3.0 SYSTEM ENGINEERING

3.1 System Description

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

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

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

Table 3-1 Top-Level System Driving Requirements

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

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

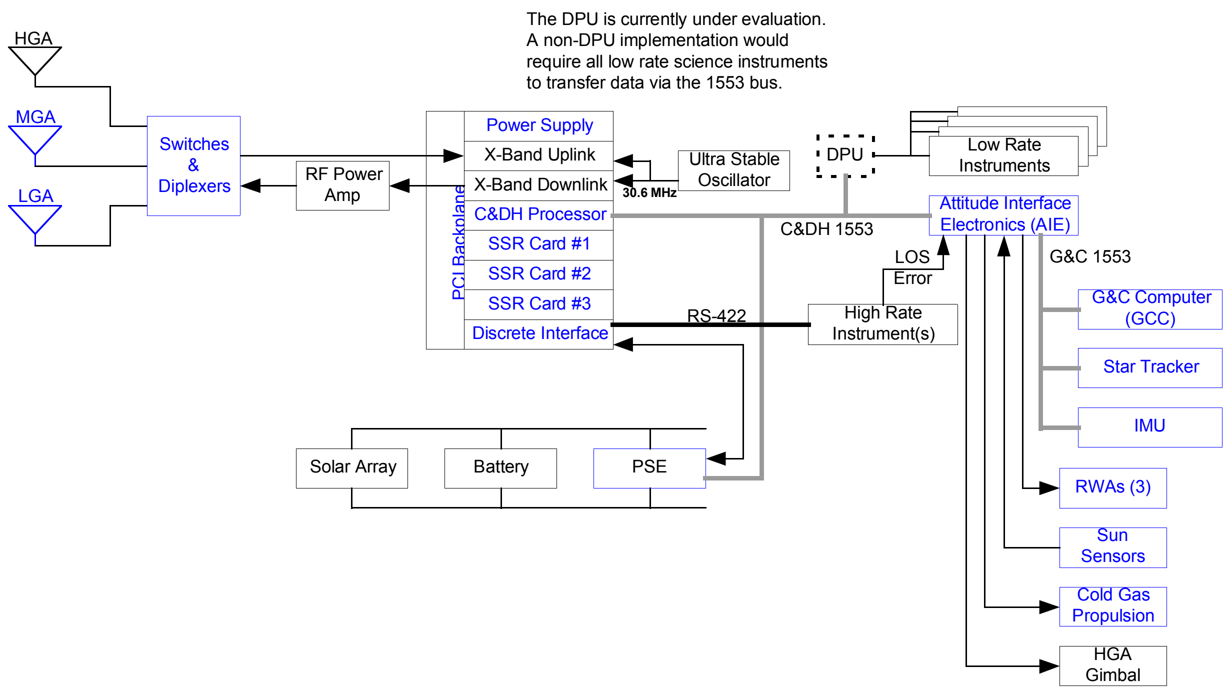


Figure 3-1 System Level Block Diagram

Table 3-2 STEREO Component Legacy

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

3.2 Spacecraft Fault Protection Architecture

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

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

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

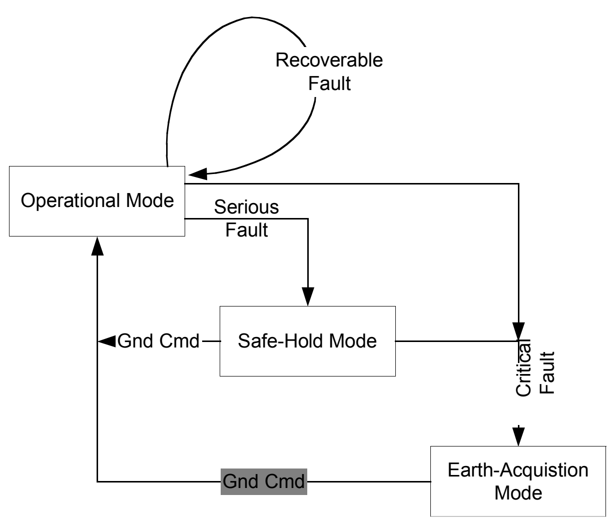


Figure 3-2 Mode Transition Diagram

Table 3-3 Spacecraft Mode Change Requirements and Mode Configurations

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

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

3.3 Top Level Spacecraft Descope Plan

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

Table 3-4 Descope Option Summary

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

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

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

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

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

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

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

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

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

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

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

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

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

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

3.4 Risk Identifcation

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

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

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

Lack of Instrument Definition. See Section 3.1.

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

Table 3-5 Risk Summary

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

Table 3-6 Failed Component Contingencies

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

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

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

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

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

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

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

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

The methodology has several benefits:

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

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

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

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

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

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

3.5 Inherent Spacecraft Redundancy

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

3.6 Selected Spacecraft Redundancy

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

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

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

Table 3-7 Risk vs. Reward for Selected Redundancy

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

Table 3-8 Sparing Philosophy

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

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

3.7 Sparing Philosophy

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

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

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

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

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

3.8 Spacecraft Mass and Power List

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

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

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

3.9 Technology Insertion Areas

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

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

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

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

Table 3-9 Mass and Power Budgets

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

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

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

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

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

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

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

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

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

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

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

3.10 Launch Vehicle

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

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

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

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

Table 3-10 Launch Vehicle Discriminator Summary

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

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

[1] Not Qualified
[2] Estimated

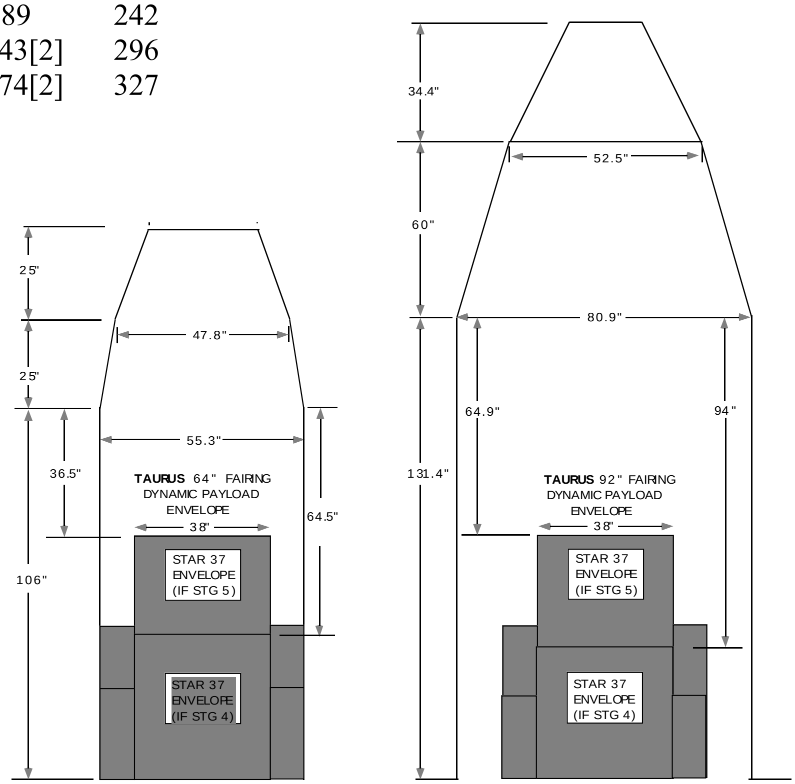


Figure 3-3 Taurus Launch Vehicle

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

[1] Not qualified
[2] Lunar Prospector configuration

92” Fairing

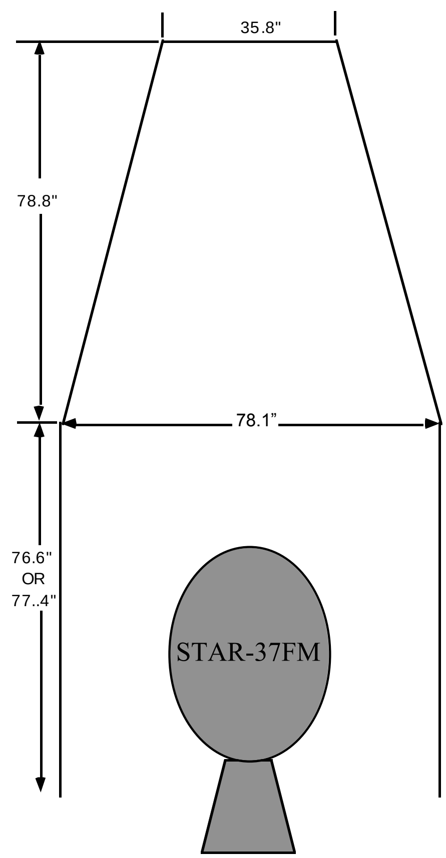


Figure 3-4 Athena II

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

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

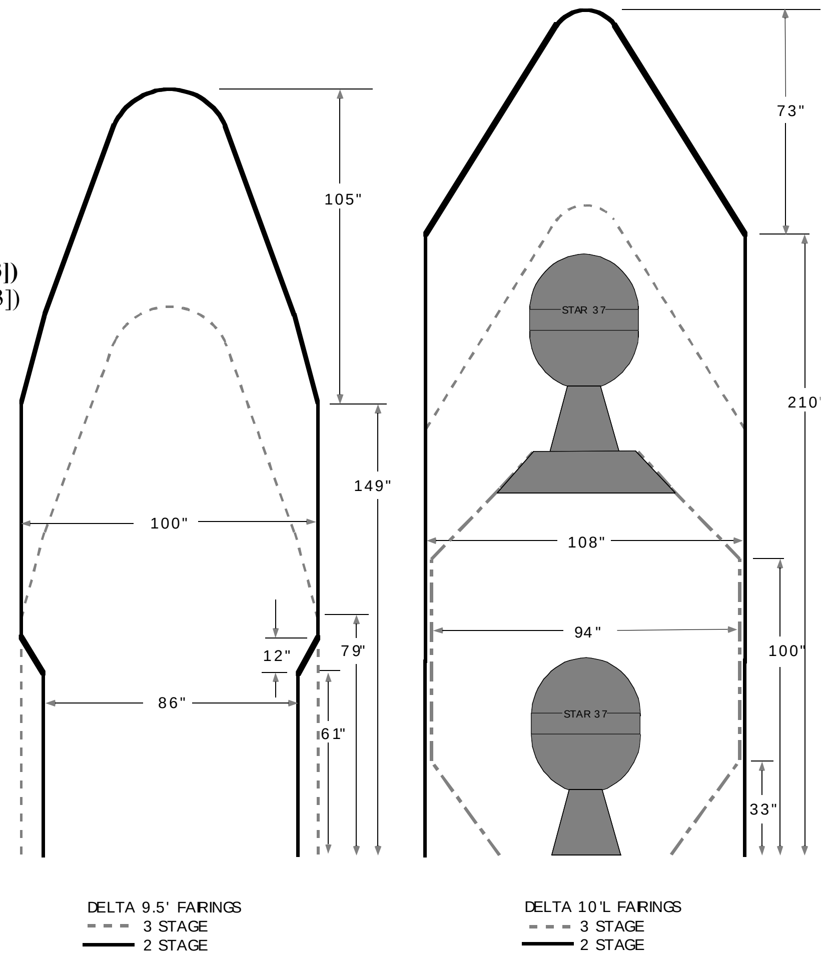
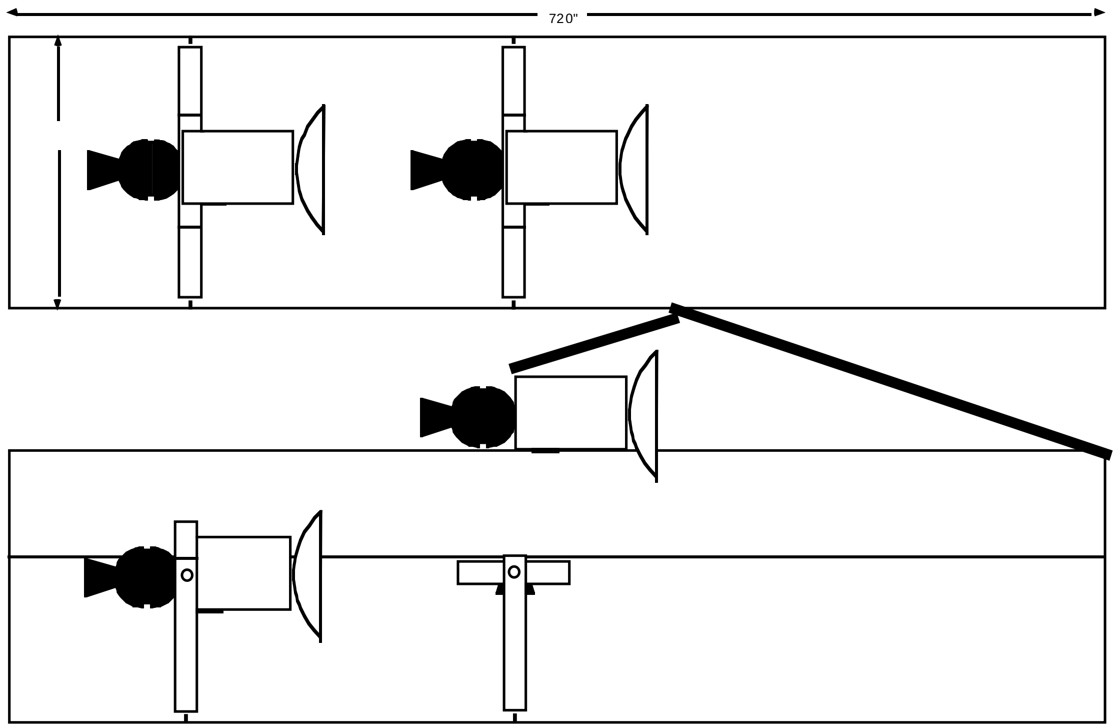


Figure 3-5 Delta II



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

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

Figure 3-6 Shuttle