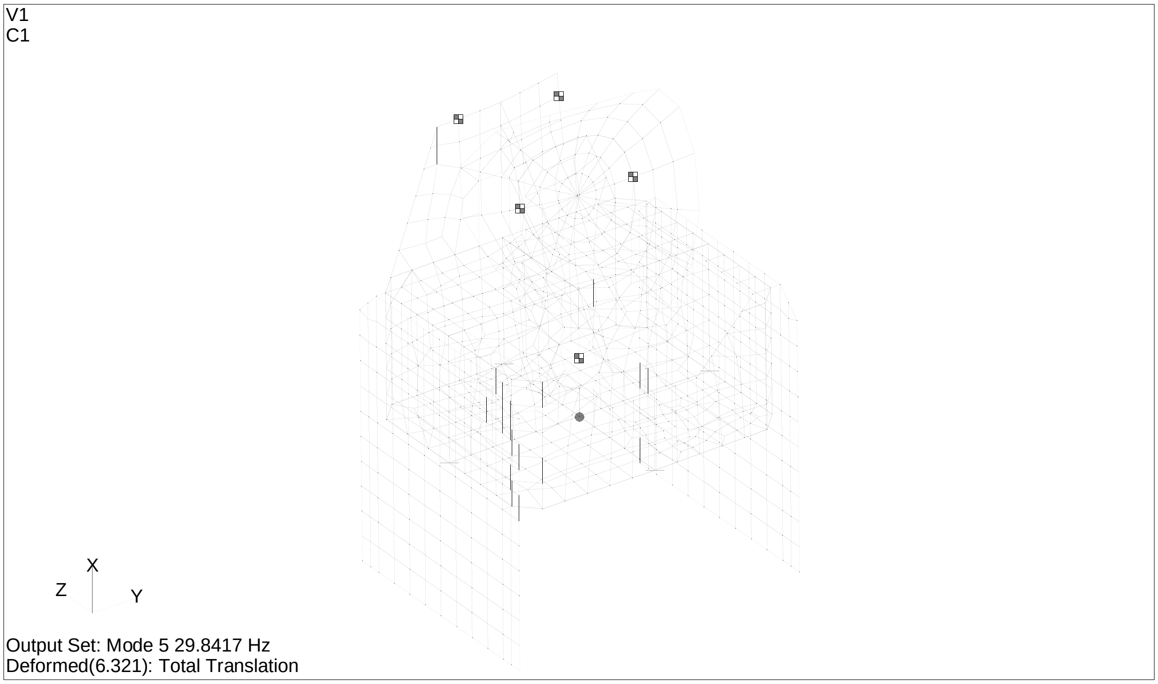


Figure 4-24 29.8 Hz, +Z Deck and High Gain Antenna Rotation about Y Axis

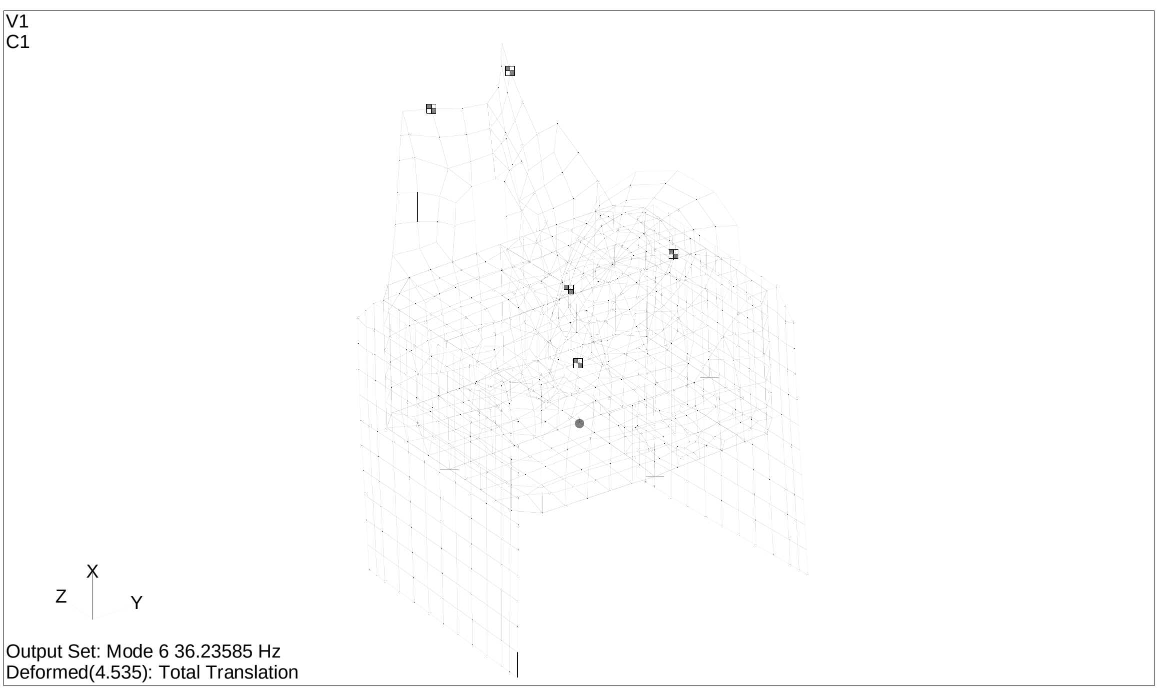


Figure 4-25 36.2 Hz, +Z Deck and High Gain Antenna Rotation about Y Axis

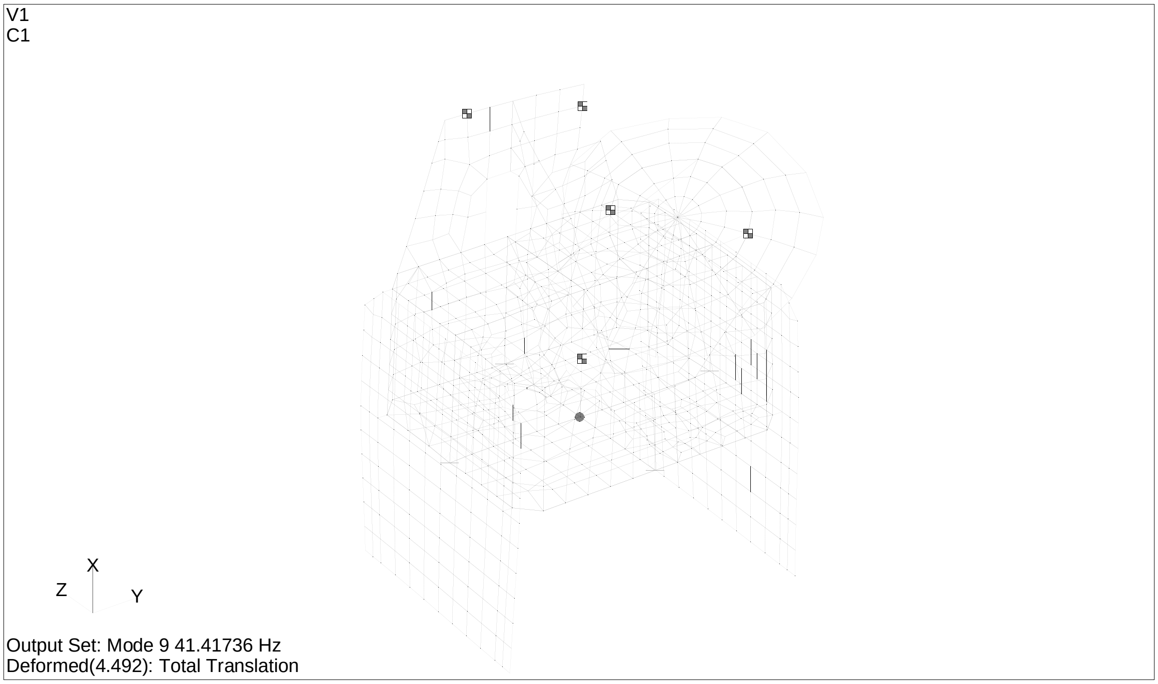


Figure 4-26 41.4 Hz,Spacecraft Minor Rotation about Z Axis

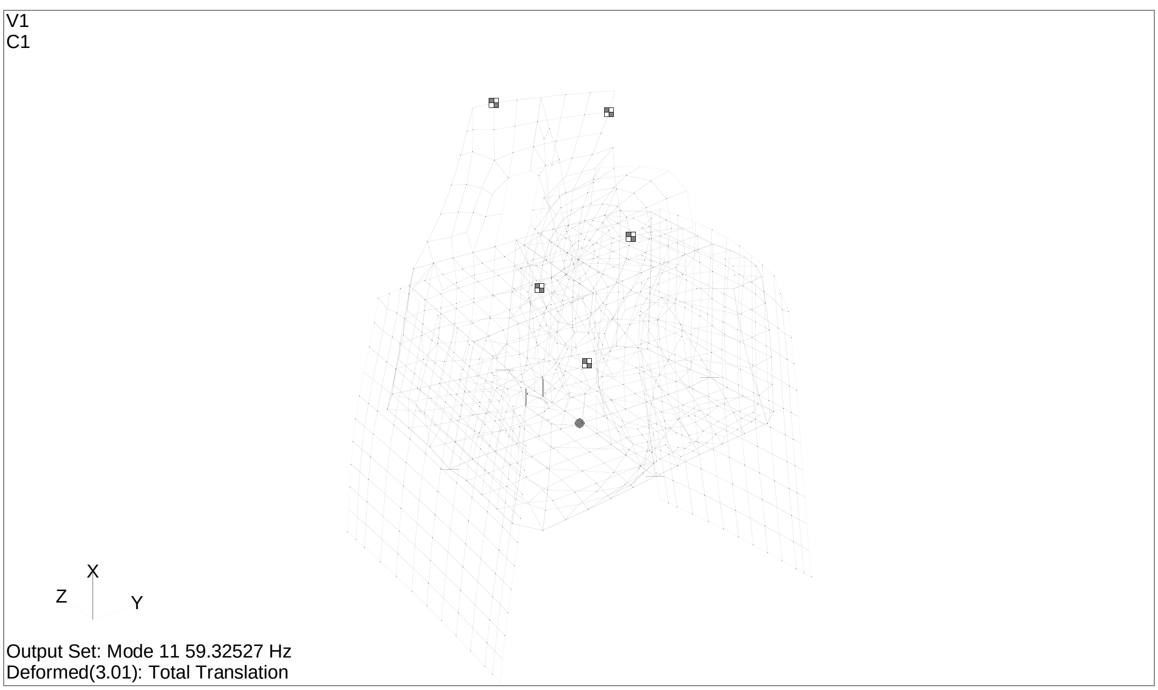


Figure 4-27 59.3 Hz, Spacecraft Racking (mostly –X deck flexure)

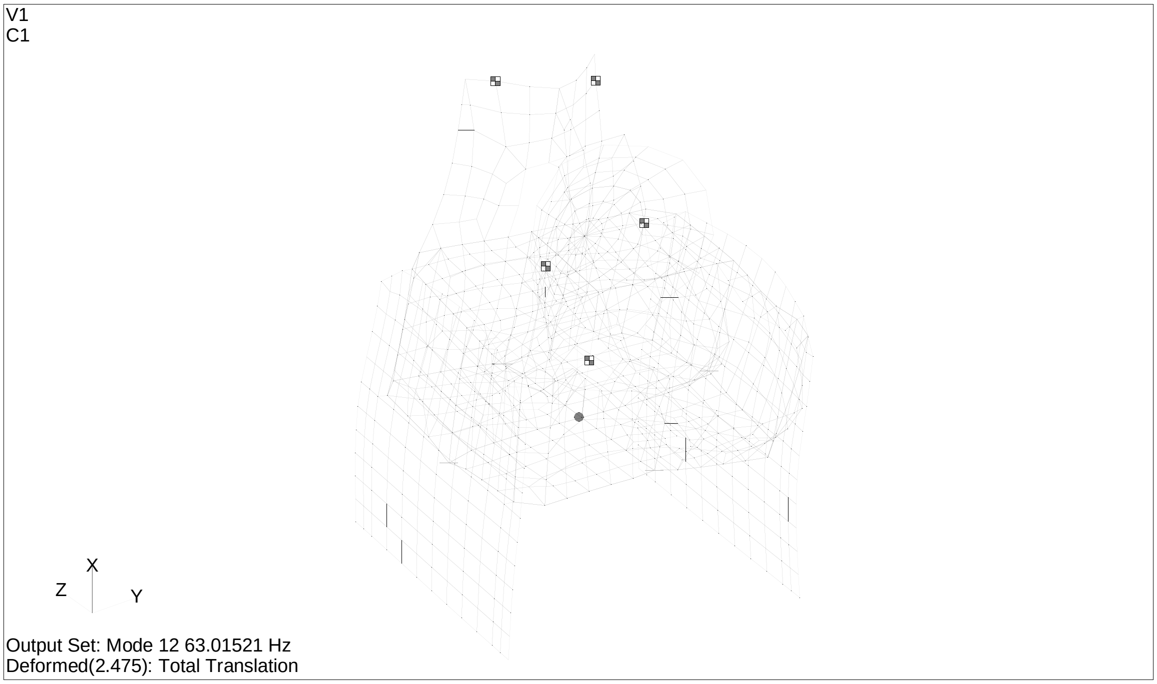


Figure 4-28 63.0 Hz, Major Spacecraft Rotation about Z Axis

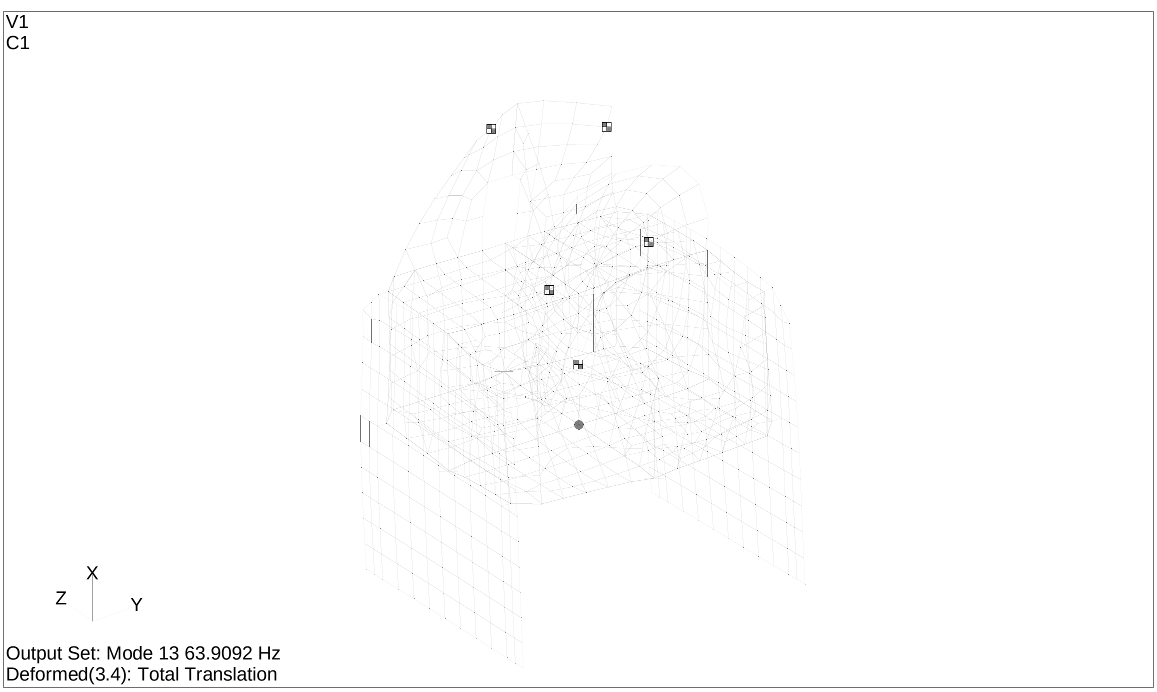


Figure 4-29 63.9 Hz, Spacecraft Rotation about Y Axis

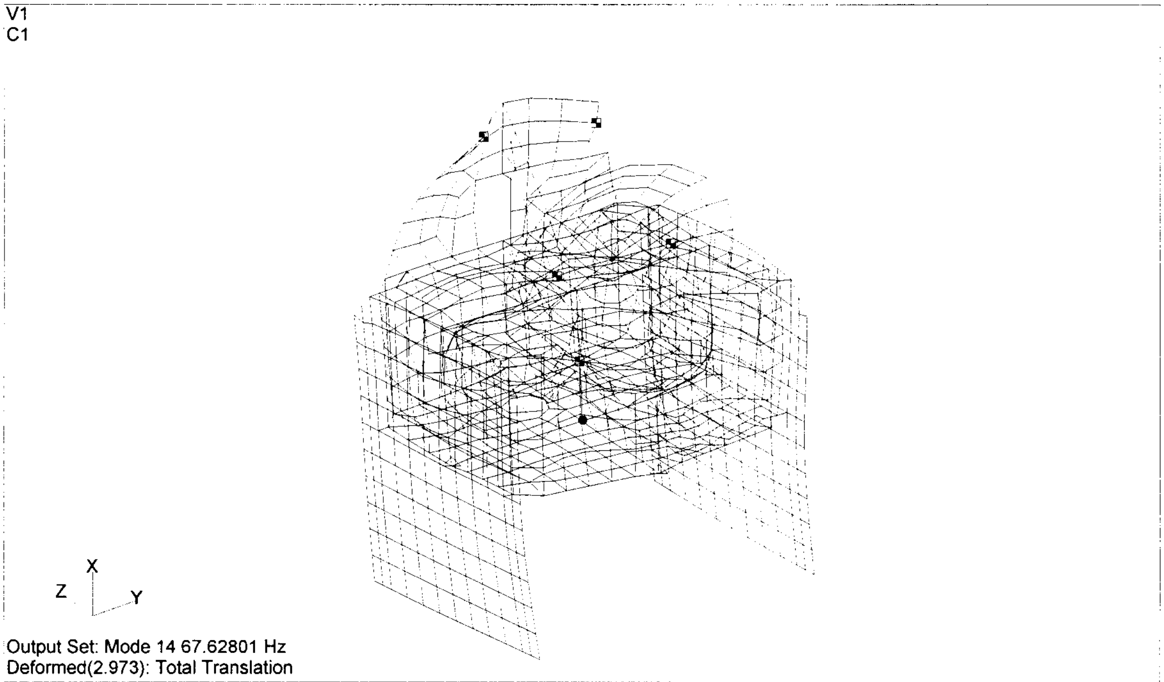


Figure 4-30 67.6 Hz, Spacecraft Rotation about Y Axis

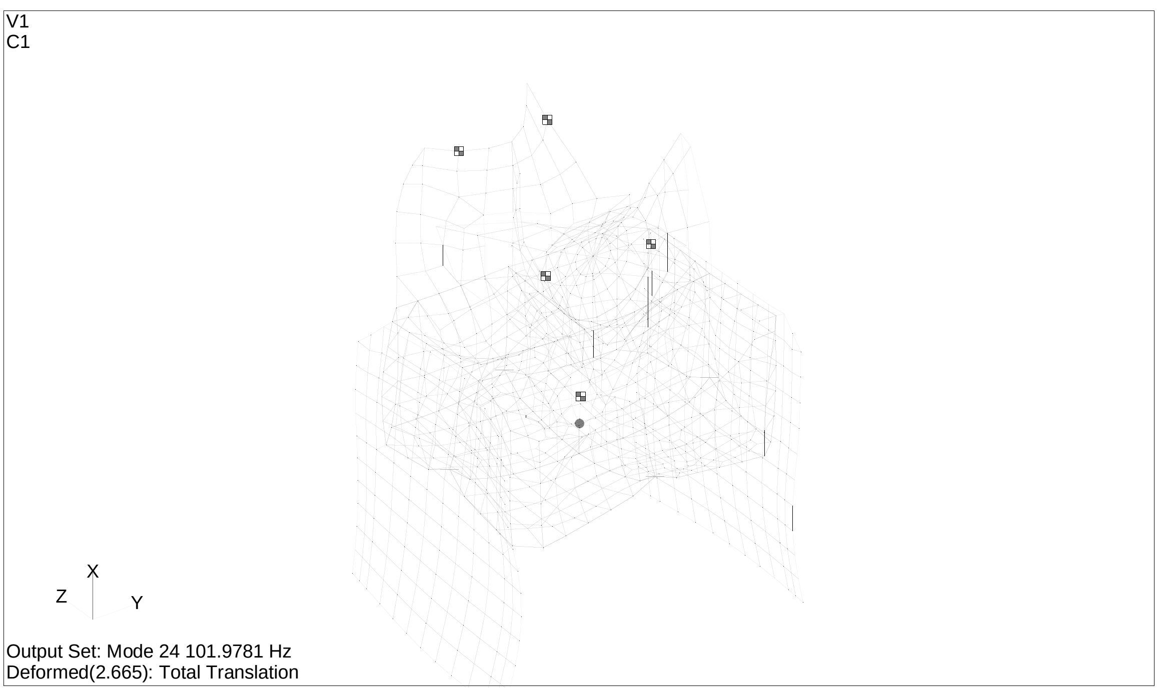


Figure 4-31 102.0 Hz, Spacecraft Thrust (X)

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 $\pm5^{o}$ 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.

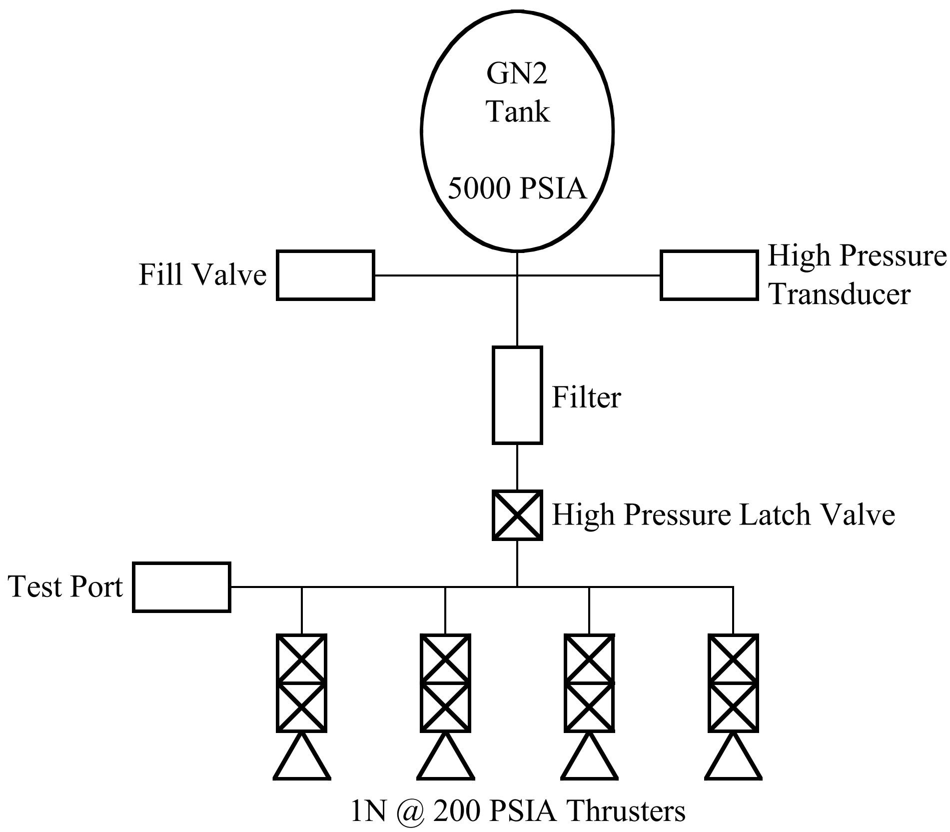


Figure 4-32 Propulsion System Schematic

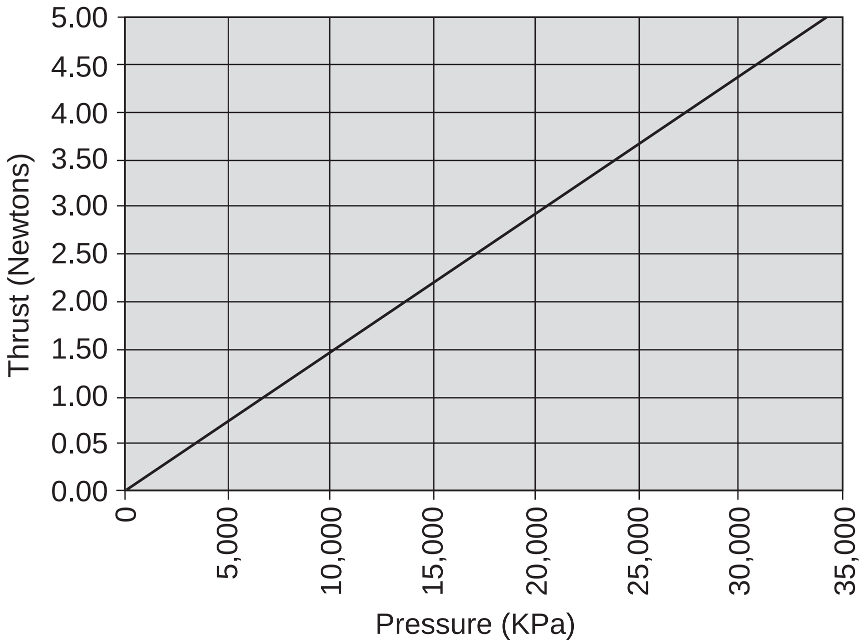
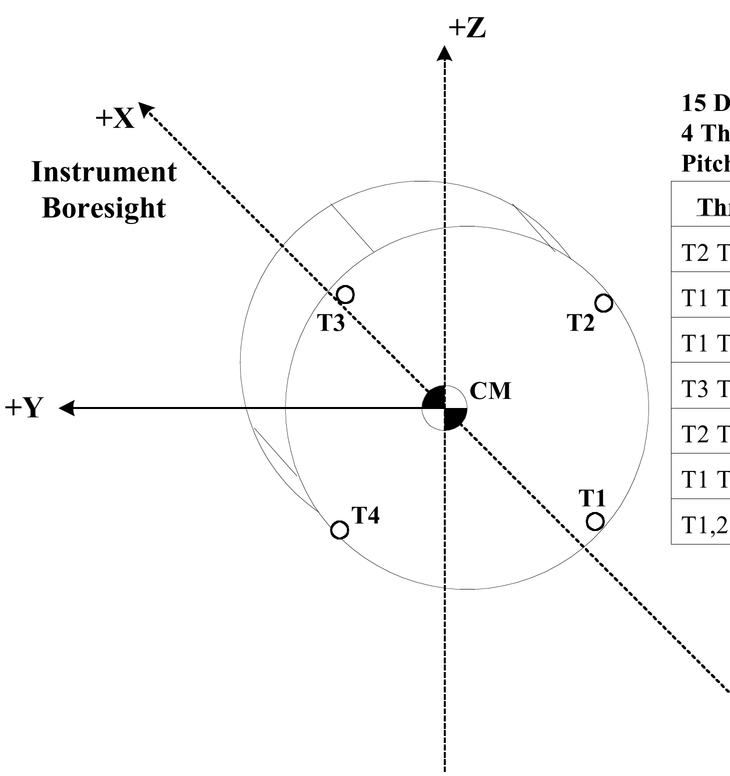


Figure 4-33 Thrust is Linear Over the (5,000–100 PSIA) Range



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