ADVANCED PROJECTS DESIGN TEAM

STUDY NAME: SUPER NOVA-ACC. PROBE (SNAP)
STUDY LEADER: ANDY GERBER

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SNAP 2002-10 TEAM X ROSTER - MONDAY, OCTOBER 28, 2002			
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1.0 EXECUTIVE SUMMARY

1.1 MISSION OBJECTIVES

The Supernova/Acceleration Probe (SNAP) mission is intended to understand the nature and origin of the recently (1998) discovered "Dark Energy" acceleration of the expansion of the universe. This discovery-based experiment is designed to precisely measure the history of this acceleration from the current epoch back in time to about 10 billion years ago from a high galactic latitude region in both hemispheres. These objectives are too be met using a large (~2 m diameter) telescope in high earth orbit over a four year period, allowing the brightness of ~2000 type Ia supernovae and the red shift of their parent galaxies to be measured using imaging with filters and spectroscopy in the Visible Near InfraRed (VNIR) spectral region.

1.2 DESIGN DESCRIPTION

MISSION DESIGN

SNAP is an earth orbiting satellite that has a nominal mission duration of 5 years. The orbit used by SNAP has a period of 72 hours with an initial periapse altitude of 10000 km. For the Team X study, the customer desired a repeat groundtrack orbit such that there was one downlink pass over Berkeley per orbit, occurring at periapse. Therefore, an apoapse altitude was selected that would satisfy the repeat groundtrack requirement. The inclination and node of the orbit were selected so SNAP would orbit in the lunar plane. The initial orbital elements and epoch supplied by the customer and adjusted for a repeat groundtrack were:

Initial epoch : Feb. 28, 2009 00:30:00

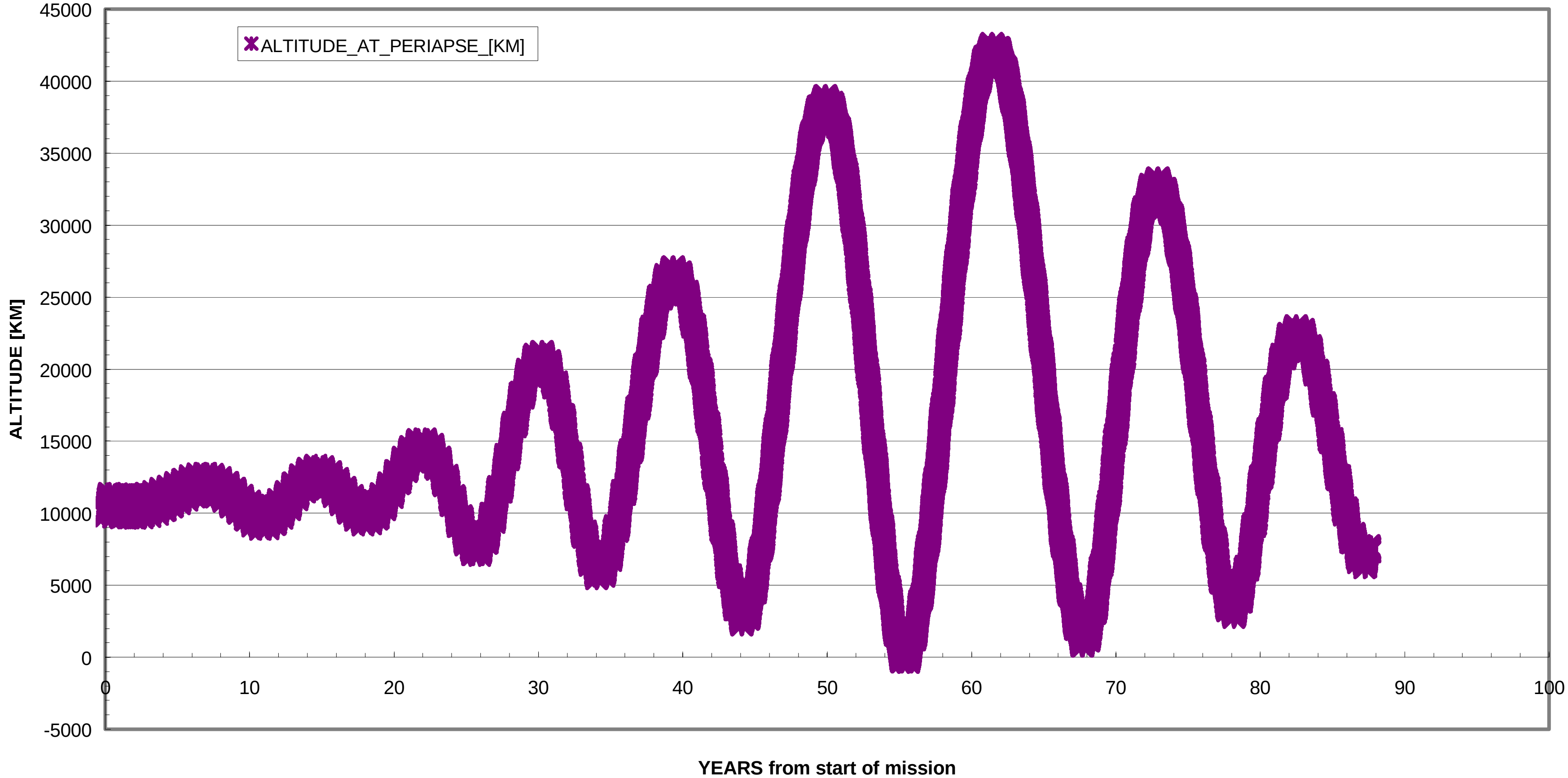
(semi-major axis [km], eccentricity , inclination [deg], longitude of the ascending node [deg], argument of periapse [deg], mean anomaly [deg]), Earth Centered Equator and Equinox of J2000: (87710.6, 0.8131, 26.3733, 349.938, 0.0, 0.0)

The orbital elements were input to a GRIST simulation and run out for 5 years with J2 (oblateness), lunar, and solar perturbations modeled. An SPK file was produced by GRIST and loaded into the SOAP visualization software by Joe Neelon. The SOAP plots indicated that in order to maintain the repeat groundtrack, the inclination would have to be adjusted. GRIST and SOAP indicated a drift in the inclination of –0.2 degrees per 30 days. Adjusting for an inclination of 0.2 degrees would require a delta-v of 3 m/s per maneuver. Assuming that a maneuver would have to be performed approximately every 30 days over a 5 year mission, 60 maneuvers would require an orbit maintenance budget of 180 m/s.

END OF LIFE/DISPOSAL

The decision was made to not dispose of the vehicle and to leave it in its final orbit. Analysis showed that the lowest delta-v required to dispose of the vehicle was approximately 250 m/s; about the same delta-v was required to lower the periapse into the ocean as was required to perform a lunar flyby sending the spacecraft into an orbit around the sun. A GRIST simulation including modeling J2, luni-solar perturbations, and solar radiation pressure was run out past the end of mission to analyze the stability of the final orbit. The GRIST results (shown below) show that the spacecraft will crash back to earth 50 years after the end of the mission.

SNAP ALTITUDE_AT_PERIAPSE_[KM]



Launch Vehicle Selection

The launch vehicle selected was the Delta 4040-12 with a capability to deliver 3040 kg to a C3 of –4.5. It was assumed that the launch vehicle would be equipped with an upper stage that could deliver the spacecraft to a final orbit periapse altitude of 10000 km and a final orbit apoapse altitude of 159000 km. The spacecraft will have to perform a maneuver to correct launch injection errors to place it in its final mission orbit. A delta-v of 50 m/s has been budgeted to clean up launch injection errors.

Mission Delta-V Budget

Maneuver	Delta-v (m/s)
Correct launch injection errors	50
Orbit Maintance	180
Total Delta-v required	230

Ground Systems and Mission Operations

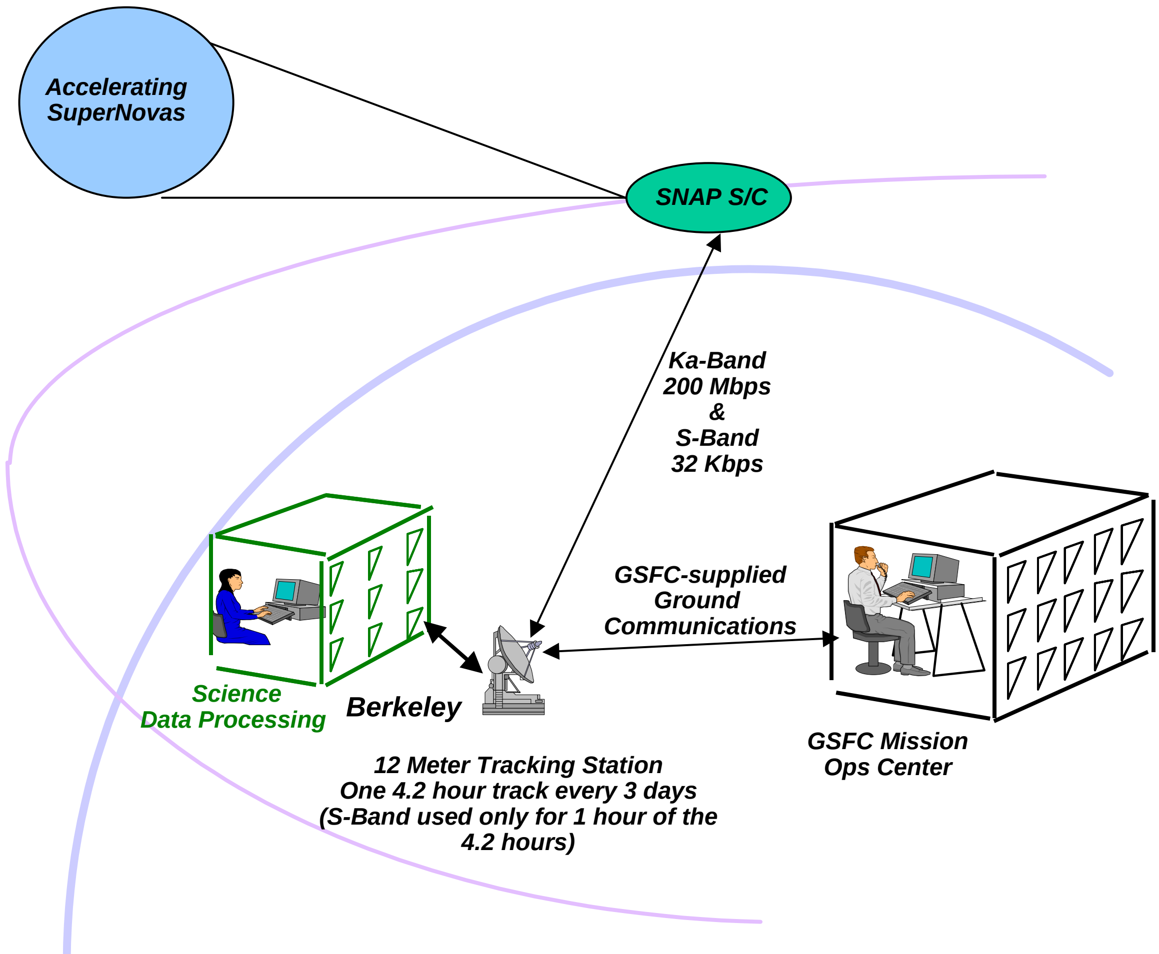
The major driving requirements on the Mission Operations System (MOS) and Ground System are described in this section.

- The MOS is responsible for all aspects of spacecraft mission operations. Specific functions provided by the MOS shall include:
    - mission planning
    - science planning
    - navigation analysis
    - sequence design and scheduling
    - command generation and transmission
    - telemetry/data receipt, storage, distribution, analysis & archiving
    - flight system health monitoring
    - fault diagnosis and correction.
- The MOS shall provide S-band uplink and Ka-band and S-Band downlink for communications with the spacecraft.
- The MOS shall be able to reliably operate and sequence spacecraft activities and science data capture/return over the mission duration.
- The MOS design should minimize operations complexity and cost. Where feasible, multi-mission operations capabilities should be utilized.
- The MOS shall be designed to reduce the risk of mission failure due to operational errors, and to facilitate the detection and resolution of spacecraft problems.
- The MOS shall accommodate degraded performance in preference to non-operating states, in the event of element failures.

DESIGN DESCRIPTION

The MOS concept for the SNAP mission is depicted below. The science data received on the Ka-band signal at the ground station at Berkeley is recorded and processed at Berkeley. The spacecraft engineering data received at S-band signal at the ground terminal at Berkeley and the radiometric data (i.e., ranging and Doppler data) is routed to Goddard Space Flight Center (GSFC) over a dedicated low-rate (approx. 56 kbps) communications link. Mission operations functions such as telemetry processing, mission monitoring, spacecraft performance assessment, navigation, and commanding are carried out by GSFC. Commands generated at GSFC are sent to the ground station over the same low-rate communications link.

THE SNAP MISSION OPERATIONS SYSTEM CONCEPT



TRADE CONSIDERATIONS

GSFC was selected as the mission operations center because of their experience with operations with science-driven earth orbiting missions. However, an alternative to GSFC would be to contract with an operations-providing company such as Universal Space Networks (USN) or Honeywell DataLynx. This alternative becomes particularly appealing if it is coupled with contracting with the same operations provider for leased tracking coverage or for the construction of a dedicated tracking station. See below for discussion of trades related to tracking coverage.

FLIGHT SYSTEM

ATTITUDE CONTROL SUBSYSTEM

Finely balanced and passively isolated reaction wheels would be used for fine pointing control during imaging. Thrusters would be used for reaction wheel momentum unloading, during initial deployment, for delta-V maneuvers, and during safe mode.

Attitude determination during initial deployment and safe mode would rely on coarse analog sun sensors and gyros. More accurate stellar inertial attitude determination using a precision star tracker and gyros would support high gain antenna pointing control. Ultra-high accuracy attitude determination would be achieved using fine guidance sensors and precision gyros. There would be a number of fine guidance sensors (FGS's) located in the telescope focal plane and capable of providing 3-sigma accuracy to within ± 0.01 arcsec per axis. A redundant set of hemispherical resonating gyros (HRG's) would provide a highly stable inertial reference with 3-sigma accuracy to within ± 0.003 arcsec over 1 second, per axis. A Kalman filter would be used to blend FGS and HRG measurements to reduce the effects of noise from each sensor.

A slow-steering mirror within the telescope would be used to dither the line of sight to enhance resolution. The proposed approach is to take four successive 300-second images with each image centered at one corner of a square box that is half a pixel (0.05 arcsec) on a side. The steering mirror assembly would make the small adjustments needed to point to the corners of the box, while the telescope boresight would remain pointed to within ± 0.03 arcsec (3 sigma) per axis over the 1200 seconds required to take four images.

POWER SUBSYSTEM

The power subsystem for the Super Nova Acc. Probe (SNAP) is required to operate for the mission duration of 4 years. The spacecraft will be place in a Sun-Synchronous High Earth Orbit (HEO) with an orbit period of 3 days. The longest spacecraft eclipse duration will be on the order of 3 hrs. The spacecraft power subsystem proposed to meet the needs of the SNAP spacecraft consists of: two Triple Junction (TJ) Rigid GaAs solar arrays with a rated conversion efficiency of 26.8%, A 12-cell 90 Ahr Ni-H_2 common pressure vessel (CPV) energy storage device, and power subsystem electronics. The components where selected on the basis of present availability (Technology cut off level of TRL 6) and flight heritage. The spacecraft subsystem components will now be discussed in detail.

The solar panels radiate on only one side and have to be below 70C. This requires 40% additional radiator area on the solar cell side, covered with silver teflon. The focal plane is cooled passively by a radiator facing cold space with conductive straps of graphite removing heat from the focal plane.

The heat rejection from the spacecraft varies from 200W to 580W. This requires a fixed radiating surface of 1m2 and 5 standard 16 blade louvers of area equal to ¼ m2, to allow for heat rejections above 200W. They are mounted on the side that is antisun. The fixed radiating area is a part of spacecraft area with a treated surface.

Heaters, multilayer insulation, tapes and films are used to provide thermal control of the space craft to keep the temperatures within flight allowable limits.

TELECOMM SUBSYSTEM

The imaging and spectroscopy requirements of the SNAP mission place significant demands on the overall telecom implementation. Because of the high data return required and orbital geometries involved, Ka-band was chosen as the operational frequency; however, since little commercial Ka-band equipment exists to meet the mission's particular needs, significant investment will have to be made to develop the appropriate capability both on the spacecraft and on the ground. The spacecraft end will require development of a high rate Ka-band telemetry transmitter and the ground will require implementation of a reasonably sized Ka-band antenna dish tied to a high rate telemetry receiver, demodulator, and decoder. Since much of the science investigation is being conducted by Lawrence Berkeley Labs and UC Berkeley, which currently operate a 10 m S-band ground station, it makes sense to build the new Ka-band ground station at the same location. This way more routine S-band communications (command, engineering telemetry, and ranging) can be conducted simultaneously with the Ka-band science return, though other existing S-band ground stations can be used for things like safemode recovery.

1.3 PROGRAMMATICS

Programmatically this mission is 3 projects: A technology development project for the detectors, a ground station upgrade project, and a science mission.

The detectors, focal plane and the ACS are challenging technical items for this project. In particular, the detector work is shown as having started a year ago, yet the detectors are not expected to be at TRL 6 by the technology cut-off date.

2.0 MISSION REQUIREMENTS

2.1 SCIENCE REQUIREMENTS

SCIENCE REQUIREMENTS

1. Perform Wide Field Imaging of the celestial poles from 0.35 to 1.7 microns with I-band (~0.8 micrometer) diffraction-limited optics.
2. Perform multiple-filter Wide Band Imaging photometry in multiple filter bands to determine supernovae color.
3. Detect supernovae at redshifts in the range of 0.3<Z<1.7.
4. Derive supernova peak luminosity to 2% (statistical) or better through multiple measurements over the supernovae light-curve.
5. Obtain supernova spectrographic observations near peak intensity with a resolution of 100 (lambda over delta lambda) over the spectral range 0.35 to 1.7 micrometers.
6. Measure supernovae host galaxy redshifts.
7. Identify and analyze over 2000 Type Ia supernovae.

MEASUREMENT OBJECTIVES

Detailed Level 1 & 2 Measurement Objectives (From Draft 1.0d "Mission Definition and Requirements Document", LBL, Dated 25 June 2002):

Obtain over 2000 classified Type Ia supernovae for analysis in the redshift range 0.3<Z<1.7

1.1.1 The observatory is to utilize a 2 m diameter telescope with I-band diffraction limited optics sensitive from 0.35 to 1.7 micrometers

1.1.2 The instrumented field of view of the telescope is to be approximately one degree (~0.7 square degrees)

1.1.3 Wide field imaging of region of low dust extinction near the ecliptic poles is to be conducted with a solar avoidance angle of 70 degrees

1.1.4 The photometric observations are to be zodical light limited

1.2 Derive supernova color and relative peak luminosity on average to 2% (statistical)

1.2.1 Obtain photometric measurement in redshifted B-band broadband filters

1.2.2 Obtain peak and off-peak (plus and minus 4 days rest-frame) multi-band photometric measurements of supernovae with an S/N of 30 or better

1.2.3 Measure rise time with detection at average 2 day after explosion at 3.8 magnitudes below the peak with S/N>3, and peak-to-tail ratio of

1.2.4 Obtain peak and off-peak multi-band photometric measurement with at least 10 points of the supernovae lightcurve

1.2.5 Measure supernova color and extinction with up to six visible light and three infrared broadband filters from 0.35 to 1.7 micrometers

1.3 Obtain supernova spectrographic observation near peak intensity with a resolution ~100 over the 0.35 to 1.7 micrometer wavelength

1.3.1 Measure supernova peak spectrum to identify and classify supernova

1.3.2 Measure supernova spectra versus epoch for a subset of supernovae with Z<0.7

1.3.3 Measure the broad (200A) Silicon (6150A rest-frame) and Sulfur (5350A rest-frame features

1.3.4 Obtain spectroscopic measurement of calibration standard stars

1.4 System is to be capable of performing deep multi-color photometric surveys with field sizes of approximately 15 and 300 square degrees

1.4.1 Mission operations and avoidance angles should permit wide field surveys up to 300 square degrees

1.4.2 A minimum of four visible broadband filters are to be available for photo-z measurements to facilitate weak-lensing surveys

INSTRUMENTS (COMPLEXITY)

1. VNIR Optical Photometry: Two meter primary, 0.34 square degree field-of-view, 441 million V pixels, 144 million NIR pixels, 350-1700 nm spectral range with 6 V and 3 NIR filters and a detector array temperature of 140 K. (Complex)
2. VNIR Spectrograph: Two meter primary, 3 X 3 arcsecond field-of-view, 350-1700 spectral range with a spectral resolution lambda/delta lamba of 100 with a detector array temperature of 140 K. (Complex)

SCIENCE MISSION PLAN

Following launch, checkout and commissioning of the observatory, the observation phase begins with the objective of detecting and measuring about 2000 type Ia supernovae in four years. The areas to be searched consist of one small (15-300 square degrees, Note: Searching 300 square degrees would support a weak lensing survey (objective 1.4 above) and could be accomplished within the 4 years of Phase E operations) region at high galactic latitude in both hemispheres. As there are a total of 9 filters, each of which cover separate detectors, a complete observation of one area involves 9 separate pointings. This process is repeated until the entire area to be surveyed in one hemisphere is covered and then it begins again. As supernovae are discovered, the spectrometer is used to provide detailed spectral information on the explosion and to provide spectral information from which the red shift of the supernova's host galaxy is determined. Photometric observations are made periodically following first detection so after a short time the observations strategy involves survey work to detect new supernovae, monitoring of supernovae already detected and spectroscopy of detected supernova and their host galaxies. These observations occur over a large portion of the orbit where particle induced noise in the detectors is not a problem and the nearer earth portion of the orbit is used to return the data from that orbit to earth.

It is estimated that a total of 183 terabits of data will be returned by this mission over a four year period.

2.2 Instrument Requirements

Subsystem-Level Requirements

Customer supplied payload. Two meter aperture three mirror anastigmat which will maintain the main mirror elements close to room temp in orbit. This allows easier test conditions for pre-launch testing activities. The focal plane assemblies are required to run at 140 K.

Design Description

The development of the detectors and the design and implementation of the clusters of detectors in the focal plane are the main design drivers. We looked at the focal plane thermal design and our analysis follows

SNAP Instrument Focal Plane Thermal Design Evaluation

The SNAP Focal Plane is a 0.7m hexagonal block containing optical detector components. It is apparently contained in a dewar to isolate it as much as possible from the warmer S/C environment. The documentation includes a phantom drawing from which the dimensions of the major components may be determined. There are six rectangular cross-section bars of varying length joining the points of the FPA hexagon to a 2.0m2 radiator. A thermal math model (TMM) was created representing these parts. The significant characteristics are as follows:

- FPA dissipation is 9.3W Peak (orbital average is 4.6W).
- Heat path bars are assumed to be solid aluminum with a section area of 0.006m2 and average length of about 1.0m. [They could be heat pipes for better thermal conductance.]
- Radiator area is 2.0m2, emittance is 0.86, view to space is unobstructed.It was assumed to be 2024 aluminum for thermal conductivity.

What is not known:

- Radiator panel thickness
- Radiant heat on the FPA
- Parasitic heat leaks from the dewar to the FPA
- MLI characteristics for the back of the radiator panel.

These latter items were treated as parameters. The model comprised seven FPAnodes and a 32 node grid for the radiator. The FPS was divided into one central node to which heat was applied and six circumferential nodes which connect to the radiator. A high thermal conductance was assumed between the central and outer FPA nodes. The variations were radiator thickness from 1mm to 5mm, MLI effective emissivity of .01 to .03 with a 20C outer layer and added FPA heat of 0W, 2W, 4W and 6W.

The results show that if the parasitic FPA heat is very small, the design will work with a 1mm radiator and an MLI E* of .02. For 2W of added FPA heat the .01 blanket is required but the thinnest radiator still works. For 4W of FPA parasitics, the radiator must be 2mm thick and for 6W, a 4mm panel is needed.

TRADE CONSIDERATIONS / RECOMMENDATIONS

- To maintain the schedule and allow some insurance to minimize the schedule risk, we recommend that a full set of spare mirrors be fabricated.
- If mass becomes a problem resource we would recommend the customer re-look at the baffle assembly which is now aluminum and consider graphite epoxy. Should be able to get 50 kg here.

SUBSYSTEM RISK

*Classifications -**High-Medium-Low***

- Detectors - high - They are still developing both of the detectors. Recommend they look at a commercially available detectors as a backup design.
- Focal planes - high - The mosaics of detectors require special mechanical and electrical design that requires the signal and clock drivers to be derived form the back side of the chip or chip carrier. The data volume and data rates are very high and the design requires low wattage dissipation at the same time.
- Telescope - medium - The light weighting of the optical elements requires state of the art equipment and experienced vendors.
- Instrument Thermal Design - high - The issue is whether the cited 10 W required to maintain the high data rate will be realized. Small 1024 pixel^2 cameras with substantially lower data rates typically require a few watts to operate.

3.0 FLIGHT SYSTEM

3.1 SYSTEMS

DESIGN DESCRIPTION

The Supernova / Acceleration Probe (SNAP) mission will observe type 1a supernovas over a 4-year time span. The purpose of the mission is to test the accelerating expansion of the universe concept using optical and infrared spectrometry. The spacecraft will enter into a highly elliptical orbit around earth with perigee at approximately 2.5 earth radii (Re) and apogee at approximately 25 Re.

For this study, we assumed that the spacecraft will launch in December 2009 aboard a Delta IV 4040 launch vehicle. This launch vehicle has a capability of 3041 kg to the expected C3 of –4.5. We assumed that the mission class is A/B. This means that the redundancy and parts class is slightly less stringent than a "flagship" mission such as Hubble. The redundancy is assumed to be "selected." Stabilization will be 3-axis. Some heritage is anticipated from the FUSE mission. Total radiation during the mission is anticipated to be 30 krad behind 200 mils of aluminum with a radiation dose margin of 2 added.

The customer a mission cost of approximately $xxxM not including operations costs. Phase A is anticipated to last 18 months, phase B 12 months, and phase C/D 48 months (although some subsystems will only require 36 months and thus save some cost). Hardware models will include a "selected" prototype spacecraft, protoflight spacecraft, and prototype as well as a flight instrument. The project will pay the technology development costs from TRL 6. It is anticipated that the spacecraft and instrument will be supplied by an industry supplier. The focal plane and integration and test will be provided by Lawrence Berkeley Labs.

The launch mass of the spacecraft converged at 1967 kg. This value includes a total of 30% contingency. Subsystems apply contingency in compliance with the JPL Design Principles. Additional contingency is then applied at the system's level to assure the overall contingency is 30%. The selected launch vehicle, Delta IV 4040, has a launch capability of 3041 kg providing a launch vehicle margin of 1074 kg or 35%. 35% is a very healthy margin. The peak power mode is the telecom mode requiring 753 W of power. Additional mass and power detail is available in the system's worksheet.

The system's guidelines, worksheet, and equipment list are presented on the following pages. Where there are slight discrepancies between the worksheet and the equipment list, the worksheet should be considered to be more accurate.

DESIGN GUIDELINES AND ASSUMPTIONS

SYSTEMS TABLE 1: STUDY GUIDELINES AND ASSUMPTIONS

Team X Study Guidelines

Supernova/Acceleration Probe

Orbiter

	Programmatics/Mission
Customer	Lawrence Berkeley Laboratory
Study Lead	Andy Gerber
Mission	Supernova/Acceleration Probe
Target Body	Deep Space (Type 1a supernova)
Trajectory	10000 km (2.5 Re) x 25 earth radius orbit
Science/Instruments	Optical and IR spectrometer and imager (2 m aperture mirror)
Potential Inst-S/C Commonality	N/A
Desired Launch Vehicle	Delta IV
Launch Date	2009/12
Mission Duration	4 years
Mission Class	A/B
Technology Cutoff	2005/12
Minimum TRL at End of Phase B	6
	Spacecraft
Redundancy	Selected
Stabilization	3-axis
Heritage	FUSE
L/V Capability, kg	3041 kg to a C3 of -4.5 w/ 0% LV Contingency taken out
Radiation Total Dose	30 krad behind 200 mils alum., RDM 2 added
GPS	No
Type of Propulsion Systems	System 1-Monoprop, System 2-0
Drag Makeup, m/s	None
Post-Launch Delta-V, m/s	230 total (orbit maintainence)
P/L Mass, kg	811
P/L Power, W	66 (average over 320 sec.)
P/L Data Rate, kb/s	1.5 Gbps
P/L Pointing, arcsec	.03 per axis for pitch and yaw, .015 knowledge, maintain .03 per axis over 300 sec.
Tracking Network	12 m
Contingency Carried By	Subsystems
	Costing
Cost Target	500 M (not including operations)
FY$ (year)	2003
Phase A Start (month)	May
Phase A Start (year)	2003
Phase A Duration (months)	18
Phase B Duration (months)	12
Phase C/D Duration (months)	48
Phase E Duration (months)	48
Hardware Models	Selected prototype S/C, protoflight S/C, prototype and flight instrument
New Development Tests	Instrument, Telescope
Project Pays Tech Costs from TRL	6
Spares Approach	Selected
Parts Class	Commercial + Military 883B
S/C Supplier	Industry
Instrument Supplier	Industry (focal plane provided by LBL)
I&T Site	LBL
Launch Site	ETR
Burdens	7% on first $.5M, 0 after that
Reserves	30%

SPACECRAFT SYSTEM

SYSTEMS TABLE 2: SYSTEM WORKSHEET

Supernova/Acceleration Probe Orbiter

SYSTEMS WORKSHEET

Analyst: Jason Andringa
Start Date: #########

Legend

(blue)	Inputs from Subsystems
(pink)	Inputs from Systems
(yellow)	Inputs from other systems
(green)	Calculated

Stabilization - cruise	3-Axis		Pointing Direction - cruise	N/A			
Stabilization - science	3-Axis		Pointing Direction - science	N/S ecliptic	Mission Duration	4.0	years
					Max probe sun distance	1	AU
Pointing Control	0.030	arcsec	Radiation Total Dose, krad	30	Instrument Data Rate	1.5	Gbps
Pointing Knowledge	0.015	arcsec	Science BER	1.00E-06	Data Storage	3000	Gb
Pointing Stability	0.030	relative pointing over 300 sec	Redundancy	Selected	tal Mission Data Volume	######	Mbits
Determined by:	Science				Maximum Link Distance	60000	km
			Technology Cutoff	2005/12	Return Data Rate	200 Mbps	Mbps

	Mass Fraction	Mass (kg)	Subsys Cont. %	CBE+ Cont. (kg)	Mode 1 Power (W) Science (62 out of 72 hours)	Mode 2 Power (W) Telecom (5 hours)	Mode 3 Power (W) Standy (5 hours)	Mode 4 Power (W) TCM	Mode 5 Power (W) Launch - 6 hrs.	NASA TRL To-day	Mass Change % since last update
Payload											
Instruments	61.9%	811.3	30%	1054.7	65.9	0.0	0.0	0.0	0.0	0	
Payload Total	61.9%	811.3	30%	1054.7	65.9	0.0	0.0	0.0	0.0		
Bus											
Attitude Control	5.9%	76.9	18%	91.1	137.0	149.8	137.0	37.0	27.0	6	
Command & Data	3.5%	45.5	29%	59.0	163.9	276.4	96.4	36.4	36.4	6	
Power	5.8%	76.2	9%	83.2	27.1	35.3	19.4	13.1	11.2	0	
Propulsion1 - Monoprop	2.8%	36.6	18%	43.1	1.3	1.3	1.3	29.3	25.3	aually Ent	
Structure	11.0%	144.0	30%	187.2	0.0	0.0	0.0	0.0	0.0	6	
S/C Adapter	1.5%	20.2	30%	26.3							
Cabling	3.2%	42.4	30%	55.1						7	
Telecomm	1.4%	18.7	27%	23.8	5.4	73.1	5.4	40.4	40.4	0	
Thermal	3.0%	39.1	9%	42.7	43.0	43.0	58.4	58.4	43.0	6	
Bus Total		499.7	22%	611.5	377.7	578.9	318.0	214.7	183.3		
Spacecraft Total (Dry)		1311.0	27%	1666.1	443.6	578.9	318.0	214.7	183.3		
Subsystem Heritage Contingency		355.2	27%								
System Contingency		38.1	3%		133.1	173.7	95.4	64.4	55.0		
Spacecraft with Contingency		1704.3			576.6	752.6	413.3	279.1	238.3		
Propellant & Pressurant1	13.4%	262.9			For S/C mass =	2000	Delta-V1, Sys 1		230	m/s	
					Separated Mass 1 =		Delta-V2, Sys 1		0	m/s	
Propellant & Pressurant2	0.0%	0.0			For S/C mass =	0	Delta-V1, Sys 2		0	m/s	
					Separated Mass 2 =		Delta-V2, Sys 2		0	m/s	
Spacecraft Total (Wet)		1967.2									
L/V Adapter		0.0									
Launch Mass		1967.2									
Launch Vehicle Capability		3041.1	Delta 4040-12 *					Launch C3	-4.5	LV TRL	9
								Mission Unique LV Contingency	0%		
								Fairing type	standard		500 M (not
Launch Vehicle Margin		1073.9	35%					Fairing dia., m	?		Cost Margin

Contingencies	Mass	Power
Instruments	30%	30%
Other	N/A	N/A
S/C, dry	30%	30%

SYSTEM TABLE 3: MASS AND EQUIPMENT LIST

Mission: Supernova/Acceleration Probe
Element: Orbiter

System Level - Summary of Equipment List

Instruments

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
2 m TMA Telescope (minus thermal	1	811.270	811.270	30%	1054.651			TBD			None

Attitude Determination and Control System

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Sun Sensors	8	0.005	0.037	10%	0.040	0.0	0.0	Adcole, coarse	Goddard	9	
Star Trackers	2	5.400	10.800	10%	11.880	10.0	10.0	Ball CT-602, 1 arcsec		7	Accuracy 1 arcsec P/Y, 7.8 by 7.8 deg FOV
IMU	1	4.500	4.500	10%	4.950	27.0	27.0	Litton, Scalable SIRU	Based on Litton SIRU	8	0.003 deg/hr bias stability, 0.0001 deg/rt-hr rand walk, 4
Reaction Wheels #1	4	14.000	56.000	20%	67.200	105.0	22.0	Honeywell, HR16 Max	Various	6	150 Nms momentum storage, 0.1 to 0.2 Nm torque
HGA Drive Motors	2	2.200	4.400	30%	5.720	20.0	12.0	MOOG (Shaeffer Mag) Type 5	HST, TOPEX, Landsat, Mars Observer, GPS	9	Min step 0.0055 deg with HD 255:1 ratio, 113 Nm max torque,
Gimbal Drive Electronics	2	0.590	1.180	10%	1.298	4.8		Spectrum Astro, DS-1 Gimbal Drive Electronics	DS-1	9	Assumed to control 4 motors per board

Command and Data System

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
SFC	2	0.520	1.040	15%	1.196	11.5	5.8	TBD		6	From JSC equipment sheet. RAD-750. 15 watts full; 2.3
NVM	2	0.320	0.640	15%	0.736	2.2	1.1	TBD		6	From equipment list - 2 Gbits - Latest Max power on R & W =
Mars05' CMIC-II	2	0.350	0.700	30%	0.910	2.2	1.1	TBD		6	3U Dual String Control card 6/02 Peer Review
Mars05 DTCI	2	0.500	1.000	30%	1.300	6.6	3.3	TBD		6	6U? 12 High Speed LVDS , 16 discreet inut, 16 Discrete
Mars05 ULDL	2	0.500	1.000	30%	1.300	3.9	2.0	TBD		6	3u Comand decoding 0 to 100kbits/sec, BCH decoding,
Mars05 GIF	2	0.350	0.700	30%	0.910	5.0	2.5	TBD		6	3U Mil-STD-1553B, 48 discrete in, 48 discreteout, Analog signal
Mars05 AAC	4	0.350	1.400	30%	1.820	5.0	2.5	TBD		6	Analog Acquisition 30 rice PCI, 48 voltage input, 16 passive
0	1	36.600	36.600	30%	47.580	240.0	63.8	TBD			0
Shielding (kg)	1	2.200	2.200	30%	2.860			TBD			None

Power

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Solar Array	1	8.895	8.895	30%	11.563			TBD		9	None
Ni-H2 (CPV) Battery	1	52.941	52.941	30%	68.824			TBD		9	None
Array Switching Boards	1	0.800	0.800	30%	1.040			TBD		9	None
Load Switching Boards	1	0.800	0.800	30%	1.040			TBD		9	None
Thruster Drivers Boards	4	0.800	3.200	30%	4.160			TBD		9	None
Pyro Switching Boards	1	0.800	0.800	30%	1.040			TBD		9	None
Converters Boards	8	0.800	6.400	30%	8.320			TBD		9	None
Shunt Regulators Boards	0	0.800	0.000	30%	0.000			TBD		9	None
Down Converter Boards	1	0.800	0.800	30%	1.040			TBD		9	None
Battery Control Boards	1	0.800	0.800	30%	1.040			TBD		9	None
ARPS Controller Boards	0	0.800	0.000	30%	0.000			TBD		9	None

SYSTEM TABLE 4: MASS AND EQUIPMENT LIST (CONTINUED)

Propulsion #1

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Gas Service Valve	6	0.230	1.380	2%	1.408			TBD			None
HP Latch Valve	2	0.350	0.700	2%	0.714			TBD			None
HP Transducer	1	0.270	0.270	2%	0.275			TBD			None
Gas Filter	5	0.110	0.550	2%	0.561			TBD			None
NC Pyro Valve	9	0.120	1.080	2%	1.102			TBD			None
Press Regulator	2	0.740	1.480	2%	1.510			TBD			None
Temp. Sensor	1	0.010	0.010	5%	0.011			TBD			None
Liq. Service Valve	1	0.280	0.280	2%	0.286			TBD			None
LP Transducer	4	0.270	1.080	2%	1.102			TBD			None
Liq. Filter	1	0.400	0.400	2%	0.408			TBD			None
LP Latch Valve	8	0.350	2.800	2%	2.856			TBD			None
Temp. Sensor	10	0.010	0.100	5%	0.105			TBD			None
Lines, Fittings, Misc.	1	1.800	1.800	50%	2.700			TBD			None
Monoprop Main Engine	0	0.000	0.000	0%	0.000			TBD			None
Monoprop Thrusters 1	8	0.330	2.640	10%	2.904			TBD			None
Pressurant Tanks	1	4.863	4.863	0%	4.863			TBD			None
Fuel Tanks	4	4.287	17.150	30%	22.294			TBD			None

Structures

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Primary Structure	1	80.958	80.958	30%	105.245			TBD		6	None
Secondary Structure	1	9.715	9.715	30%	12.629			TBD		6	None
Telescope Interface Structures	1	6.342	6.342	30%	8.244			TBD		6	None
Baffle support structure	1	8.856	8.856	30%	11.512			TBD		6	None
Solar Array Structure	2	5.397	10.793	30%	14.031			TBD		6	None
Antenna Articulation Mechanism	1	8.700	8.700	30%	11.310			TBD		6	None
Integration Hardware & MHSE	1	5.667	5.667	30%	7.367			TBD		6	None
Balance Mass	1	12.981	12.981	30%	16.875			TBD		6	None
Adapter, Spacecraft side	1	20.243	20.243	30%	26.316			TBD		6	None
Cabling	1	42.373	42.373	30%	55.084			TBD		7	None

Telecomm

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
X/Ka 0.5m diam High Gain Antenna(l	1	0.700	0.700	30%	0.910			TBD		0	0.5m
S-band omni Lockheed-Martin	4	0.300	1.200	30%	1.560			TBD		0	0
STDN Compatible Transponder w/FI	2	2.500	5.000	20%	6.000	40.4	40.4	TBD		0	None
Ka-band (10Mbps) Xmtr w/Turbo6 co	2	3.900	7.800	30%	10.140			TBD		0	None
Ka-band SSPA, RF = 3W	2	0.500	1.000	30%	1.300	20.0	20.0	TBD		0	None
Additional Hardware	8	0.375	3.000	30%	3.900			TBD		0	None

SYSTEM TABLE 5: MASS AND EQUIPMENT LIST (CONTINUED)

Thermal

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Multilayer Insulation			9.553	30%	12.419			TBD		6	None
Thermal Surfaces			0.643	30%	0.836			TBD		7	None
Thermal Conduction Control			1.673	30%	2.175			TBD		6	None
Louvers Total Mass	5	0.975	4.875	30%	6.338			TBD		7	None
Heaters/Thermostats			2.65	21%	3.206	85.9	43	TBD		7	None
Temp Sensors			0.3	10%	0.33			TBD			None
valves			2		2			TBD			None
radiator			16		16			TBD			None
graphite straps			1.4		1.4			TBD			None

Element: **Orbiter**

System Level - Summary of Equipment List

Subsystem	Flt Units		Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Total Peak Power per Unit (W)	Total Aver. Power per Unit (W)	Note: The accompanying powers listing reflect the artificial situation when all of the devices on the spacecraft are on or at peak power. Please refer to system sheet for proper modes.		Lowest TRL	Description/Comments
Instruments	1		811.3	30%	1054.7	0.0	65.5			0	
Attitude Determination and Control System	20		76.9	18%	91.1	166.0	71.0			6	
Command and Data System	18		45.3	29%	58.6	278.4	62.1			6	
Power	21		75.4	30%	98.1	0.0	0.0			0	
Propulsion #1	64		36.6	18%	43.1	0.0	0.0			0	
Propulsion #2	0		0.0		0.0	0.0	0.0			0	
Structures	11		206.6	30%	268.6	0.0	0.0			6	includes cabling and s/c adapter
Telecomm	19		18.7	27%	23.8	60.4	60.4			0	
Thermal	5		39.1	14%	44.7	85.9	43.0			6	
System Total	159		1309.9	28%	1682.6	589.5	322.4			6	

SYSTEM RISK

There are a couple items about this mission that make it risky. Manufacturability of the focal plane may be very difficult. The mission requires a 2 m aperture focal plane. The pointing requirements are also significantly tighter than other similar missions that either are flying (Hubble) or are in development (Kepler).

3.2 Propulsion

Subsystem-Level Requirements

This propulsion system is designed to provide the following functions for the SNAP:

1. Momentum wheel desaturation approximately once every 4 days.
2. 3-axis spacecraft control and safing.
3. Delta V maneuvers per the following Delta-V table:

Figure 1. Delta V Budget.

Maneuver	Delta V (m/s)	Propellant Mass (kg)
Launch cleanup	230	195.44
4 yrs RCS/Momentum dump		60.00
Residuals		6.39
Total		261.83

Sixty kilograms of propellant are allocated for momentum wheel desaturation maneuvers. This estimate was derived by Attitude Control and is based on the assumption of monopropellant hydrazine thrusters.

Other requirements include:

1. Minimize slosh dynamics. This slosh effect will adversely affect the ultimate pointing control that is achievable.
2. Selected redundancy. Redundancy will be applied only for critical systems with a reasonable chance of failure.

Design Description

The major design driver for this mission is minimization of slosh dynamics. To this end, a regulated monopropellant hydrazine system has been designed for momentum dumping and routine trajectory correction (including launch vehicle dispersion correction). This system has extensive flight heritage and is routinely used for all types of in space propulsion needs.

In order to minimize slosh, the monopropellant system utilizes 4 tanks with elastomeric diaphragms. Tanks are filled to near capacity at launch with liquid ultra pure anhydrous hydrazine (N_2H_4). In operation, only one tank is used at a time. In this way, it is postulated that the net dynamic effect of slosh is reduced by a factor of 4 when compared with using all tanks simultaneously. In addition, the pressure regulation system allows tanks to be filled to near capacity at launch, thereby limiting the amount of slosh in the unused tanks. Finally, the elastomeric diaphragm in each tank will provide a small additional amount of slosh dissipation.

The pressurization system is based on similar systems flown on Cassini, MGS, and Mars Odyssey. A two stage regulator provides regulation from an initial pressurant tank pressure of 3500 psia to a nominal propellant tank pressure of about 300 psi. A high pressure latch valve isolates the regulator from the high pressure Helium during non operational periods. In the event of a regulator failure, a parallel path is included that uses a single high pressure solenoid valve in series with an orifice for 'bang bang' pressure control. In normal operation, the regulator will be used to bring the tank ullage to 300 psia. Then the high pressure latch valve will be closed, isolating the regulator. The system will then operate in blowdown mode until a minimum pressure is reached (probably near to 200 psia). Then the cycle will repeat. This minimizes the amount of total operating time required of the regulator over the mission life.

The feedsystem is designed to draw propellant from each tank individually. Each tank is isolated with a latch valve. The latch valve is opened when the tank is to be used and closed when the tank has been emptied. Closing of the latch valve also prevents thermal pumping of propellant between two open tanks.

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Contin- gency	CBE+ Contin- gency (kg)
Gas Service Valve	6	0.230	1.380	2%	1.408
HP Latch Valve	2	0.350	0.700	2%	0.714
HP Transducer	1	0.270	0.270	2%	0.275
Gas Filter	5	0.110	0.550	2%	0.561
NC Pyro Valve	9	0.120	1.080	2%	1.102
Press Regulator	2	0.740	1.480	2%	1.510
Temp. Sensor	1	0.010	0.010	5%	0.011
Liq. Service Valve	1	0.280	0.280	2%	0.286
LP Transducer	4	0.270	1.080	2%	1.102
Liq. Filter	1	0.400	0.400	2%	0.408
LP Latch Valve	8	0.350	2.800	2%	2.856
Temp. Sensor	10	0.010	0.100	5%	0.105
Lines, Fittings, Misc.	1	1.800	1.800	50%	2.700
Monoprop Thrusters	8	0.330	2.640	10%	2.904
Pressurant Tanks	1	4.863	4.863	0%	4.863
Fuel Tanks	4	4.287	17.150	30%	22.294
Total			36.583	18%	43.098
		Total Hydrazine			261.8
		Total Helium			1.1
		Total Wet Propulsion System			305.998

Eight 4.5N thrusters are included for general attitude control, momentum unloading, and Delta-V. These thrusters have a minimum impulse bit of about 7 milliNewton-sec. Average power consumption is dominated by the catalyst bed heater, which is estimated to consume about 1.5W per thruster for a total of 12W. The catalyst bed heater must be used for 20-40 minutes before using the engines to preserve thruster life. For possible safe mode applications, they should be powered at all times. The valve requires a typical opening power of 8W. These thrusters have extensive heritage including Intelsat, MPF, and MER.

The mass-equipment list for this system is shown above.

FIGURE 2. PRESSURE CONTROL ASSEMBLY.

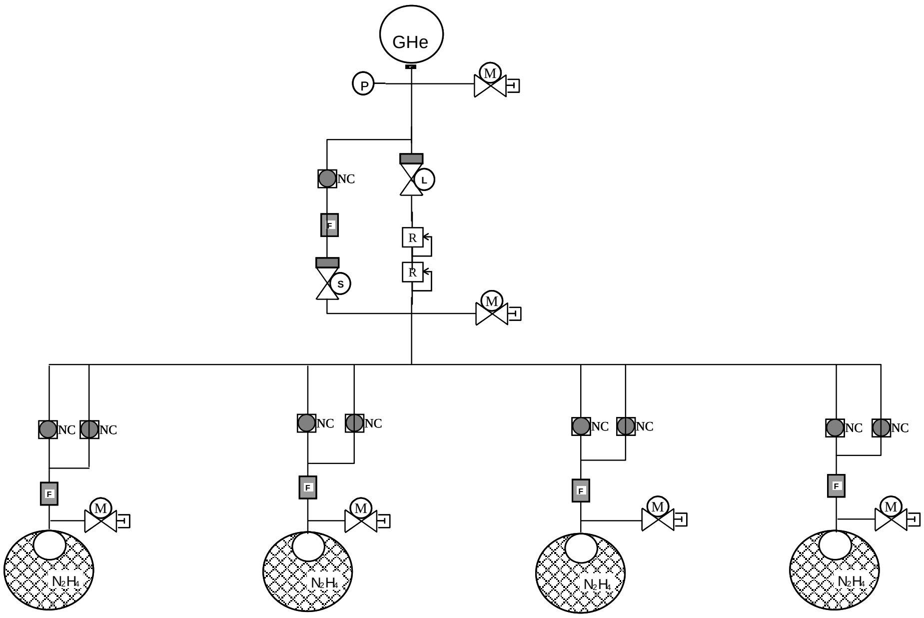
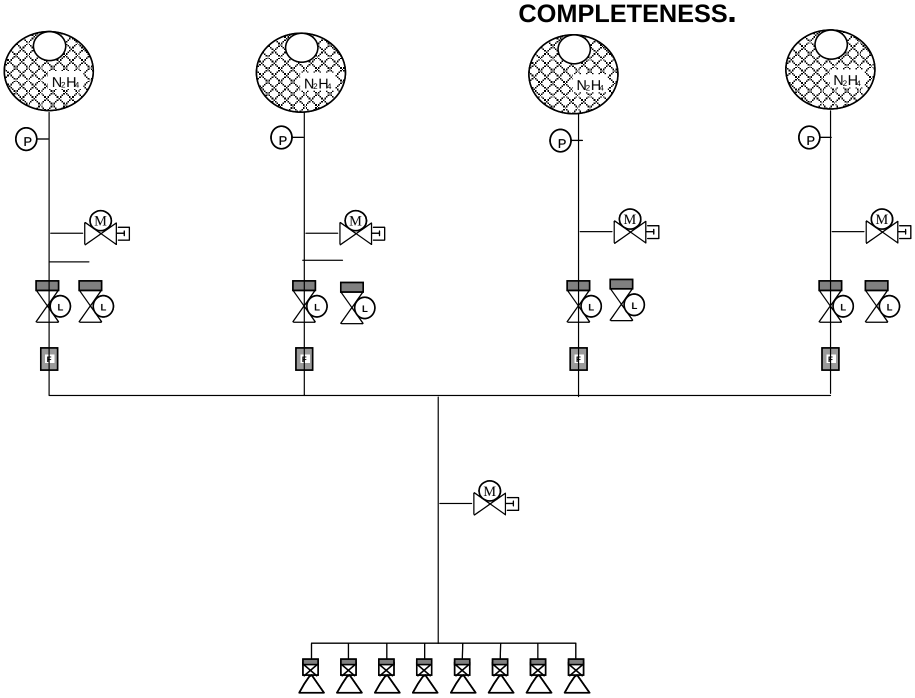


FIGURE 3. PROPELLANT ISOLATION ASSEMBLY. NOTE: TANKS REPEATED FOR COMPLETENESS.



TRADE CONSIDERATIONS

There are several options available for consideration including, but not limited to, bipropellant, dual mode, and a combination of solid rocket and monopropellant.

A bipropellant or dual mode system would reduce the total propellant mass by up to 20%. Bipropellant systems typically use fuels such as hydrazine (N_2H_4) and monomethyl-hydrazine (CH_3NHNH_2) and oxidizers such as nitrogen tetroxide (N_2O_4) and mixed oxides of nitogen (MON-X). Bipropellant engines require a pressure regulated propellant feed for proper operation. Biprop systems tend to be more complex and more expensive than comparable monopropellant hydrazine systems. A dual mode system is a subclass of bipropellant systems that use N2H4 and usually N2O4. In these systems, the main engine operates in bipropellant mode while the RCS thrusters operate in catalytic monopropellant mode. Dual mode systems are more optimal for spacecraft requiring many small maneuvers and a few large maneuvers with RCS control. In this case there was no substantial benefit to using either a bipropellant or dual mode system other than a somewhat reduced mass.

A solid rocket motor typically uses a mixture of ammonium perchlorate, aluminum powder, and a binder enclosed in a titanium case. The primary benefits of the SRM are zero slosh, excellent propellant mass fraction (typically 88% or better), and low cost. The SRM is ideal for the end of mission maneuver required of the SNAP spacecraft. However, several issues led away from this choice. First, the configuration of the spacecraft was complicated, in particular, the SRM interfered with the placement of the high gain antenna. Also, there are concerns about the long term storability of an SRM. The longest deep space missions using SRMs were Magellan (6 months) and Mars Pathfinder (6 months). SNAP would require several years of storage. That said, the manufacturer, ATK, claims there is no demonstrated issue with the propellants that are used today. In addition to an SRM, a small hydrazine monopropellant system would be used for RCS control and for control of the SRM during its maneuver.

A monopropellant hydrazine system was chosen to minimize risk and complexity.

SUBSYSTEM RISK

This system is in the Medium to Low risk range.

REASONS WHY

This system has an enormous amount of heritage on both commercial and NASA spacecraft. The system can be built and operated successfully with a high degree of confidence. The only credible risk is that of slosh dynamics. It is possible that the current design will not be able to achieve the level of slosh dynamics that the flight system will require. This question can be answered with analysis and test in Phase A.

3.3 ACS

<u>Subsystem-Level Requirements</u>

Pointing Stability: The goal for telescope pointing stability during imaging is within ±0.01 arcsec (1 sigma) in pitch and yaw (boresight) over 300 seconds per image for four successive images (which would take a total of 1200 seconds). The requirement is pointing stability within ±0.03 arcsec (1 sigma) in pitch and yaw. That is, the requirement is 3 times looser than the goal. These are numbers provided by the customer. The smallest pixel size for the instrument would be 0.1 by 0.1 arcsec, assuming a 10.5 μm pixel pitch and a 21.65 m focal length. The telescope would have a blur spot with a diameter somewhat wider than two pixels (e.g., 0.24 arcsec). A 1-sigma pitch or yaw error of 0.01 arcsec (best case) would correspond to one-tenth of a pixel smear. A 3-sigma pitch or yaw error of 0.09 arcsec (worst case) would correspond to nearly a full pixel (38% of the blur spot) smear.

The goal for pointing stability in roll (twist about the boresight) is to within ±0.27 arcsec (1 sigma) over 1200 seconds. The requirement is stability to within ±0.67 arcsec (1 sigma) over 1200 seconds. These values were derived assuming a radial distance of 13 milli-rad from the telescope boresight to the pixel farthest from the boresight. In that case, a 3-sigma peak roll error of 2 arcsec would result in about one-quarter of a pixel error in the worst location on the focal plane. A 3-sigma peak roll error of 0.81 arcsec would result in about one-tenth of a pixel error in the worst location on the focal plane.

Pointing Control: In information provided by the customer to Team X, there is a stated requirement for pointing accuracy to within ±1 arcsec (3 sigma) in pitch and yaw and to within ±2 arcsec (3 sigma) in roll for periods of up to 1000 seconds. Once calibration is complete and the fine guidance sensors (FGS's) have acquired and are providing data for attitude determination, the stated requirements would be met. However, the real requirement for pitch and yaw is pointing control within ±0.09 arcsec (3 sigma) over 1200 seconds so that four successive images can be taken as part of the dithering scheme described below. The requirement for roll may be correctly stated.

There is a requirement for pointing control during FGS acquisition prior to calibration of the bus attitude reference relative to the FGS's. Assuming a FGS FOV of 102 by 102 arcsec, the requirement would be to point the FGS's to within ±51 arcsec (3 sigma) in pitch and yaw to support calibration. This corresponds to pointing control to within half of the FGS FOV.

The requirement for FGS pointing control during post-calibration re-acquisition would be to point to within ±10 arcsec (3 sigma) in pitch and yaw. This corresponds to pointing control within one-tenth of the FGS FOV.

There is a relatively loose requirement for pointing control of the high gain antenna (HGA) boresight. Pointing control of around 0.05 degrees (3 sigma) may be adequate to support the required downlink rates using a 2.7-m antenna at Ka-band. The HGA must be pointed while tracking a ground station on earth during a perigee pass.

Pointing Knowledge: A proposed requirement for knowledge of the telescope boresight is accuracy within ±0.01 arcsec (3 sigma) over 1200 seconds. This would support the goal of pitch and yaw pointing control to within ±0.03 arcsec (3 sigma). The proposed requirement for roll (twist about the boresight) is accuracy within ±0.7 arcsec (3 sigma) over 1200 seconds. This would support the goal of pointing control to within ±0.81 arcsec (3 sigma).

The proposed requirement for knowledge of the telescope boresight prior to calibration is accuracy within ±50 arcsec (3 sigma) in pitch and yaw. This would support the requirement to point the boresights of the FGS's to within half of their FOV's.

The proposed requirement for knowledge of the telescope boresight during post-calibration re-acquisition is accuracy within ±9 arcsec (3 sigma) in pitch and yaw. This would support pointing control within one-tenth of the FGS FOV.

The loose requirement for knowledge of the HGA boresight is accuracy within 0.025 degrees (3 sigma). This is half of the pointing control requirement.

Slew Maneuvers: There would be both small and large angle slews required over the mission. Small angle slews would be up to 3 arcmin between observations, and there would presumably be several to many of these per day. The time allocated for slew and settle was not discussed in detail during the Team X study, but the customer provided preliminary information that indicates a minimum time of 1800 seconds for re-targeting.

There would be roll slews of 90 degrees four times a year. These large angle slews would be required to insure that the thermal shield and solar array can be properly oriented throughout the year. The time allocated for these large angle slews was not discussed during the Team X study, but the reaction wheels were sized to achieve such a slew within tens of minutes. The time required for the telescope to reach a new thermal equilibrium would likely be considerably longer than the time required for slew and settle.

PRELIMINARY ERROR BUDGET

Shown below is a preliminary error budget for pointing stability over four consecutive imaging intervals taking a total of 1200 seconds. The structure of the error budget is based on a similar error budget for SIRTF.

The top-level requirement is ±0.03 arcsec (1 sigma) in pitch and yaw. Thermal/mechanical stability and pointing stability in the star tracker assembly (STA) frame are assigned equal allocations. Similarly, disturbances and attitude determination and control are assigned equal sub-allocations under pointing stability in the STA frame. And sub-allocations under thermal/mechanical stability are nearly equal. These allocations and sub-allocations serve as preliminary placeholders until better numbers can be determined.

In each case, the value of a goal is just 3 times smaller than the requirement. The goals may not be reasonable or achievable. However, if the goals were achieved, the overall pointing stability would be ±0.01 arcsec (1 sigma) in pitch and yaw.

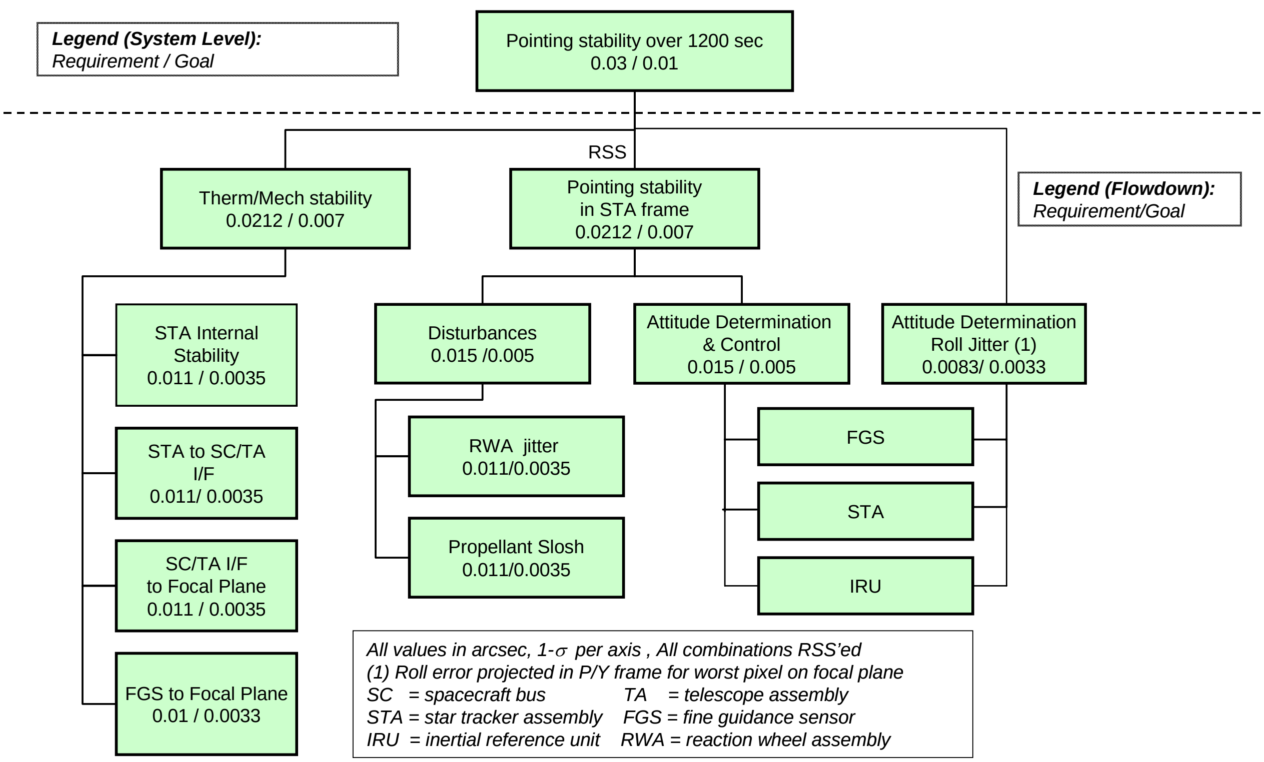
The values for thermal/mechanical stability requirements are in the ballpark of predictions for SIRTF. The values for thermal/mechanical stability goals may not be reasonable. Follow-up is recommended to determine what the limiting factors are for SIRTF or other similar missions.

The values for disturbances are in the ballpark of predictions for SIRTF. The values for disturbance goals may not be reasonable. Follow-up is recommended.

The value for the attitude determination and control stability requirement is a factor of 2 looser than the Hubble Space Telescope (HST) requirement. It should be achievable. HST experienced closed-loop errors due to gyro noise of 0.002 to 0.003 arcsec (1 sigma), which was about half of their requirement. HST has a significantly larger inertia than SNAP would, so the effects of gyro noise could be significantly larger for SNAP. The goal of 0.005 arcsec (1 sigma) for SNAP attitude determination and control stability may not be reasonable, particularly if the effects of gyro noise are larger. Follow-up is recommended.

The requirement and goal for attitude determination roll jitter are based on a preliminary estimate of the pitch/yaw sensitivity to roll errors at the worst location in the focal plane. See the discussion above regarding pointing stability under "Subsystem Level Requirements" for ACS.

Error Budget for Pointing Stability (over 1200 sec)



DESIGN DESCRIPTION

OVERVIEW OF ACS DESIGN APPROACH

The baseline is to provide relatively coarse bus pointing control during initial deployment, for a safe mode, and during reaction wheel momentum unloading. Tighter pointing control would be provided for the high gain antenna (HGA) during data downlink. The tightest bus pointing control would be provided for the telescope during imaging.

The proposed orbit is highly elliptic with a perigee of 2.5 earth radii (10,000 km altitude) and an apogee of 25 earth radii. The orbit period would be 72 hours (3 days). Imaging would be performed for 62 hours while away from the earth. For 10 hours during which the vehicle is below 60,000 km altitude there would be no imaging. Data would be downlinked during a 5-hour window while near perigee.

The mission involves a near-IR telescope with a focal plane passively cooled to 140 °K. The pointing stability goal for the telescope (within ±0.03 arcsec over 300 seconds, 3 sigma) is only about 40% looser than the requirement for the Hubble Space Telescope (HST). This mission is fairly challenging and in a class with few peers.

The telescope will include fine guidance sensors (FGS's) on the focal plane. The details of the fine guidance sensors need to be worked out. Suppose the FGS were a CCD array with 1024 by 1024 pixels and a 10.5 μm pixel pitch. Then for a 21.65 m focal length, the angular pixel size would be 0.1 by 0.1 arcsec and the FOV of the FGS would be 102.4 by 102.4 arcsec. In that case, pointing control of the telescope boresight during FGS acquisition would ideally be within ±10 arcsec (3 sigma). During initial calibration of the bus attitude reference relative to the FGS's, looser pointing control to within ±50 arcsec (3 sigma) might be acceptable, depending on the calibration approach.

There would be at least four FGS's at the corners of a box in the focal plane. At least 2 would be in use at one time. The number in the baseline and the number in use need to be considered further in the context of star availability studies. Having more stars available would help to reduce the effects of noise.

Once the FGS's have acquired targeted stars, they will be used to provide star measurement data to the on-board computer on the bus. The bus will include a set of precision gyros that provide a highly stable short-term reference. A Kalman filter will be used to blend FGS measurements with gyro measurements in order to filter out noise from both types of sensors.

The Kalman filter gains need to be worked out in detail during later studies. The concept discussed during the Team X study was that the FGS's is expected provide an extremely accurate knowledge reference that can be trusted heavily. Nonetheless, the gyros are expected to provide a very stable short-term reference that would be useful for filtering out FGS noise and improving accuracy. The gyros would have a bandwidth of 200 Hz and be capable of accurately sensing jitter rates up to and beyond 20 Hz. The FGS's might have a sample rate of 10 or 20 Hz or perhaps higher. A sample rate of 20 Hz would be sufficient to accurately sense jitter rates up to a few Hz.

The spacecraft is expected to have no vibration modes below 20 Hz. The plan is to build a relatively stiff structure with a body mounted solar array. The only moving appendage would be the HGA, and the HGA would remain fixed during imaging. The HGA gimbal mechanism and drive motors would be designed to provide a relatively high degree of stiffness to preclude any antenna vibration modes with frequencies below 20 Hz. Since passive cooling is used for the focal plane, there would be no disturbance associated with a cryo-cooler pump. The reaction wheels are the only anticipated disturbance source. They will be passively isolated to minimize the transmission of disturbances above 10 to 15 Hz, and the wheels will be operated at speeds that do not generate any significant disturbances below 10 to 15 Hz.

The expectation is that there will be minimal jitter, so there will not need to be a fast steering mirror assembly in the optical train. However, there is a dithering requirement to shift the telescope LOS by half a pixel between images, and a steering mirror assembly is recommended to accomplish this. The proposed plan is to take four 300-second images in succession at the corners of a square box that is half pixel on a side (e.g., 0.05 arcsec on a side) to increase the effective resolution of the instruments. Ideally, the box would be centered within one particular pixel. A flight-proven approach to accomplish this type of fine adjustment is to use a slow steering mirror assembly. The SNAP secondary mirror focus mechanism may have enough capability to be used as a slow steering mirror for this purpose. This can be explored further in follow-on studies.

The pointing stability goal during imaging is to within ±0.03 arcsec (3 sigma) in pitch and yaw over 1200 seconds for four images. The goal for roll is to within ±0.81 arcsec (3 sigma) over 1200 seconds. For dithering, the pointing control commands would target the corners of the desired square box. Worst-case pointing control errors would have a magnitude similar to the width of the box, so the actual pointing control might not reproduce a square at all. For instance, there is a small statistical chance that two corners might be nearly coincident, resulting in a triangle. According to Team X Instruments (Ed Danielson), variations from a perfect square are acceptable. Most of the time, actual pointing control would result in nearly a square, and post-processing on earth can be used to correct for variations and generate enhanced resolution.

For the moment, assume a FGS pixel size of 0.1 by 0.1 arcsec and sub-pixel interpolation during star centroiding to within one-tenth of a pixel (fairly conservative). This would correspond to per-axis knowledge accuracy to within ±0.01 arcsec (3 sigma). In that case, the allocation for all other errors would be ±0.02 arcsec (3 sigma) if the intent were to meet the goal of pointing control within ±0.03 arcsec (3 sigma). One of the largest error sources for HST was gyro noise, which induced errors of ±0.006 to ±0.009 arcsec (3 sigma) per axis in the closed loop control system. The expectation for SNAP is that gyro noise would likewise by a significant error source, but that the pointing control goal might nonetheless be met.

In information provided by the customer to Team X, there is stated a requirement to point the telescope boresight to within ±1 arcsec (3 sigma) in pitch and yaw and to within ±2 arcsec (3 sigma) in roll for periods of up to 1000 seconds. Once the FGS's have acquired and the FGS measurements are being used for attitude determination, the stated requirements would be met. However, the real goal for pitch and yaw is to point the boresight to within ±0.03 arcsec (3 sigma) over 1200 seconds so that four successive images can be taken as part of the dithering scheme described above.

Primary Attitude Control Approach for the Bus

The spacecraft bus will include reaction wheels used for fine pointing control of the telescope. Thrusters will be used to unload excess angular momentum accumulated in the reaction wheels due to external torques.

REACTION WHEELS

A set of 4 reaction wheels will be used for fine pointing control. Four Honeywell model HR16 reaction wheels are included in the baseline. These will be oriented in a pyramid configuration. The symmetry axis of the pyramid that passes through the apex would be oriented to optimize the wheel torque and momentum capabilities as needed to meet slew and momentum storage requirements. Each wheel has the following characteristics:

- Maximum torque of 0.2 Nm,
- Maximum momentum storage of 150 Nms
- Peak power of 105 Watts
- Average power of 22 Watts to maintain constant speed at maximum momentum
- Maximum 14 kg mass for one wheel with electronics

The reaction wheels will need to be finely balanced and have high quality bearings to minimize the possibility of inducing vibrations that could be transmitted through the structure to the telescope. In addition, the Team X baseline includes passive vibration isolation between the wheels and the instrument package because of the tight pointing requirements. Structures is carrying the vibration isolation mass. The goal would be to isolate wheel induced disturbances above 10 to 15 Hz.

The primary wheel disturbance of concern for the Hubble was an axial vibration mode (along the wheel spin axis) due to imperfect bearings. There was some potential for wheel-induced excitation of spacecraft vibration modes, resulting in jitter in the optical path. The wheels used on the Hubble were passively isolated from the rest of the spacecraft structure to help minimize transmission of disturbances above 15 to 20 Hz, and effort was spent to make sure the wheel bearings were manufactured within tight tolerances and the wheels were finely balanced.

The baseline approach for this mission is to passively isolate the reaction wheels from the rest of the spacecraft to minimize the transmission of disturbances above 10 to 15 Hz. Ideally, the wheels would not generate any significant disturbance with a frequency less than 10 Hz.

One of the reaction wheels used for SIRTF was thoroughly characterized through extensive testing during the spacecraft design phase. That wheel was determined to generate no significant disturbances when operated in a range between 200 and 1200 RPM, but it has the potential to generate a significant disturbance when operated above 1200 RPM. Consequently, the SIRTF approach is to command the wheels such that they remain within the 200 to 1200 RPM range.

The baseline approach proposed for this mission is similar to that used for SIRTF. One of the reaction wheels would be thoroughly tested and characterized by the vendor (Honeywell) if that model has not already been analyzed. A suitable range of wheel speeds for nominal operations would be identified. The intent would be to find a relatively wide range in which the wheels are not likely to generate a significant disturbance. The range would be biased away from zero speed to avoid the possibility of stiction as a wheel passes through zero.

THRUSTERS

The thruster set includes sixteen 0.7-N hydrazine minimum impulse thrusters (MIT thrusters) and four 22-N hydrazine thrusters. The 22-N thrusters would be used for delta-V maneuvers. If needed, they could be off-modulated for pitch and yaw control. A subset of the MIT thrusters would be used for roll control during delta-V maneuvers.

The sixteen MIT thrusters would be configured in redundant strings. The intent would be to fire MIT thrusters as couples to provide 3-axis attitude control in the following situations:

- When unloading momentum from the reaction wheels
- When the reaction wheels are unavailable due to failure
- During initial deployment
- For roll control during delta-V maneuvers

See the Propulsion subsystem section for more detail regarding the thrusters. See the Power subsystem section for a description of the propulsion drive electronics.

BACKUP ATTITUDE CONTROL APPROACH FOR THE BUS

If the reaction wheels were not available to support fine pointing control, the bus would resort to a backup pointing control capability using MIT thrusters. This would not be expected to meet the pointing stability requirement or to support imaging.

PRIMARY ATTITUDE DETERMINATION APPROACH DURING FGS ACQUISITION

A precision star tracker and precision gyros will be used for stellar inertial attitude determination on the bus to support FGS acquisition. Measurements from the star tracker and gyros will be blended in a Kalman filter to help minimize the effects of noise in each type of sensor and improve the bus attitude reference. The bus will be able to point the telescope boresight to within ±50 arcsec (3 sigma) during FGS acquisition.

STAR TRACKER

The baseline includes redundant Ball CT-602 star trackers, which have a high degree of heritage. The CT-602 provides pitch and yaw accuracy to within ±1 arcsec (1 sigma). It has a mass of 5.4 kg and requires 10 Watts of power. The baseline would be to align the boresight of the star trackers at a large angle off the boresight (or anti-boresight) of the telescope. This would provide greater accuracy for determination of the telescope roll angle (twist about the boresight).

PRECISION GYROS

The baseline includes an internally redundant Litton Scalable SIRU containing hemispherical resonating gyros (HRG's). The SIRU has redundant electronics and contains 4 HRG's. The gyros are fully internally cross-strapped so that any combination can be used with either set of electronics. Each set of electronics includes a processor, power supply, and I/O. The mass of the IMU is 4.5 kg, power is 27 Watts (including 7 Watts for thermal control), and the cost is estimated at $x.x M in FY '03 $ before procurement burden.

The predicted 1-sigma bias stability is 0.0003 arcsec/sec and the predicted 1-sigma random walk is 0.00001 deg/rt-hr. Over an interval of 1 second (conservative estimate of longest time between FGS measurements), an HRG would provide 3-sigma accuracy to within ±0.003 arcsec, per axis. This would be a very stable short-term attitude reference to complement the star tracker and FGS's.

The SIRU is robust to radiation and can tolerate a dose of 100 krad. Assuming that the IMU is located internal to the bus where it is shielded by the equivalent of 100 mils of Aluminum, it will require no additional shielding.

Backup Attitude Determination Approach

During initial deployment, for fault detection, and in safe mode, coarse sun sensors and gyros will be used for attitude determination. The gyros would be the same HRG's used for primary attitude determination. The sun sensors are described below.

Sun Sensors

The baseline includes 8 Adcole coarse analog sun sensors. These have a mass of grams several grams each and use virtually no power. They are accurate to within a few degrees per axis. The intent is to use them to provide coarse knowledge of where the sun is for a safe mode or during initial deployment.

Pointing Control for the High Gain Antenna

The baseline approach is to attach the high gain antenna (HGA) to the bus via a gimbal mechanism with 2 degrees of freedom. The detailed design of the gimbal mechanism and drive motors needs to be worked out. A proposed concept would be to use stepper motors with harmonic drives that result in a relatively high stiffness when the motor is off. The plan would be to leave the HGA in a fixed orientation relative to the bus during imaging. It would be rotated as needed to track a ground station during each perigee pass while there is no imaging. The goal is to keep the spacecraft's lowest frequency vibration mode above 20 Hz. So, the HGA gimbal mechanism and drive motors need to be designed with enough inherent stiffness to keep antenna vibration modes above 20 Hz.

Stepper Motors

The baseline includes two MOOG (Shaeffer Magnetics) Type 5 stepper motors and redundant Spectrum Astro gimbal drive electronics boards. The electronic boards have heritage from Deep Space 1. Each has a mass of 0.59 kg and requires up to 4.8 Watts of power.

Type 5 stepper motors have a minimum step size of 0.0059 deg based on a harmonic drive with a 255:1 gear ratio. They can provide a maximum torque of 113 Nm, and a maximum power-off holding torque of 35 Nm. The mass of one drive motor is 2.2 kg (for just the motor, not the complete gimbal mechanism). Peak power is 20 Watts and average power is 12 Watts during power-on states. The maximum output rotation speed is 1.76 deg/sec. The design of the motor has heritage from HST, TOPEX, Landsat, and various other missions. There are a variety of motor, gear, position sensing, and redundancy options, including 2 or 3 phase steppers or brushless DC motors.

Solar Array

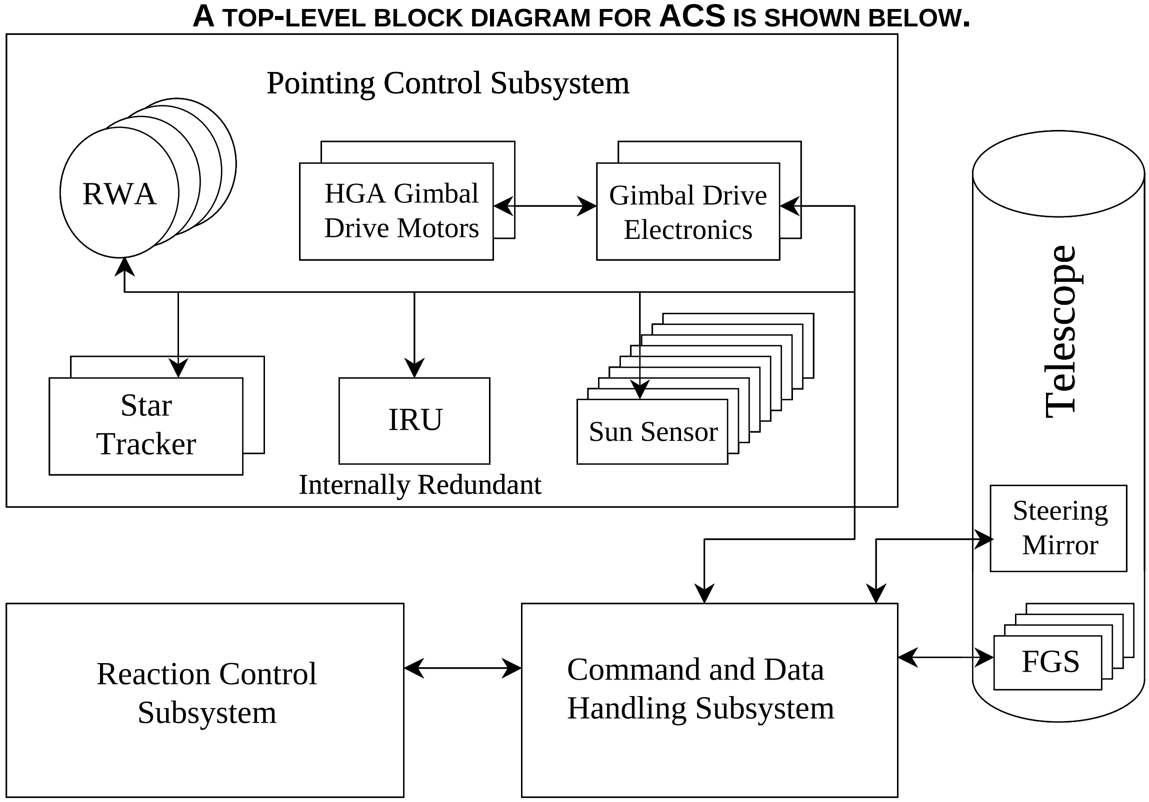
There is a rigid solar array fixed to the bus.

Block Diagram

The ACS baseline of flight hardware includes the components shown in the following table.

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Average Power per Unit (W)	Vendor	TRL	Description/Comments
Sun Sensors	8	0.00	0.04	0.0	0.0	Adcole, coarse	9	Accurate to a few degrees. Coarse analog sun sensors that use virtually no power.
Star Trackers	2	5.40	10.80	10.0	10.0	Ball CT-602, 1 arcsec	7	Accuracy 1 arcsec P/Y; 7.8 by 7.8 deg FOV
IRU	1	4.50	4.50	27.0	27.0	Litton, Scalable SIRU	6	0.0003 deg/hr bias stability, 0.00001 deg/rt-hr rand walk. 4 gyros, 2 sets electronics, 2 pwr supplies, fully cross-strapped
Reaction Wheels	4	14.00	56.00	105.0	22.0	Honeywell, HR16 Max	6	150 Nms momentum storage, 0.1 to 0.2 Nm torque
HGA Drive Motors	2	2.20	4.40	20.0	12.0	MOOG (Shaeffer Mag) Type 5	9	Min step 0.0059 deg with HD 255:1 ratio, 113 Nm max torque, 545 Kg-m-m max inertial capability
Gimbal Drive Electronics	2	0.59	1.18	4.8	2.4	Spectrum Astro, DS-1 Gimbal Drive Electronics	9	Assumed to control 4 motors per board
Shielding	1	0.00	0.00					

A top-level block diagram for ACS is shown below.



TRADE CONSIDERATIONS

GIMBALED HGA AND DATA DOWNLINK STRATEGY

The HGA is attached to the spacecraft via a gimbal with 2 DOF's. The HGA will remain fixed during imaging to avoid exciting the structure. The gimbals will be used during one 5-hour window every 3 days (i.e., 5 out of every 72 hours). The plan is to point the HGA rather than the spacecraft to avoid changing the thermal environment for the telescope.

The baseline approach is to schedule a dedicated 5-hour window for data downlink near perigee, with no requirement to simultaneously do imaging. Because of the gimbals, there is the possibility of re-orienting the telescope while downlinking data to initialize its orientation for imaging at the end of the perigee pass.

USE OF GPS RECEIVERS FOR ORBIT POSITION DETERMINATION

Early in the study, the proposed baseline included GPS receivers for position determination. There were questions raised as to whether GPS could be used for this particular orbit. This issue was that the spacecraft velocity near perigee will be relatively high, so there would be a high velocity relative to one or more GPS satellites.

Tom Yunck is one of JPL's resident GPS experts. His opinion is that GPS receivers could be made to work for this orbit with some relatively minor modifications to the scheduling algorithms. The GPS signals would be accessible primarily while near perigee since the GPS orbits are at 22,000 km altitude (and perigee is at 10,000 km).

Later in the study, the decision was made to drop the GPS receivers because the required orbit determination accuracy can be achieved using ground tracking resources.

FAST STEERING MIRROR TO MITIGATE JITTER EFFECTS

Early in the Team X study, the use of a fast steering mirror (FSM) assembly in the optical chain was proposed and ruled out. The concept was ruled out because it appears that jitter can be mitigated without a FSM and because there were concerns about how to sense angular motion.

The structure is expected to have no vibration modes below 20 Hz. The reaction wheels will be passively isolated to minimize disturbances transmitted to the structure above 10 Hz. Originally, a FSM was proposed as a means to handle any residual jitter below 10 Hz. The proposed baseline is to operate the reaction wheels in a speed range that minimizes disturbances below 10 Hz. The expectation is that residual jitter will be minimal, so there will not be a need for a FSM.

If a FSM were needed, angular rates would need to be sensed accurately. The preferred approach would be to locate a sensor on the focal plane. There would be difficulty using gyros on the focal plane they generate a significant amount of heat and the focal plane needs to be kept cold. Alternatively, the FGS's could potentially provide information from which angular rates could be estimated. However, this would require differentiation of angular measurements at relatively high sample rates, which tends to introduce a significant amount of noise in the rate estimates. The minimum sample rate would need to be 20 Hz in order to sense rates up to 10 Hz. At 20 Hz, there could also be a significant processing load related to FGS measurements.

Use of Deployable Solar Reflector to Minimize Solar Torque

Early in the Team X study, the use of a deployable solar reflector to minimize solar torque was suggested. The predicted offset between the center of mass and the center of pressure is about 1.7 meters. The effective surface area of the bus plus telescope would be about 19 square meters. Consequently, there would be a continual solar torque of 0.3 milli-Nm. The momentum accumulated due to solar torque over 62 hours would be 65 Nms. Over 4 years, about 35 kg of propellant would be needed just to unload momentum due to solar torque.

Team X Structures took a quick look at how a rigid, deployable solar reflector could be designed such that the center of pressure would be close to the center of mass for the vehicle. This trade showed that a mass on the same order as the propellant mass would be needed in order to design a suitably stiff structure. In addition, there would be an issue as to where to locate thrusters to avoid plume impingement on the deployable reflector. In view of this, the idea of a deployable solar reflector was discarded.

Choice of Gyros for Use in Attitude Determination

One approach suggested during the Team X study was to use all stellar attitude determination during imaging. The choice was made to use stellar inertial attitude determination since a highly stable gyro reference can help to minimize the effects of noise in the FGS's (as well as the gyros). The Hubble and SIRTF both use this approach.

The Litton Scalable SIRU was selected because it has no known wearout mechanisms and provides low bias stability and random walk over a long design life. The SIRU contains four hemispherical resonating gyros (HRG's) that are internally cross-strapped with dual power supplies and electronics for full redundancy. The Kearfott SKIRU V is another alternative that may be attractive. SKIRU V uses mechanical gyros that provide very low noise and long life as well. A full comparison between these and other alternatives is recommended for the follow-on.

Reaction Wheel Sizing and Propellant Usage

The wheels need enough capability to accumulate angular momentum due to solar torque for up to 62 hours. This would be consistent with firing thrusters during a 10-hour window near perigee when imaging is not being done.

The expected magnitude of solar pressure would be 9 $\mu N/m^2$ for a reflective surface flat on to the sun at 1 AU. The total dimensions of the telescope plus the bus are expected to be roughly 2.8 by 6.7 meters according to Team X Structures. The effective surface area would be about 19 square meters, so the total solar pressure would be 0.17 milli-N. Because much of the vehicle mass is concentrated at one end (the bus end), the center of pressure will be offset from the center of mass by about 1.7 meters. The estimated solar torque would therefore be 0.29 milli-Nm. The accumulated momentum due to solar torque acting over 62 hours would be 65 Nms.

The thruster configuration was not worked out in detail during the Team X study. For sizing, it was assumed that the thruster moment arms would be 0.65 meters for those thrusters used to unload momentum accumulated due to solar torque. This is based on a bus diameter of 2.8 meters and width of 1.25 meters. It was assumed that thrusters would be fired as couples during momentum unloading in order to limit delta-V. For those thrusters used to unload momentum due to solar torque, it was assumed that they be aligned to fire perpendicular to the telescope boresight. This would carry exhaust gases away from the focal plane along the most direct path. If later studies show that contamination of the focal plane due to thruster exhaust is not an issue, then it might be possible to locate and orient the thrusters for a moment arm larger than 0.65 m.

For a 4 year mission, the total accumulated momentum due to solar torque would be 36,760 Nms. Assuming a 0.65-m thruster moment arm and an ISP of 180, the total propellant required to unload the momentum would be about 35 kg. A 20% margin would be recommended to support the possibility of an extended mission. That would bring the total propellant up to 42 kg for unloading momentum accumulated due to solar torque. The propellant allocation recommended by ACS to Team X Propulsion was 60 kg, to allow for the possibility of using thrusters during initial deployment and for safe mode.

The reaction wheels would need to be sized to carry up to 65 Nms accumulated due to solar torque over 62 hours. This would be the momentum storage capability of the set of wheels. Preliminary sizing was done assuming a pyramid configuration and one wheel failed.

The pyramid assumed would have each wheel spin axis offset by 45 degrees from a symmetry axis that passes through the apex of the pyramid. With all wheels in operation, the momentum capability in the symmetry axis would be 2.8 times that of a single wheel. The capability in either of the other two axed would be 1.4 times the capability of a single wheel. The symmetry axis could be oriented to optimize the momentum and torque capability in one particular direction fixed in the bus. Optimizing was not assumed during this preliminary wheel sizing.

In the event of a single wheel failed, the momentum storage capability in the symmetry axis would be 1.4 times that of a single wheel, and the storage capability in either of the other two axes would be 0.7 times the capability of a single wheel.

In order to store 65 Nms of momentum in the worst axis in the case of a single wheel failure, each wheel would need a momentum storage capability of 92 Nms. The Honeywell HR16 wheels in the baseline each have a maximum capability of 150 Nms, which leaves a margin of 63%.

On the other hand, this mission may require that the wheels be operated in a restricted speed range, so the wheels may also have a restricted momentum storage capability. The HR16 wheels have a speed range of from 0 to 6000 RPM. To carry 92 Nms, a wheel would need to run up to 3680 RPM. For comparison, SIRTF is not planning on operating reaction wheels above 1200 RPM.

For this mission, it may turn out that the reaction wheels do not have enough momentum storage capability to support nominal science operations for 62 hours in the case of one wheel failed. If so, one option would be to unload momentum more frequently. This might be an acceptable contingency approach since the chances of wheel failure are slight. Another option would be to carry 6 wheels instead of 4. A third option might be to talk to Honeywell about a version of the HR16 that can carry 200 Nms but has less torque capability. Honeywell produced such a version for MSX.

If all four wheels are operating, then to store 65 Nms in the worst axis, each wheel would need a momentum storage capability of 46 Nms. In that case, the max wheel speed would be 1838 RPM. If it hasn't already been characterized, the HR16 wheel still needs to be tested to determine the range of operating speeds for which disturbances are acceptably low. At this point, it seems reasonable that such a range might include speeds up to around 1800 RPM. So, the reaction wheels appear to be sized correctly for the nominal case in which all four are available.

One other factor in wheel sizing is the requirement to slew about roll through 90 degrees in tens of minutes. The inertia of the spacecraft about the roll axis is expected to be < 1500 kg-m^2. The momentum and torque required to achieve the slew would be small and are not drivers for wheel sizing.

SUBSYSTEM RISK

Low

LIST Reasons Why

Risk for nominal ACS functions is relatively low. High heritage algorithms, software, and hardware can be used to support spacecraft bus pointing control during initial deployment, reaction wheel momentum unloading, delta-V maneuvers, safe mode, and FGS acquisition. ACS includes full on-board redundancy and fault detection logic to guard against critical single-point failures.

Risk for ultra-fine precision pointing during imaging is medium. Careful design, analysis, system engineering, integration and test, and calibration will be needed to achieve the pointing control goals for the mission. The risk is that pointing stability might not meet the goal, resulting in some degradation of science.

3.4 CDS

Level 3 Requirements

- 18mon. phase A
- 12mon. phase B
- 48mon.phase C/D
- 2003 start of mission
- Store 375.2Gbyte of science data per orbit. Record one orbit.
- Down link data rate 300Mbps
- Provide computing and interface's to the ACS actuators and sensors
- Provide computing and interfaces to the Telecom system
- Provide monitoring and control for power conditioning, switching and pyro switching
- Provide command and engineering data collection for the telescope/instrument
- Goal to keep cost down

Functionality

The CDS is required to perform many critical spacecraft functions. Several examples are shown below:

- Uplink command processing and distribution
- Sequence storage and control
- Maintenance and distribution spacecraft time
- Collection and formatting of engineering spacecraft sensor data
- Bulk storage of science and engineering data
- Subsystem control and services
- Spacecraft system control services (non-attitude control)
- Spacecraft fault protection

Design Description

The following design assumes that a majority of the spacecraft buss is procured. We used the Mar05, Mars Reconnaissance Orbiter as a reference for the Avionics. The assumption is that the ACS, Telecom and Power elements are compatible with the Mars05 Avionics interfaces.

The complexity of this mission is thought to be less than the Mars05 mission. So the avionics has substantial margin.

The CDS design also follows the Mars05 approach of directing the high rate science data directly to the Solid State Recorder. The Instrument provides the paketization. When its time to downlink the data the SSR is directed to feed the stored data directly to the Ka Telecom transmitter.

Commanding and engineering data are transmitted and received via the redundant S-Band telecom transceivers.

The CDS support of ACS assumes that ACS does not need sampling and actuation rates greater than 10 hertz. The control system for the fast steering mirror is performed in the instrument. The instrument develops the low rate pointing change command and is transmitted to the low rate ACS spacecraft pointing function.

CDS provides the computing and interface to the High Rate Antenna Gimbal for gimbal control.

The Mars 05 CDS derivative consists of:

Block Redundant

- 2 each X2000 SFC
- 2 each X2000 NVM 2Gbit
- 2 each GN&C Interface (GIF)
- 2 each Uplink/Downlink (ULDL)
- 4 each Analog Acquisition (AAC)
- 2 each C&DH Module Interface (CMIC)
- 2 each C&DH Power Supply
- 2 each Data Telemetry & Cmd I/F (DTCI)
- 1 each backplane
- Chassis

CDS Enclosure

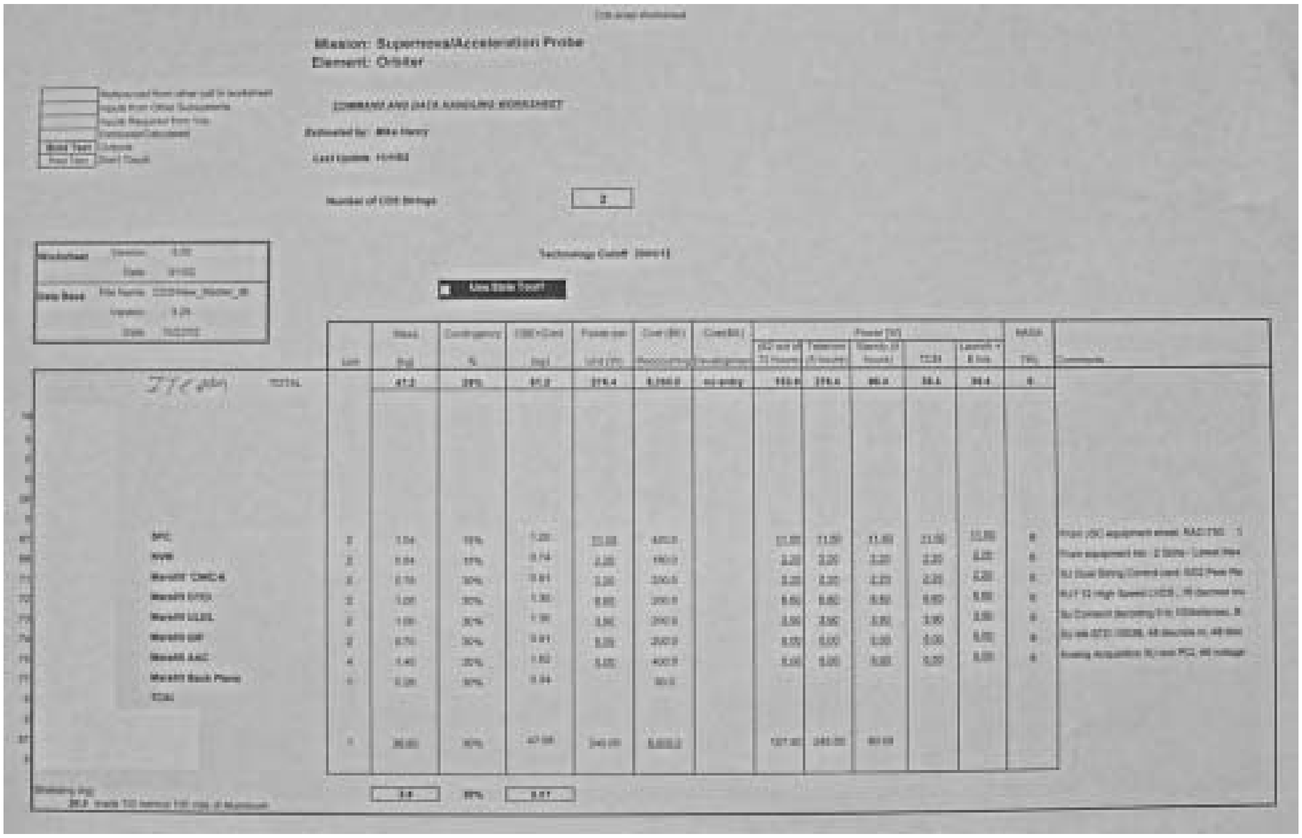
Each CDS string will have nine boards. There is a mix of 3U and 6U cards. The estimated mass of the CDS, including both strings and chassis is 61 Kg including a 30% contingency.

The predicted radiation environment is 30 krads with an RDM of 2 Behind 100mils of Al . Radiation hardened electronics parts are recommended for this mission. Electronic components should be Class S or Mil-Std-883B screened devices. The selected electronic components should have SEL and SEU immunity no less than 100 MeV/mg-cm^2.

The Power Subsystem will provide power supplies to the CDS strings.

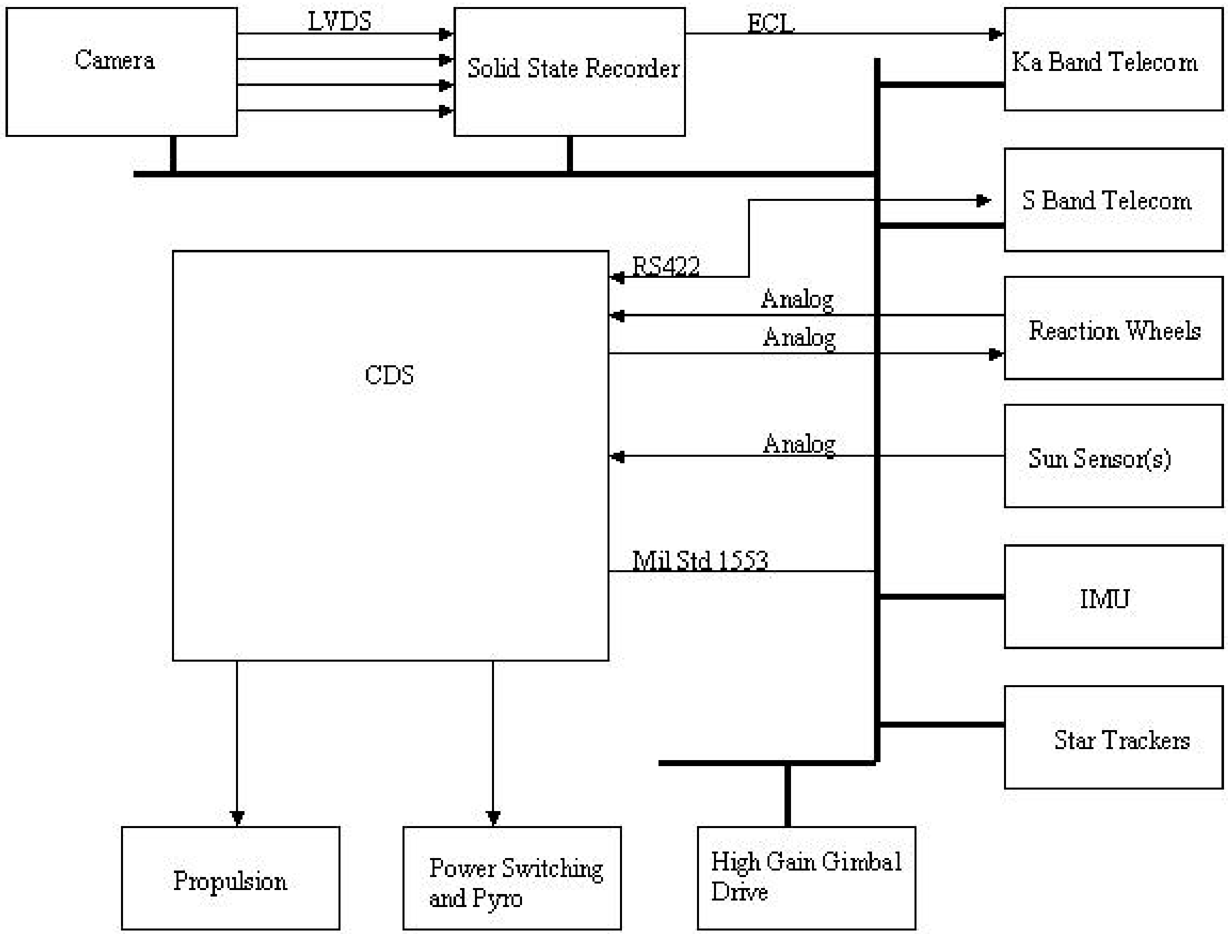
Mass and Power Estimate Summary

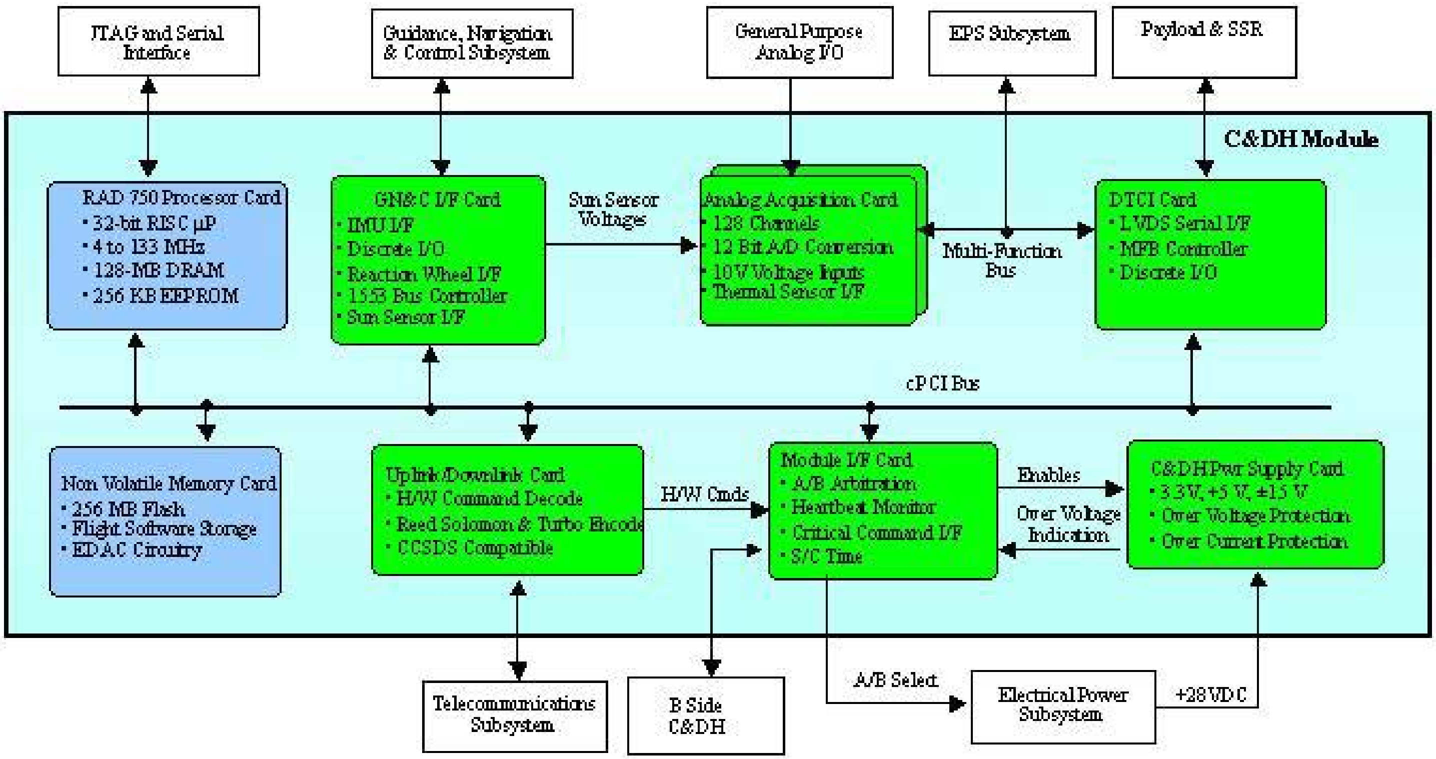
The mass, power and flight hardware recurring engineering estimates are shown in the table below. Table 6 is copied from the CEM distributed spread sheet.



Block Diagram

SNAP CDS Interface Block Diagram




JTAG and Serial Interface
Guidance, Navigation & Control Subsystem
General Purpose Analog I/O
EPS Subsystem
Payload & SSR
C&DH Module
RAD 750 Processor Card
• 32-bit RISC µP
• 4 to 133 MHz
• 128-MB DRAM
• 256 KB EEPROM
GN&C I/F Card
• IMU I/F
• Discrete I/O
• Reaction Wheel I/F
• 1553 Bus Controller
• Sun Sensor I/F
Sun Sensor Voltages
Analog Acquisition Card
• 128 Channels
• 12 Bit A/D Conversion
• 10 V Voltage Inputs
• Thermal Sensor I/F
Multi-Function Bus
DTCI Card
• LVDS Serial I/F
• MFB Controller
• Discrete I/O
cPCI Bus
Non Volatile Memory Card
• 256 MB Flash
• Flight Software Storage
• EDAC Circuitry
Uplink/Downlink Card
• H/W Command Decode
• Reed Solomon & Turbo Encode
• CCSDS Compatible
H/W Cmds
Module I/F Card
• A/B Arbitration
• Heartbeat Monitor
• Critical Command I/F
• S/C Time
Enables
Over Voltage Indication
C&DH Pwr Supply Card
• 3.3 V, +5 V, ±15 V
• Over Voltage Protection
• Over Current Protection
Telecommunications Subsystem
B Side C&DH
A/B Select
Electrical Power Subsystem
+28 VDC

NEW TECHNOLOGIES REQUIRED

No new technologies are required for the CDS

TRADE CONSIDERATIONS

1. If the customer would like a better probability of returning data we could add another ground station and or add another Solid State Recorder. The existing spacecraft design allows the total return of science data using 3 hours of a 5 hour telecom window. So if a pass was lost the stored data would come down in the extra 2 hour period for the two following passes. Each flight SSR cost $xM and requires 240W during data transfer and 120W during storage standby
2. JPL X2000 Avionics are always considered. X2000 interfaces to telecom and ACS are not standard. The cost of all the items to adapt to X2000 would be excessive.

SUBSYSTEM RISK

This is a fairly low risk CDS design. It will have been proven on the Mars O5 mission

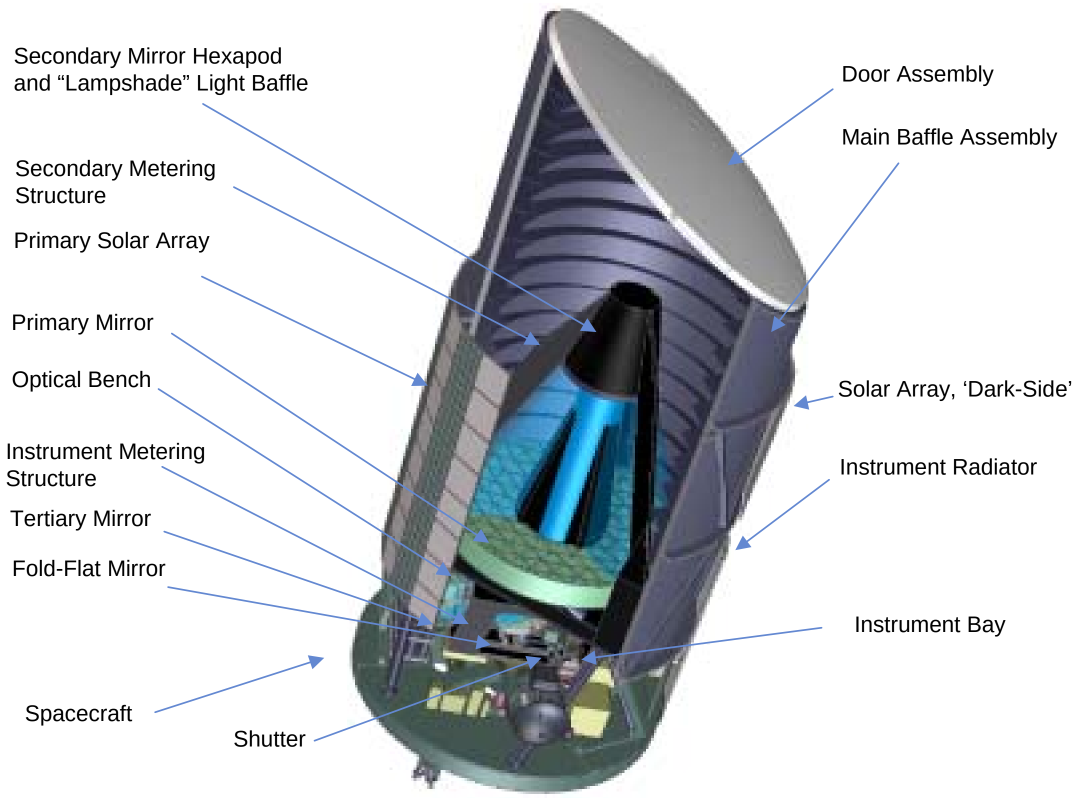
3.5 Power

Subsystem-Level Requirements

The spacecraft will be flying in a Sun-Synchronous High Earth Orbit (HEO). The spacecraft is expected to orbit earth once every 3 days (4320 minutes). The maximum eclipse time the spacecraft will experience is 3 hours (180 minutes). The power subsystem is designed to handle the spacecraft power requirements for mission duration of four years, as well as support the launch power requirements for six hours during spacecraft deployment.

Design Description

Figure 1. **Customer Concept of the Super Nova Acc. Probe spacecraft.**



The spacecraft power subsystem is designed to support the spacecraft power requirements during launch, deployment, and operation including eclipses.

Solar Array Design

Figure 2. **Spacecraft solar array design summary.**

Figure 2 summarizes the size and cost was for both solar arrays combined.

The solar arrays are designed to wrap around the instrument as depicted in Figure 1. The solar array technology chosen for both the primary and emergency (Dark side) solar array is GaAs triple junction (TJ) with 26.8% energy conversion efficiency. The solar arrays are designed taking into account the solar flux losses arising from the cosine and beta prime angles due to solar array/ spacecraft geometry and spacecraft orbit. The solar array optical surface reflectors (OSRs), which are designed to remove heat from the solar array, where not included in this study. Team X thermal analysis concluded that OSRs might not be necessary.

The primary solar array was designed to an end-of-life (EOL) power output of 616 W. The primary solar array is designed to provide the required spacecraft power during the “science mode” as well as the power required to recharge the battery after a “Telecom” mode or eclipse. The mass and area of the main solar array are represented in figure 2 and are 3.5 kg and 5.2 m^2 respectively.

The emergency solar array is designed to provide power to the spacecraft in the event that the spacecraft losses altitude control. The power rating of the emergency solar array is 441 W and was sized to provide power during the spacecraft “Stand by” mode. The emergency array is also designed to be able to charge the battery after a possible eclipse for added mission robustness. The mass and area of the emergency solar array are 2.51 kg and 3.71 m^2 respectively.

Energy Storage Device

FIGURE 3. **SPACECRAFT ENERGY STORAGE DEVICE (BATTERY) DESIGN SUMMARY.**

A 12-cell 90 Ahr common pressure vessel (CPV) $Ni\text{-}H_2$ battery will serve as the spacecrafts energy storage device. The mass of the battery is 53 kg as shown in energy storage device summary in Figure 3. The battery volume is estimated at 67.5 liters. The battery sizing was driven by a 6 hr “launch” mode deployment rule. The solar array was assumed to be in active during the spacecraft deployment. It was decided, in the Team X session, that spacecraft “telecom” mode would not be used during mission eclipse or this mode would drive the energy storage device sizing.

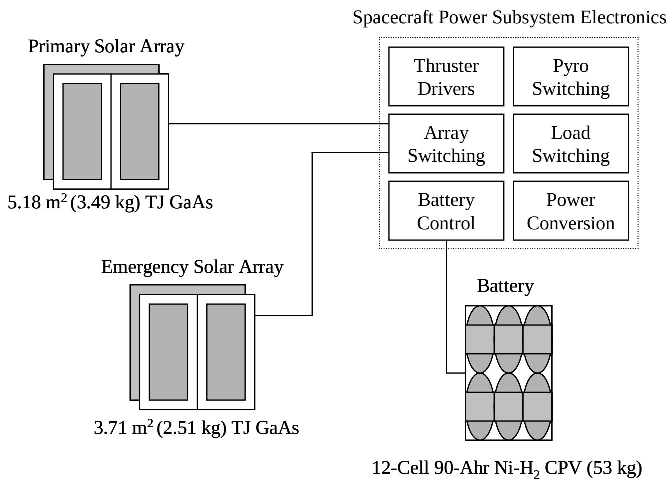
Power Electronics

Figure 4. **Spacecraft power subsystem electronics design summary.**

The spacecraft power subsystem electronics have been designed using a "grassroots" power electronics template. Solid state switching and COTS power conversion technology were chosen to satisfy the *off-the-shelf* component requirement. The power subsystem electronics is made up of 18 boards that include: array switching, pyro switching, thruster drivers, power converters, load switching, battery control, and power and control switching. The electronics are functionally redundant. As summarized in figure 4, the mass of the power electronics is 14.4 kg.

Block Diagram

Figure 5. **Power Subsystem Block Diagram**



Trade Considerations

Two power subsystem trades performed for this study revolved around the energy storage device (battery selection). Li-Ion as well as Ni-H_2 IPV chemistries and cell design have been considered.

Li-Ion

An 8-cell 100 Ahr Li-Ion battery with an energy capacity of 2800 Whr would be a good trade for this spacecraft. The mass and volume of this Li-Ion battery are 28.4 kg and 18 L respectively. The estimated cost from the Team X power subsystem tool is 918 K. Dual strings are required. Li-Ion batteries are only considered for three-year missions.

Ni-H_2 IPV (Individual Pressure Vessel)

A 23 cell 94 Ahr Ni-H_2 IPV battery is a possible trade for this spacecraft. The mass and volume for this IPV battery is 58 kg and 90 L respectively. The estimated cost for this battery is 1734K. An IPV battery is more robust than a CPV battery. However, An IPV battery is heavier and more expensive than an equivalent CPV battery.

SUBSYSTEM RISK

Low to Medium

Reasons Why

The current spacecraft solar array and power electronics designs are considered low risk because all of the components are based on *off-the-shelf* technologies that have flight heritage. System redundancy and operation configuration introduces some risk. Full dual string power electronics are preferred over functionally redundant. The CPV Ni-H_2 battery design can impair the mission with a single cell failure. Operating the "Telecom" mode on the battery could lead to risk in spacecraft safe hold modes.

3.6 Thermal

Subsystem-Level Requirements

The solar array is required to be below 70deg C. The back surface faces the telescope and heat flux to the telescope is to be minimized. This requires approximately 40 to 50% excess radiator area on the sun facing side of the solar panel free of solar cells. The material in this area has a high ratio of emissivity to solar absorption viz. Silver teflon.

The focal plane is to operate at 140 K and a passive radiator is required to reject this heat.

Design Description

The solar array is wrapped with multiplayer insulation in the back and the solar heat is radiated from the side that faces the sun. To maintain the temperature of the array below 70C, 40% of the area should be silver Teflon.

The heat rejected by the bus varies from 200W to 580W. This requires a fixed radiating surface of 1m2 and 5 standard 16 blade louvers of area equal to ¼ m2, to allow for heat rejections above 200W. They are mounted on the side that is antisun. The fixed radiating area is a part of spacecraft area with a treated surface.

The focal plane is to operate at 140 K and a passive radiator is required to reject this heat at 110 K.. Assume that there is a drop of 20K from the focal plane to the base of the radiator through the graphite straps. Ten conductive graphite straps lead heat from the focal plane to a radiator of size 1.3m x 1.3m facing cold space. The back surface is MLI covered. The radiator gains about 3 W from the spacecraft and has to radiate 17 W including this heat If the cold radiator faces the sun during manuvers or accidentally, the temperatures of the facal plane will rise to 217K because the radiator surface is covered with silver teflon. MLI blankets, heaters for tanks and propellant lines and conduction materials are used for thermal control to keep the temperatures in allowable flight limits.]

Design Assumptions

Target Planet	Deep space
Launch Date	12/1/09
Launch Vehicle	
Orbit	Highly elliptical earth orbit
MOI	
Final Orbit	
Mission life	8 yr
Communication	
Technology Cutoff	2005

Subsystem Risk

Low risk

Reasons Why

Passive cooling uses proven components

3.7 STRUCTURES

SUBSYSTEM-LEVEL REQUIREMENTS

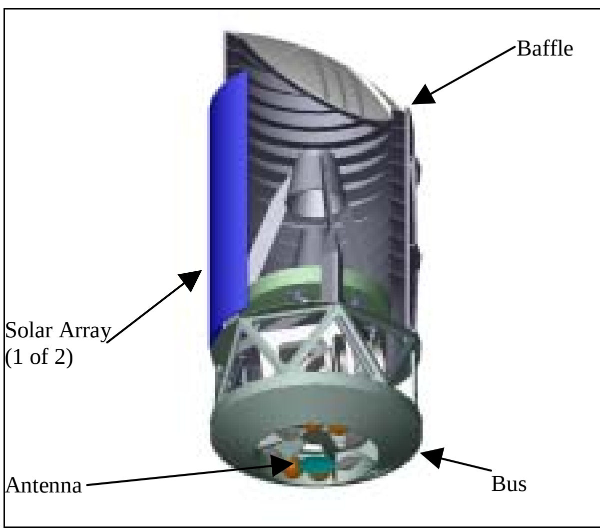
The primary purpose of the structures subsystem is to provide mounting and structural support for all spacecraft subsystems. In this case the principal structural supports are for the telescope and the baffle associated with it. In addition, the baffles need to support the solar arrays and the doors mounted on top.

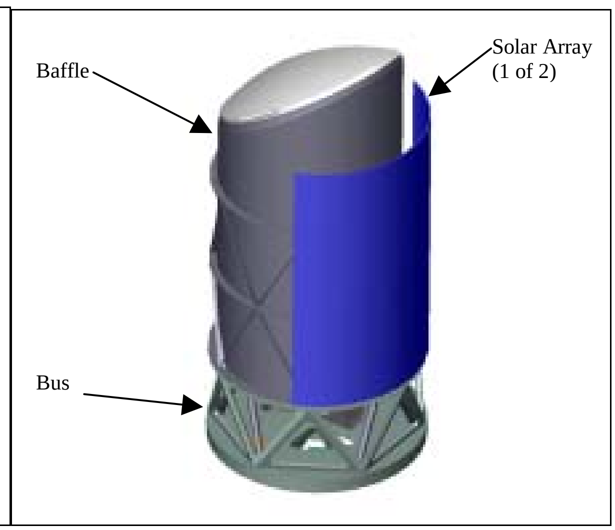
The second purpose of the structure subsystem is to account for major mechanisms. In this case the only mechanism of consideration is the antenna gimbals. According to mission design and Telecomm, the antenna is required to see the earth even when the spacecraft rotates 360° about the line of sight of the telescope and ±30° rotations about any other orthogonal axis. This is to account for the earth's relative movement during the orbit and during changes in seasons.

The entire configuration must fit on Delta IV with a 4-m fairing.

DESIGN DESCRIPTION

The largest element in the design is the telescope and is also the heaviest. The primary bus structure attached directly to the bottom of the telescope at the anchor points provided by the customer. The bus then has a stiff structural element to support the baffle assembly (including the solar arrays and doors.). Finally, the rest of the subsystems are scattered where appropriate within the remaining volume of the bus, with the antenna hanging off the bottom of the bus for clear viewing. See Attached figure.





TRADE CONSIDERATIONS

Structures did not evaluate any trades with in the subsystem. However, there were two important trades considered with other subsystems, one regarding offsetting the center of solar pressure and one regarding the antenna placement.

The first trade regarded offsetting the center of solar pressure. In essence, the there was approximately 1 meter spacing between the center of mass and center of pressure, which caused an increase of 70 kg in propellant to maintain attitude. (Details on this can be found in the ACS and/or propulsion section and are included here only in approximate terms for reference.) The goal was to deploy a structure that would move the center of pressure closer to the center of mass and thereby reduce the amount of propellant used. We estimate that such a structure could be made that weighs about 25 kg and another 15 kg in deployment and latching mechanisms. This is a slightly riskier option and therefore the three chairs concluded that we should carry the propulsive solution as the point design.

The second trade of concern regarded the placement of the antenna. Initially, the mission included a Star motor to de-orbit the spacecraft that was placed at the center of the bus where the antenna is shown in the figure above. This clearly prevented the antenna from being placed on the bottom center as it is now. This causes the antenna to be obscured fairly frequently during the mission and we had decided to mount three antennas around the bus. This makes both subsystems much heavier. The reason for dropping the Star was unrelated to this issue and therefore may become an issue again. (See propulsion section.) Another option is to deploy one antenna on a boom, but this required a significant amount of additional mechanism and structure, especially since the antenna has to re-stow before the Star burn.

DESIGN ASSUMPTIONS

For the Team-X Structures evaluation, a fresh-start spacecraft design is initially assumed to baseline the parametric description of an optimized spacecraft bus, as a point design for comparison. The structure mass is estimated parametrically using the TeamX linked estimating tools, based on the masses of the other subsystems which the structure supports, plus specific identified substructures, components and mechanisms. This estimate is, however, strictly parametric and does not initially include any configuration aspects.

For this mission, the initial configuration that was developed (discussed below) did not impose unusual geometric constraints, and thus the parametric mass estimates should be reasonably valid. The structure mass was estimated parametrically using the TeamX linked estimating tools, based on the masses of the other subsystems which the structure supports (including the Landers), plus specific identified substructures, components and mechanisms.

In case any commercial spacecraft bus might be considered for this mission, the characteristics of such a bus can then be assessed relative to this point design, and specific configuration studies can be performed with candidate launch vehicles and fairings. Use of a commercial satellite bus typically results in lower costs, but somewhat (or a lot) higher mass, than the point design due to the structural inefficiencies of adapting to a standard bus design and the inclusion of bus mass associated with capabilities which are in excess of those required for this specific mission.

The instrument subsystem provided a mass for the telescope and baffle based on the customer's MEL. Since the customer had bookkept a mass for the solar array panels, these were subtracted from the instrument mass and bookkept as part of Structures as is customary for Team-X. In the spreadsheet, the mass of the baffle was also broken apart to account for the fact that the mass to support it had not yet been accounted but the mass to support the telescope had been accounted for.

The solar arrays are fixed multi-panel honeycomb structures interlaced with Optical Solar Reflectors. The solar array mass is bookkept under Power, but the solar array panel structure is bookkept here under Structures.

The HGA is 1 m diameter and is pointed by a two-axis actuator; the drive and motor are bookkept under ACS, but the actuator mechanism and launch latch/release hardware are bookkept here under Structures.

MASS ESTIMATES

For the Spacecraft, the main bus structure is 9.70kg, plus 5.7kg for Interface and Integration hardware, 13.0kg for balance mass (even three axis stabilized spacecraft usually require some stabilization mass), and 20.2kg for the adapter structure between the spacecraft and the launch vehicle. There are 2 sets of solar array panels for a total of 10.8kg. Cabling is estimated at 42.4kg. The antenna gimbals requires 8.7kg of mass. (Not including the drive motor, which are bookkept by ACS.)

The Adapter and support structure for the baffle is estimated at 8.9 kg due to the length required to span in order to reach it. The telescope interface is 6.3kg, to cover the addition strengthening required to support it and the geometry associated with those attach points.

STRUCTURES EQUIPMENT AND MASS TABLE 1: MASS ESTIMATES FOR ORBITER BUS STRUCTURE

				Mass (kg)		
Materials:	Composites		Units	Best Est.	Contin-gency	CBE + Conting.
TOTAL (less Cabling, LV adapters)				**144.0**	**30%**	**187.2**
Primary Structure			1	81.0	30%	105.2
Secondary Structure			1	9.7	30%	12.6
Telescope Interface Structures			1	6.3	30%	8.2
Baffle support structure			1	8.9	30%	11.5
Solar Array Structure			2	10.8	30%	14.0
Antenna Articulation Mechanism		(2-axis)	1	8.7	30%	11.3
Integration Hardware & MHSE			1	5.7	30%	7.4
Balance Mass		(3-axis)	1	13.0	30%	16.9
Adapter, Spacecraft side	(not in mass total)		1	20.2	30%	26.3
Cabling				**42.4**	**30%**	**55.1**

A short general comment to the mass estimates: Primary structure supports the ACS, C&DH, telecom, bus-mounted power electronics and batteries, dry propulsion system, thermal, and instruments; and structural strengthening for carrying the liquid propellant mass and any stacked stages. Secondary structure allows for junctions, stiffeners, brackets and fittings; and solar array, antenna and other outrigger support (if any). Interface and integration hardware covers fasteners, shims, cable clamps, and such. Balance mass is normally bookkept at 1% of spacecraft dry mass for a three-axis stabilized vehicle and 2.5% if the vehicle is spin-stabilized or is launched on a spinning upper stage. The adapters are scaled to the (wet) spacecraft launch mass; if the launch vehicle is identified and adapter data is available, that value will be used for the launch-vehicle side adapter; in the case of a Delta, it is included in the stated launch vehicle capability.

Cabling harness mass estimate is based on historical factors times the mass of the separate electrically connected subsystems; as subsystems get smaller, the spacecraft bus gets smaller and the cable lengths get shorter, but all the connectors are still there. It is unrealistic to expect any mass optimization of cabling; in fact, the cabling harness mass is historically under-estimated by projects. The customer MEL included 10 kg for cabling. This was assumed to be cabling between the various components of the telescope and there for structures carried another factor to connect the telescope to the various subsystems.

Structures and Mechanisms equipment list and mass tables (including Cabling harness) are compiled with the other subsystems lists in the Appendix.

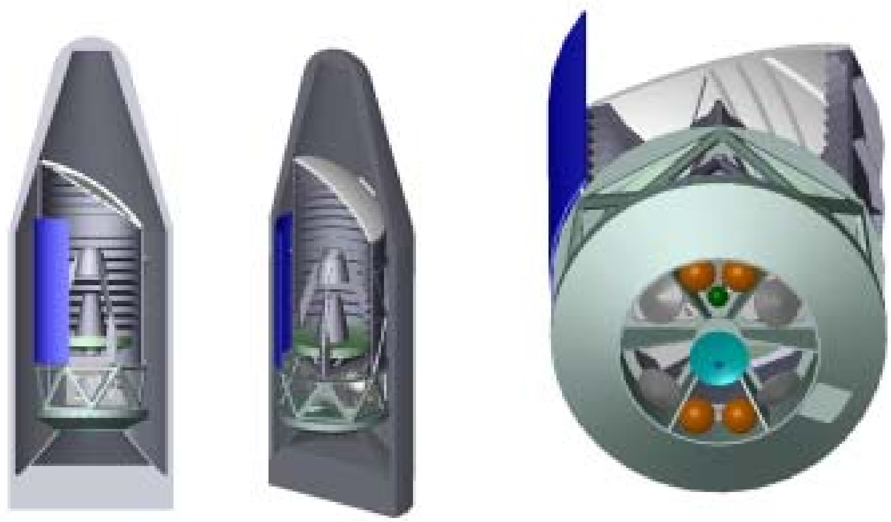
SUBSYSTEM RISK

Structures technology risk is basically low per usual Structures design philosophy; assumed to employ current best technology, standard design procedures and testing, and appropriate stress margins.

Bus structure and mechanisms are almost always new specific designs and thus should be set at 30% mass contingency in accordance with JPL design guidelines. Cabling mass contingency should also be 30%.

CONFIGURATION

Intent of configuration graphics is to show the major physical components and spatial relationships, within the study constraints of time and workforce. Numerous details (hopefully none pivotal) will not be able to be included in the graphics. Most of the graphics are from the customer's own CAD files. The major Team-X modules (Antenna, Solar arrays, Reaction wheels, etc.) were updated to reflect the size as determined in the study.



Two pictures of the spacecraft are included above. Show here are (left to right) a front cut away view of SNAP in the Delta IV launch fairing; and isometric cut-away view of SNAP in the Launch fairing; and a bottom view showing the position of the large elements in the bus.

3.8 TELECOM

SUBSYSTEM-LEVEL REQUIREMENTS

1. The downlink bit error rate (BER) shall be 10^{-6} or better.
2. The uplink bit error error rate (BER) shall be 10^{-5} or better.
3. The coding is 7 ½ convolutional concatenated with Reed-Solomon.
4. The data margin and carrier margin of all telecom links shall be at least 3 dB.
5. The communications system shall support a science return rate of 200 Mbps; the engineering telemetry rate is 32 kbps.
6. The communications system shall support command rates of 2 kbps (nominal) and 1 kbps (emergency).
7. The nominal ground station for Ka-band is a new 12 m reflector at Berkeley; the nominal ground station for S-band is the existing 10 m reflector at Berkeley.

DESIGN DESCRIPTION

The telecom subsystem includes S-band up/downlink capability for command, engineering telemetry return, and ranging, and Ka-band downlink capability for science return. The S-band portion consists of two Spaceflight Tracking and Data Network (STDN) transponders with standard 3 W transmitters, diplexers, and four LGA's to accommodate the varying orientations of the spacecraft. The Ka-band portion includes two modified X-band telemetry transmitters, two 4 W SSPA's, and a 0.5 m HGA.

The communication periods occur during perigee when the spacecraft is within a reasonable range over Berkeley. No science occurs during the communication passes. The passes last approximately 4 hrs. at a downlink rate of 240 Mbps (including coding and framing overhead). A gimbal is used to slew the HGA by approx. $\pm$ 5° from nadir during a pass. The max. range during a pass is 60,000 km. For one of the four hours, engineering telemetry is returned via the S-band link at 32 kbps; the spacecraft can be commanded at 2000 bps. For safemode during the science portion of the orbit, the spacecraft will orient one of its four S-band LGA's to Earth and return data at 10 kbps; the spacecraft can accept commands in this mode at 1000 bps.

Figure A-1 shows the Ka-Band high rate link at 60,000 km. The data rate is 240 Mbps into a new 12 m Berkeley ground station. Coding is a constraint length 7, rate ½ convolutional code concatenated with Reed-Solomon encoding. The BER is 1E-06 and the link margin is about 4 dB. Though not explicitly required, QPSK modulation will be used for bandwidth efficiency.

Figure A-2 shows the nominal S-band downlink at 60,000 km. The data rate is 45 kbps into the 10 m Berkeley ground station. Coding consists of a constraint length 7, rate ½ convolutional code concatenated with Reed-Solomon encoding. The BER is 1E-06 and the link margin is about 10 dB. Though the data return capability exceeds the need, it was judged more cost efficient to use the standard product line radio rather than introduce costly modifications.

Figure A-3 shows the nominal S-band command uplink at 60,000 km. The data rate is 2 kbps through the spacecraft LGA and STDN transponder. The data is uncoded and the BER is 1E-05. The link margin is about 10 dB.

Figure A-4 shows the safemode S-band downlink at 164,600 km (max. range). In safemode the spacecraft will align one of the four LGA's to Earth (Nadir pointed) using an Earth sensor. When this done, the Berkeley ground station will be less than 5 deg. off the antenna boresight at max. range. The achievable data rate is 10 kbps. Coding consists of a constraint length 7, rate ½ convolutional code concatenated with Reed-Solomon encoding. The BER is 1E-06 and the link margin is about 8 dB.

Figure A-5 shows the emergency S-band command uplink at 164,600 km. The data rate is 1 kbps through the spacecraft LGA and STDN transponder. The data is uncoded and the BER is 1E-05. The link margin is about 4.5 dB.

TELECOM HARDWARE

The table below contains a summary of the telecommunications hardware.

Component	Flt Units	Total Mass (kg)	DC Power (W)	Description/Comments
Ka-band HGA, 0.5m diameter	1	0.7		Gain = 42dBi
S-band LGA	4	1.2		-3dB beamwidth ~ +/- 45o
S-band STDN Transponder	2	5.0	40.4	COTS
Ka-band Transmitter	2	7.8	45.0	
Ka-band SSPA, RF = 4W	2	1.0	20.0	
Additional Hardware	10	3.000		includes diplexers, CXS, attenuators, WGTS, rotary joints, coax/WG
Total	**21**	**18.7**		

TELECOMMUNICATIONS SYSTEM MASS AND POWER

The Ka-band transmitter is assumed to be a modified L3Com/Conic X-band QPSK transmitter. The internal up-converter must be redesigned from X-band to Ka-band. The X-band power amplifier is replaced with a Ka-band drive amplifier. This accommodates the use of an external SSPA. It is assumed that the DC power consumption remains the same as the unmodified transmitter.

The Ka-band SSPA must be developed. The latest deep space Ka-band SSPA was flown by DS-1. However, the output of the DS-1 Ka-band SSPA is not adequate to meet the needs of this mission, and the parts are likely not available. Therefore, a new SSPA must be designed to accommodate the 4W output. It is assumed that an efficiency of 25% is achievable.

Power Mode #1 (Science only): S-band transponder is in receive mode = 5.4W
Ka-band transmitter is off

Power Mode #2* (Telecom): S-band transponder is in receive/transmit mode = 8.4W (40.4W @ 20%)

Non-science telemetry transmitted for one out of five hours.

Ka-band transmitter is on = 65W (modulator = 45W, SSPA = 20W)

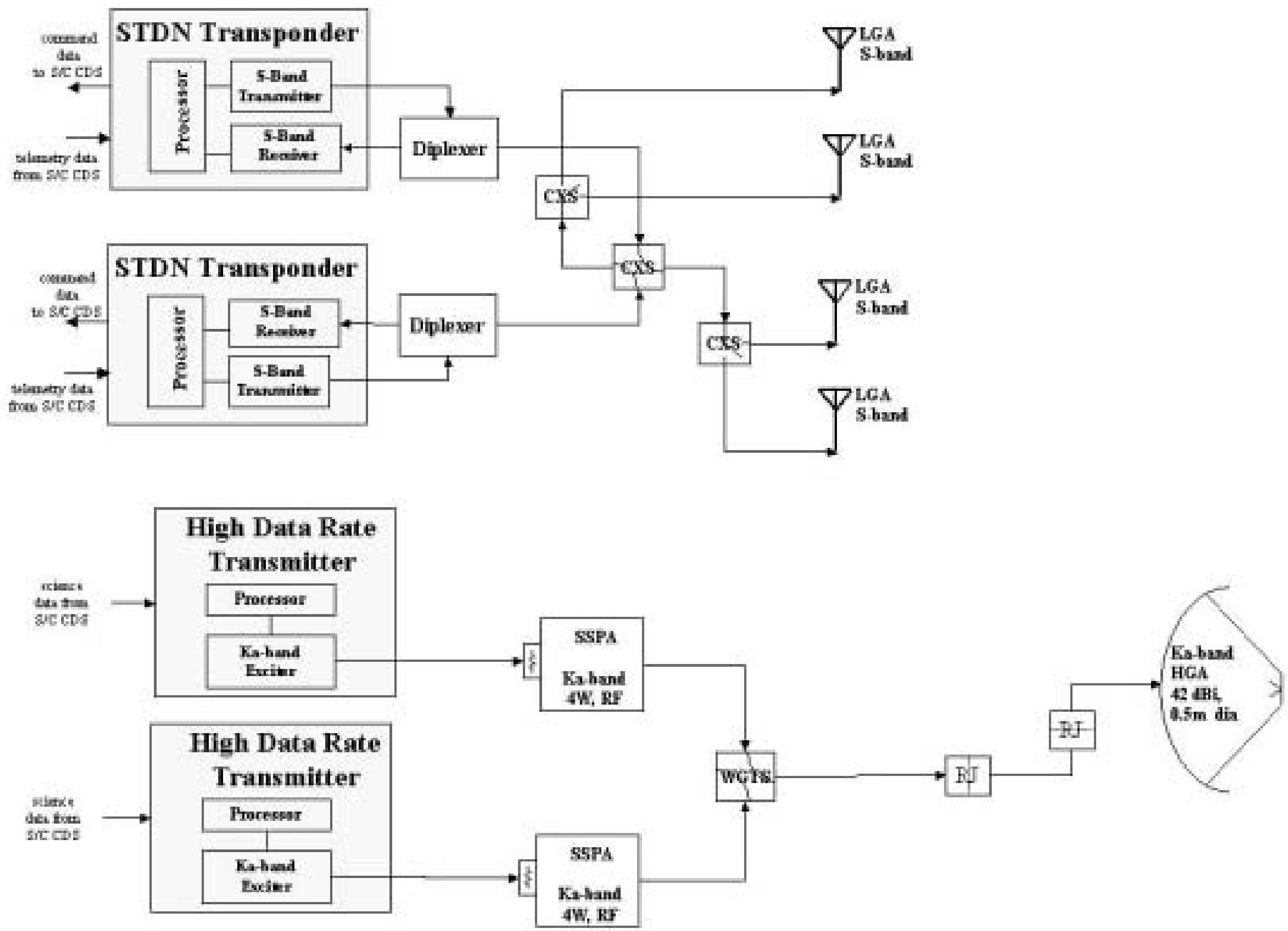
Power Mode #3 (Standby): S-band transponder is in receive mode = 5.4W
Ka-band transmitter is off

Power Mode #4 (TCM): S-band transponder is in receive / transmit mode = 40.4W
Ka-band transmitter is off

Power Mode #5 (Launch): S-band transponder is in receive / transmit mode = 40.4W
Ka-band transmitter is off

*After post-session analysis, it was determined that the receiver must stay on during the entire period of Mode #2. A straight 20% average of the receive/transmit power is not appropriate. Therefore, the power consumption for Power Mode #2 should be 12.4W.

BLOCK DIAGRAM



TRADE CONSIDERATIONS

I. The first trade considered the use of optical communications versus traditional RF communications. It was decided that optical comm. is not a better choice over RF. A flight optical transmitter is not commercially available. Ground station diversity is extremely important for optical comm. This defeats the desire of operating a single ground station on site.

II. The second trade considered the use of a Ka-band phased array amplifier/antenna assembly versus an articulated HGA. The trade was considered because of the desire to limit as many moving components as possible. The design of an adequate Ka-band phased array to cover the slew angle of 20 degrees would have been very challenging. This would have driven the cost very high. The ACS subsystem decided to allow a gimbal. This eliminated the requirement for a phased array.

SUBSYSTEM RISK

MEDIUM

This subsystem requires an engineering development for two components.

TELECOM HARDWARE COST

Total Cost = $xx.xM

Total Work Force = 37.5 FTE

COST ASSUMPTIONS

- Phase B = 1 year
- Phase C/D = 3years
- Work Force
- Management (PEM) - 4.0 FTE (phase B,C/D)
- Telecom Systems Analysis – 14.4 FTE (phase A,B,C/D)
- Antenna CogE – 3.5 FTE (phase B,C/D)
- Antenna Subsystem Support Engineering – 1 FTE (phase C/D)
- RFS CogE – 4.0 FTE (phase B,C/D)
- S-band Transponder CTM – 1.75 FTE (phase B,C)
- Ka-band Transmitter CTM/CogE – 2.25 FTE (phase B,C)
- Ka-band SSPA CTM/CogE – 2.25 FTE (phase B,C)
- RFS Support Engineering / ATLO – 4.2 FTE (phase C/D)
- Full spares included for S-band Transponder, Ka-band Transmitter, Ka-band SSPA

3.9 SOFTWARE

SCOPE

This report covers two major areas of the software efforts: the spacecraft flight software (or FSW below) and the project software engineering.

The flight software part includes, for example, software required to interface, command and control the various subsystems of the flight system (e.g., AACS, CDS) and instruments and payloads (though only the part that is not embedded in instrument or payload hardware). The term "flight software" here not only covers the flight software design and development, but also covers management, system engineering, test bed, fault protection, integration and test effort required to complete the flight software product. However, it does not cover science investigative and other software delivered with the instruments or payloads (which is covered in Instruments/Payloads report), control algorithm design and analysis (covered in AACS report), or the operating system and device driver level software effort (covered in CDS report).

The project software engineering part includes, for example, development of software policies and practices, software requirements, design, implementation, test issues, flight/ground tradeoffs, and project interface to independent verification and validation (IV&V).

MISSION SUMMARY

This Supernova/Acceleration Probe mission, or SNAP in short, is summarized below:

- The SNAP mission will launch in December 2009. During its 4-year mission lifetime, the satellite will measure up to 2,000 distinct supernovae each year. This satellite is a space-based telescope with a one square degree field of view with one billion pixels.
- The mission class is determined to be A/B.
- 2-meter aperture mirror, 1° degree half-million pixel camera, optical and IR spectroscopy**FLIGHT SOFTWARE DESIGN ASSUMPTIONS:**

The flight computer on the carrier is the main and only flight computer that controls the whole spacecraft. The flight software development is assumed to follow general JPL software development practices, including, for example, using C++ on RAD750 with VxWorks or similar real-time operating system. We further assume no FSW framework is employed (such as MDS or RTC whose cost structure is not well defined, but could mean a higher cost amount in general.)

Below we consider a few FSW cost drivers:

- Pointing:
    - The telescope should always point away from the sun's glare and the moon's glow.
    - The spacecraft attitude control system should keep the system stable in attitude to about 10 milli-arcseconds RMS one sigma one axis for a time interval of about 300 seconds while each deep exposure is taken.
    - To assist the spacecraft ACS system in this task, fine guider signals will be generated in a sensor array on the focal plane and transmitted to the

spacecraft. A fine-motion image motion compensator (located within the telescope - probably at the secondary mirror) may be required.

  - Entire payload needs to be stable < 0.01 arcsec RMS in pitch and yaw, and < 0.2 arcsec in roll, during science exposures of up to 1000 seconds duration. We intend to use payload fine guidance sensors for this purpose.
  - Feedback from the focal plane is thought to provide sufficient pointing accuracy so that nothing special needs be done with the sensors.
  - Uses a full set of standard ACS devices: IMU, Star Trackers, Sun Sensors, Reaction Wheels and so on. In addition, SNAP uses GPS as well.

- Data collection scenario:
  - Telescope in High Earth Orbit (or highly elliptical orbit) - repeatedly scans a survey region of approx 7.5 sq degrees, for 16 months; repeated in a second survey region another 16 months.
  - On-board data handling: customer's general guideline is to send all data straight down to ground for processing. We assume on-board data compression is employed nevertheless, since this would largely reduce downlink data rate required at a low cost.
  - Data Rates (based on SNAP public documentation):
    - General guideline: maximum rate is 300 M bits/sec
    - Focal plane average data rate during an observation is 40 M bits/sec (144 CCD arrays with 1600x1600 pixels, 44 HgCdTe arrays with 2000x2000 pixels, 16 bits per pixel, 220 seconds to complete observation plus read out focal plane)
    - Average data rate during spectrograph observation is 73 K bits/sec (one HgCdTe array with 1000 x 1000 pixels, 16 bits per pixel)
  - Data collected per orbit (assuming about half focal plane observations and half spectrograph observations, assume data compression by factor of 2)

Focal plane telemetry per orbit	2.1 T bits
Spectrograph telemetry per orbit	4.0 G bits
Housekeeping telemetry per orbit	47 M bits

If focal plane data taken all the time, and data compressed by a factor of two, total data collected during orbit is about 4.2 T bits.

- Instrument: The instrument consists of a telescope and associated focal plane equipment, which includes a wide field imager and a spectrograph. The telescope is maintained at a temperature of about 290K with the detector package passively cooled to 140K. NIR Sensors: HgCdTe detecteors
  - CCDs
  - Electronics: ASIC development is required

- o Filters
- o SpectrographMiscellaneous:
- o In-flight reprogramming requirements: Customer does not anticipate extensive on orbit reprogramming of the instrument. They will, however, maintain the capability to load a complete set of instrument software and operating tables (approximately 6 Mbytes), and we may include a capability to reprogram FPGA's. Note that the primary high-speed science data handling and memory management will be done in hardware circuits.
- o Development phase C/D: 4 years.
- o Mission operation phase E: 4 years.

FLIGHT SOFTWARE COST ESTIMATION BASED ON ANALOGY:

We estimate the SNAP FSW cost by using analogy from two missions: SIRTF and MRO. We shall use SIRTF as our base since it is close enough in many ways, and we shall use MRO as a reference so we know our estimate is reasonable from another perspective.

- SIRTF mission is similar to SNAP in both the mission scenarios (they are space-based telescope missions) and flight software functionalities. Here we note the major difference in terms of the FSW cost drivers:
  - o Pointing: SNAP's 0.03 arc-seconds pointing accuracy is higher than SIRTF's 6 arc-seconds. However, SIRTF's instruments have no mirrors and thus need the spacecraft to turn to point each one at the desired direction one at a time. Secondly, SIRTF's cryogenics system is very sensitive to Sun (heat) and thus need to be pointed carefully all the time. Lastly, SIRTF is an Earth-trailing Heliocentric orbiting mission, while SNAP is an Earth orbiting mission that would take the advantage of GPS system. We thus consider the pointing requirements of both missions are comparable.
  - o Data Processing: both missions take image data (visible and/or infrared) with high data rate – SNAP's data rate at about 30-40 Mbps is somewhat higher than SIRTF's 2.2 Mbps. Both missions do not process data on board other than the loss-less data compression. In terms of overall data processing complexity both missions should be comparable.
  - o Payload operations: SIRTF has to operate the Infrared Array Camera (IRAC), the Infrared Spectrograph (IRS), and the Multiband Imaging Photometer for SIRTF (MIPS) and the associated cryogenics system. SNAP's instruments, as listed above, are different but no harder, and it does not have the cryogenics system to worry about. The overall complexity should still be similar, or SNAP could be slightly simpler.
  - o One major difference could be in the non-technical aspect: the SIRTF mission suffers managerial problems with its contractor, such as unexpected personnel turn over, coordination and bad assumptions about multiple development organizations, among other things. The overall cost increase due to such factors could be significant, which we shall estimate at 30% as there is no way to get the exact figure.

- Taking all the above into consideration, we shall use 70% the current SIRTF cost as our first estimate.
- The SIRTF FSW cost is about $xx.xM in 2002 dollar, so the SNAP cost is estimated as $xx.xM x 70% = **$xx.xM**

- The MRO (Mars Reconnaissance Orbiter) mission is more deviated from SNAP as it has different set of mission scenarios, but its flight software functionalities (the static building blocks) appear to be very similar to that of SNAP's. For this reason, we thought it might serve as a good reference mission. Here let's also consider the few cost drivers that may be important:
  - Pointing: the MRO mission requires a pointing accuracy at 0.124 arc-seconds and stability at 0.124 arc-seconds over 3 ms (or 0.185 arc-seconds over 12 ms based on its HiRISE instrument). SNAP requires an even higher accuracy at 0.03 arc-seconds. We don't have the requirements on the pointing stability, but it should be very high as it is in the imaging mode most of the time while sending data to ground.
  - Device control, both missions have a full set of ACS devices including reaction wheels and optical navigation camera. Though MRO has more instruments (HiRISE, MCS, SHARAD, CRISM, MARCI, Context Imager), while SNAP has GPS and Earth sensors.
  - Data processing: SNAP has a higher data rate requirement.
  - Overall the SNAP FSW is somewhat more complex than MRO FSW. We assume about 15% higher than MRO.
  - MRO FSW cost is estimated at $xxM in 2002 dollar.
  - We do not have a record on the MRO mission class, but based on our estimate it is likely either class A/B with cost underestimated (it is a lot cheaper than MPF, say), or it is really of class B+. To adjust it up to the comparable level, we add 30% to it.
  - Taking the above into account: $xxM x 130% x 115% = **$xxM**
- The average of the above is **$xxM**. Since this figure includes the "software development testbeds" that the customer requested to exclude (and bookkept in the overall "Testbeds" cost instead), we deduct that cost, by about 14% or $xM, to **$xxM** as our estimate for the total flight software cost for SNAP.
- We use the software cost modeling tool, COCOMO II, to calculate the cost based on an inflated software size from MRO's 47K to 47K*150%=70.5K SLOC. This modeling result is shown below. Note the range provided by this model: it is likely from $xxM to $xxM, with the most likely value $xxM (all including the "software development testbeds").

- The Project Software Engineering is estimated at 10% of the total flight software, or **$x.xM**. The range is from $x.xM to $xM.

4.0 COST AND PROGRAMMATICS

4.1 Programmatics

Subsystem-level requirements:

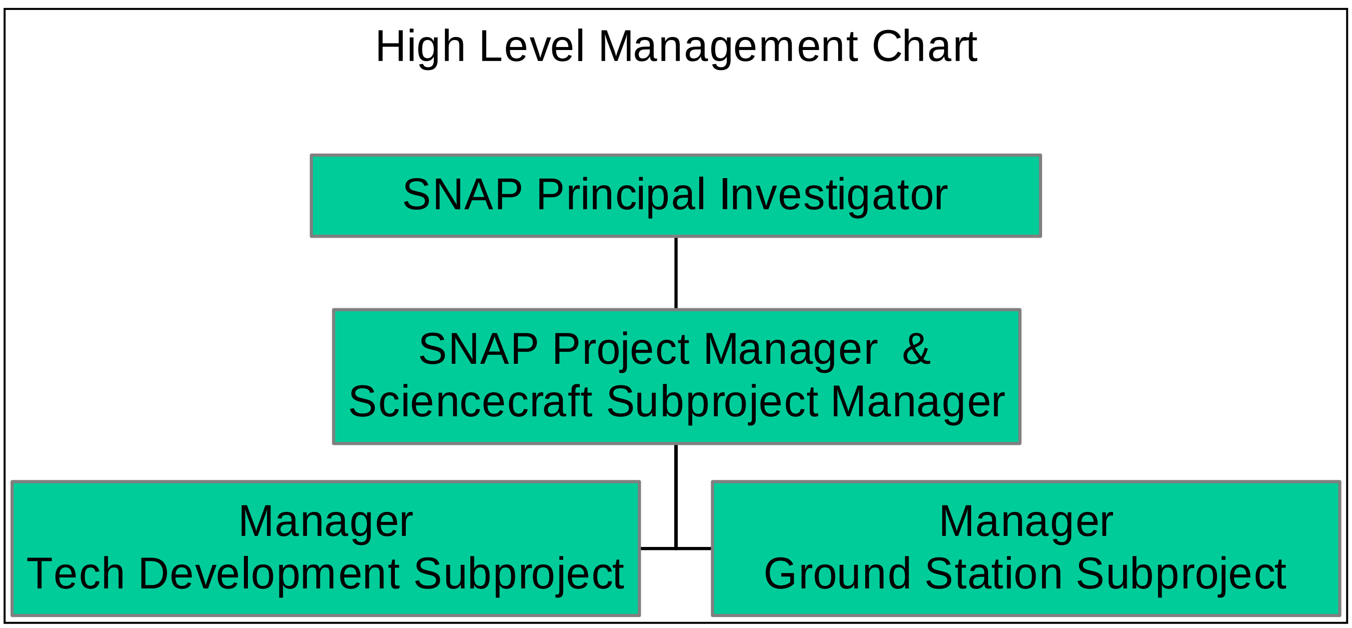
Programmatically this mission is 3 projects: A technology development subproject for the detectors, a sciencecraft subproject, and a ground station upgrade subproject. The overriding programmatic requirement is that each subproject be completed prior to launch.

Design assumptions/Drivers/Narrative description of design:

The Supernova/Acceleration Probe (SNAP) mission is intended to understand the “Dark Energy” acceleration of the expansion of the universe. This science mission will fly a large (~2 m diameter) telescope in high earth orbit to measure the brightness of certain supernovae and the red shift of their parent galaxies using imaging with filters and spectroscopy in the Visible Near InfraRed (VNIR) spectral region.

Certain enabling technologies will be necessary, and the existing ground station will need to be upgraded to receive the volume of data projected from each pass. In order to allow the technology development and ground station subprojects to receive their share of management attention, two additional managers were added to the budget, one to manage each of the additional subprojects. Sciencecraft Subproject Manager would essentially fulfill the role of SNAP’s Deputy Project Manager.

Figure 1 Suggested High Level Org Chart



Technology Development:

The detectors, focal plane and the ACS are challenging technology development items for this project.

- A NEW KIND of detector must be developed. The current schedule, even assuming that the development has been on going since the beginning of FY 2002, indicates that the detectors will not be available until the launch campaign.
- Focal planes near this size have been flown before, but none quite this large.
- The ACS must be very stable, requiring a new technology isolation system for the reaction wheels; it is estimated that it will take 1 – 1.5 years to develop.

The customer has allocated 8 years for detector development, starting one year ago. Margins included in this time span are unknown. In any case, the detector is assumed to be an exception to the rule that ALL technologies are to be at least TRL 6 by the Preliminary Design Review. If the development takes this long (which it may or may not), the detector will have to be delivered to the launch site. This is a less than optimum situation because test facilities at the launch site are less extensive than at I&T sites, so testing will be less comprehensive than otherwise. In addition, it can be difficult to make minor modifications in the bus to accommodate late instrument requirements at the launch site. **This is a HIGH RISK situation**-minor delays may become show stoppers. A later launch date might allow adequate schedule margin, and is recommended. This decision should be made no later than PDR.

Since technology development continues after the PDR, an additional manager is recommended.

SCIENCECRAFT:

The thermal demands of this mission are strict—but not unique.

Our scheduling database includes only recent successful JPL missions. These may not be applicable to this mission. However, they indicate that Phase A can be shortened by six months to save costs. Some costs may also be saved by starting implementation late for the spacecraft, except for ACS and thermal. Those subsystems will have to be on-board to work with the instrument development.

We treated Phase A as separate from the tech development effort for the purposes of our model and costing. Note the high risk of the detectors not being ready by launch.

Termination clauses need to be included in the contracts for the long-lead items--especially things such as glass for the mirrors.

It is recommended that an industrial partner be located before the beginning of the project, to avoid a 9-month schedule delay for a system-level procurement.

Detectors:

- LBL is doing the tech development on this. They will then hand the designs over to industry. It will be necessary to evaluate vendor capability of producing these in quantity and to flight quality. JPL has experience in this area and might like to be included on the bidder's list. It is expected that there will be a high defect rate, causing the loss of **many** detectors from each batch. The project should plan on buying enough additional detectors for testing and replacement needs.

Mirrors

- Glass for the mirrors: JPL thinks it will take about 2 years to shape the glass. This agrees with the customer's opinion.

Other development:

- The RF transmitter, while not below TRL 6, still needs development. This development is estimated to take about 3 years before delivery to beginning of ATLO. This could be a schedule problem.
- Ka-band Phased Array: This was dropped in favor of a gimbaled antenna, due, at least partially, to the 3-year+ development effort to bring this to TRL 6.

ORBITAL DEBRIS

Since the spacecraft is in a high orbit the only debris concern is that it sometimes crosses Geo-synchronous orbits. It was initially assumed that this would not be a problem. On checking, it was determined that while the spacecraft can cross the orbits indefinitely, there IS a requirement that it discharge all residual energy at end of mission.

GROUND SYSTEM:

Antenna development and location:

A 12-meter Ka band antenna needs to be built or retrofitted to an existing station. Clouds are a problem for this frequency, causing a 13% dropout at Goldstone—and Berkeley is MUCH wetter than Goldstone. However, when two antennae are used together (about 100 km apart), this decreases to 4-5% dropout at Goldstone. We can expect a similar improvement for a Berkeley antenna system, especially if the second antenna is located on the eastern side of the mountains, above the fog level. It is recommended that the possibility of a second antenna in a non-identical climate (where it might be raining on a different day) be investigated.

Downlink time

The reporting phase is 5 hours, of which 3.3 is used to download the data from one pass. This means that a missed pass will take more than one reporting phase to download. Therefore, back-up is recommended. USN (Universal Space Network) will provide back up services for a fee. Since they do not currently have Ka band, they might be persuaded to locate their first station where it is most useful to SNAP. SNAP might also check into the availability of JPL's DSN. It does not seem likely, but JPL might also provide back-up services for exchange in kind (allowing the DSN to use the antenna when SNAP is not reporting).

Data transport on the ground

The data downloaded to the ground station will be in tremendous volumes. The bandwidth needed to transmit the data from a remote ground station to Berkeley prohibits locating the ground station far from Berkeley. The project might consider locating the ground station(s) within driving distance (1-3 hours) of Berkeley (but in a dryer climate more appropriate for Ka-band). The data could then be transported via courier on physical disks.

Also, the Project should evaluate how industry is progressing in developing larger bandwidth capability. It may be that by 2009, the technology will be in place to handle SNAP's volumes of data, thus eliminating the requirement of locating the ground station at or near Berkeley.

SCHEDULE

The following sample schedule shows the technology development time requirements, assuming that each item is started as late as possible. The critical detector development time span was provided by the customer. An 18-month phase A is shown, followed by a 12-month phase B. These are longer than would be used on a typical JPL mission, but may be justified by the importance of the technology development in this mission.

The overall implementation phases are shown as 48 months from PDR to launch. To save money, only the ACS and thermal parts, and the systems engineers start on time. The rest of the project, predominantly the bus, will start a year after the early implementation portion. The parts come together at ATLO. The JPL margins from the Design Principles are incorporated into this part of the schedule.

The Ground Station development time spans shown are recommended by the JPL Team X ground systems seat, with the latest possible start date

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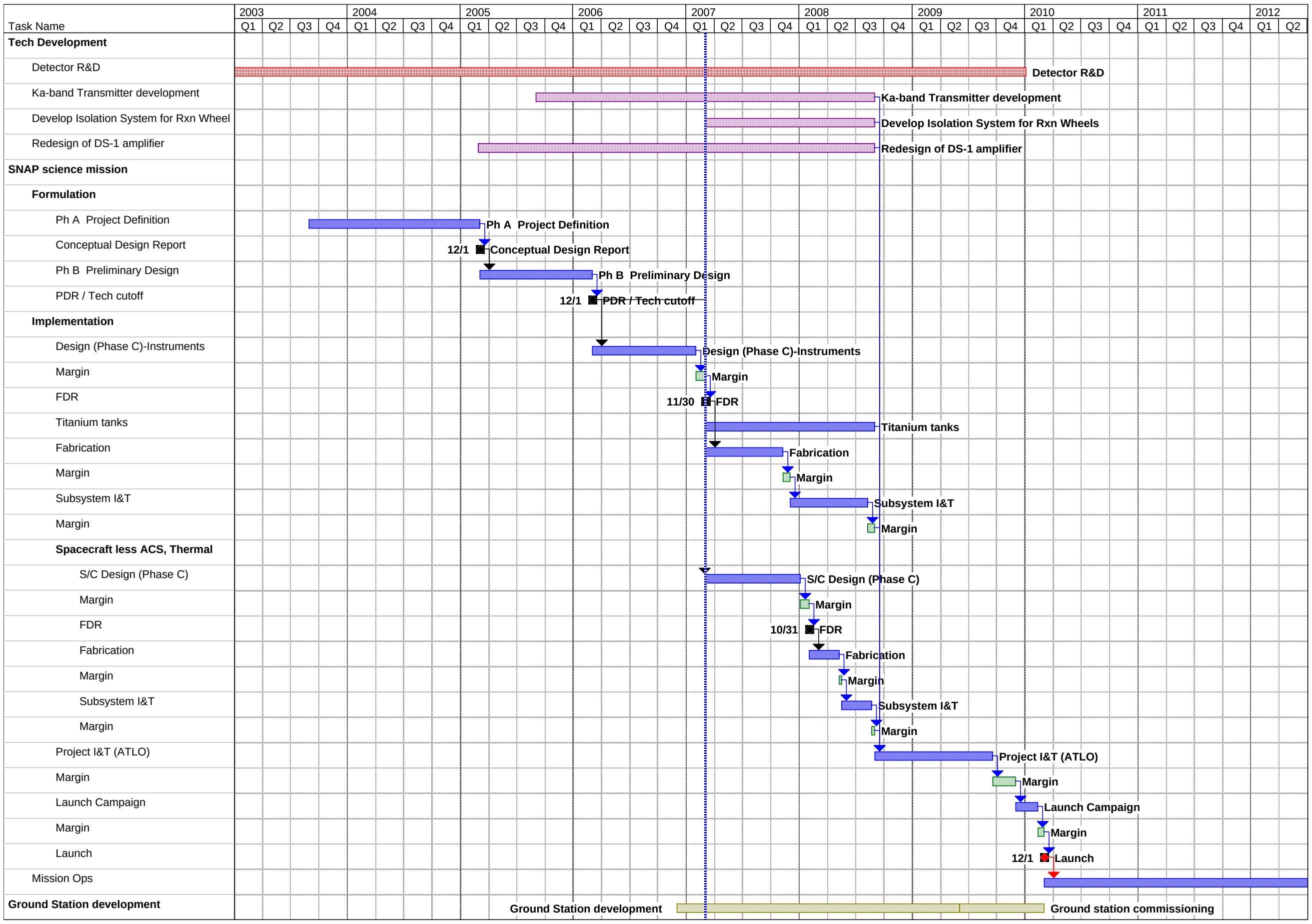
FIGURE 2. SAMPLE SCHEDULE FOR TECH DEVELOPMENT, SCIENCE MISSION, AND GROUND STATION SUBPROJECTS



TRADES CONSIDERATIONS:
The project can accept the risk that the detectors will not be ready by the projected launch date, or plan on a later launch to allow margin for unexpected problems in development. It is recommended that the project implementation phases be delayed until the detector development has reached TRL 6.

4.2 COST

COSTING METHODOLOGY

File Location for this study is under: \\PDC-NT1\TeamX\ACTIVE STUDIES\SPACE PHYSICS STUDIES\Super Nova-Acc.Probe(SNAP)2002-10.

STUDY OVERVIEW

- The SupernovaAcceleration Probe (SNAP) is a space-based large field-of-view system that will study the nature of Dark Energy and the expansion of the universe by measuring the redshift and brightness of distant type Ia supernova. Type Ia supernovae are stars that explode in thermonuclear cataclysms brighter than entire galaxies. They make ideal "standard candles" to survey the universe.
- The mission will have a single instrument containing a reflecting 2 meter aperture telescope with a 0.7 degree wide field-of-view measuring both visible and IR with 36 detectors/CCD. The telescope is "warm", not like SIRTF that is cryogenic cooled. A large passive radiator will be used instead.
- Lawrence Berkeley National Laboratory will manage both the spacecraft and instrument contracts. Both the flight system and the instrument are supplied by the industry, except the FPAs and the Optical train are GFE.

STUDY COST GUIDELINES

- All costs are reported in FY2003 dollars.
- Cost cap is $xxxM, excluding operations. Goal for this study is something much lower than the cost cap.
- Partially redundant system with heritage to FUSE and Class B parts.
- Launch Vehicle is a Delta 4040 with a cost of $xx.xM.
- Launch date is December 2009.
- Mission duration is 4 years.
- Mission Class is A/B.
- Technology cutoff date is December 2005.
- Radiation Total Dose is 30 krad.
- Phase A start is May 2003. Mission duration is 18 months for Phase A, 12 months for Phase B, 48 months for Phase C/D, and 48 months for Phase E. This information was used to develop the “Project Budget” provided below.
- Prototype units for the instrument, prototype and protoflight units for the spacecraft.
- Reserves are 30% for Phases A through E.
- Outreach is a NASA cost element. This is a Department of Energy program. Therefore, outreach is not applicable and there are no costs for this element.
- Project Mission Assurance is 3% of costs without Reserves for Development and 1% of costs without Reserves for Operations.

- The DOE breaks total mission costs into 2 elements, Construction and Operations. The total mission costs will be broken out according to the DOE format. . DOE Construction costs include development, launch, and mission ops ground hardware. It does not include pre-launch costs (ground system will move these costs to Phase E).

RESULTS/COMMENTS

- The total cost for the mission (Construction and Operations) is estimated at $xxxM with a range (-10%/+20%) of $xxxM to $xxxM. The costs include $xxxM for the Orbiter, $xxxM for the Instruments, and $xxM for the LV. These costs are shown on the Project Cost Summary sheets provided below.
- The ATLO estimate of $xxM is the cost for assembly, test and launch operations for just the Orbiter. The costs does not include IA&T for the instrument or instrument to spacecraft I&T. The instrument will be built by Lawrence Berkeley National Laboratory (LBNL) and the Orbiter will be sent to LBNL for integration with the instrument. LBNL has provided a cost of $xxM for the integration activity, which is assumed to include both instrument IA&T and instrument to spacecraft I&T.
- The instrument was not estimated by Team X. The cost for the instrument was provided by the customer as an estimate by LBNL for $xxxM total.
- Construction costs include Technology Development. The customer provided an estimate of $xx.xM which includes reserves. The Phase A costs determined by Team X were subtracted from the customer Technology Development estimate.
- The IPAO / HQ Reserve Model suggests 40-50% reserves for a mission of this type. This study carried 30% reserves for development. The model assumptions used include High complexity of hardware and software, High difficulties in systems engineering or integration & test, and Moderate for the remaining parameters. The model run with assumptions is saved in the \Super Nova-Acc.Probe(SNAP)2002-10\ directory for reference as HQResvModel-SNAP.xls.
- A review of the spacecraft database at http://kalel.jpl.nasa.gov:8088/spacecraft/menu.jsp provides two comparable missions with a dry mass and wet mass within a 40% variance of SNAP (1704 kg dry, 1967 kg wet). These include Mars Observer (1074 kg dry, 2457 kg wet) and Galileo (1156 kg dry, 2113 kg wet) with a total mission cost of $xxxM FY01 ($xxxM FY03) and $xxx5M ($xxxx FY03).

PROJECT COST SUMMARY (CONSTRUCTION COSTS)

PROJECT COST SUMMARY (OPERATION COSTS)

ELEMENT COST SUMMARY

PROJECT BUDGET

DETAILED SUMMARY OF COST ESTIMATES BY WBS ELEMENT

PHASE A & B
The costs for the early design and development of this project have been estimated as a percentage of the Phase A, B, C/D costs.

WBS 1.1: PROJECT MANAGER & STAFF
The costs for the management of the project as a whole have been estimated as a percentage of the WBS 2, 3, 4, and 5 costs.

WBS 1.2: LAUNCH APPROVAL
This estimate for launch approval is largely a placeholder, and should be replaced after further analysis with experts in that field. This placeholder is likely appropriate for fairly safe spacecraft being launched on well-tested launch vehicles.

This estimate for launch approval is largely a placeholder, and should be replaced after further analysis with experts in that field. This placeholder is likely appropriate for relatively risky spacecraft (e.g., carrying nuclear material) being launched on well-tested launch vehicles.

WBS 1.3: EDUCATION AND PUBLIC OUTREACH
The costs for public outreach for this project have been estimated as being 2% of the total project cost (without launch vehicle). The split between Phases C/D and E is probably not representative of the actual spending on this WBS element, which is likely heavily weighted towards phase E.

WBS 2.0: SCIENCE TEAM
The costs for the Science Team assume that this project will support the equivalent of XX full-time scientists for Phases C/D. During Phase E, the project will support the equivalent of YY full-time scientists. This is roughly one FTE per instrument for phase C/D (e.g., a full-time PI, or part-time PI and Co-I), and two FTEs per instrument for phase E (e.g., a full-time PI and Co-I, or a PI and several part-time Co-Is and graduate students).

SCIENCE TEAM COSTS
The Science Team costs for this mission in FY'03 dollars are $xx.x M for the development phases ABCD (these phases involve requirements, algorithm and observation planning development) and $xx.x M for the operations phase E (this phase includes observation execution and processing of data to the equivalent of physical units (calibration) and the validation of these data). These costs include a full time Project Scientist and a full time Science Office Manager. Not included are any "Guest Observers". Such investigations are contemplated but it is assumed here that they would take place in an extended mission whose costs were not estimated in this exercise. Also not included are costs associated with the actual archiving and distribution of the reduced (calibrated & validated) data produced by this mission. The calibration and validation costs themselves are included.

PLANETARY PROTECTION

This is a Category I mission according to the official NASA Planetary Protection guidelines, "NPG 8020.12B Planetary Protection Provisions for Robotic Extraterrestrial Missions." Category I includes missions to targets not of direct interest for understanding the process of chemical evolution.. This mission requires no planetary protection provisions. There are no planetary protection costs for this mission category.

WBS 3.0: PROJECT & MISSION ENGINEERING

Cost could be reduced by not having the repeat groundtrack orbit requirement. If the spacecraft node and inclination is allowed to drift unconstrained, and therefore is not required to downlink to Berkeley every pass, then the orbit maintenance delta-v will be less than the estimated 180 m/s. If no orbit maintenance is required and spacecraft is allowed to drift freely throughout its mission duration, then the total delta-v needed would be 50 m/s for launch vehicle injection errors.

WBS 4.1.1: S/C BUS MANAGEMENT

The costs of managing the development and implementation of the spacecraft bus have been estimated as a percent of the subsystem costs.

WBS 4.1.2: S/C BUS ENGINEERING

The costs for engineering the spacecraft bus have been estimated as a percent of the subsystem costs.

WBS 4.1.3: ATTITUDE DETERMINATION AND CONTROL SUBSYSTEM

This cost estimate is for pointing control, knowledge, and stability related to the bus and the telescope.

The estimate is in FY '03 dollars.

The estimate is for a Class A/B mission with high quality ACS parts.

The technology cutoff date is 2005.

The schedule for the project is as follows:

- Phase A is 18 months.
- Phase B is 12 months.
- Phase C/D is 48 months.

Spacecraft system engineering is assumed to take place over the full Phase A/B/C/D. Other effort related to the spacecraft build (excluding the telescope) is assumed to take place over Phase C/D only.

An industry supplier is assumed. However, the ACS parametric cost tool provides an estimate of what it would cost for JPL to build the spacecraft. An industry supplier is expected to have a lower cost due to higher re-use of existing designs. Consequently, this estimate includes the following discounts:

- 30% less effort for system engineering and control design and analysis. In other words, JPL would spend 43% more effort than an industry supplier.

- 15% less effort for integration and testing. JPL would spend 18% more effort.

Since there is an industry supplier, the estimate also includes $x.x M for Controls IV&V and ACS related system engineering and fault tree engineering. The breakdown below assumes that system engineering takes place over Phase A/B and Phase C/D and other efforts take place over Phase C/D only.

The cost of flight hardware, EM's and flight spares includes a 10% procurement burden.

The costs shown below were assumed for ACS hardware. These include the 10% procurement burden.

- Coarse sun sensors (8 flight units, 1 EM, 2 spares): $xxK
- Precision Star Trackers (2 flight units, 1 EM, spare electronics and focal plane): $x.x M
  - Spare electronics and focal assumed to cost 50% of flight unit price
- High Precision Gyros in an Internally Redundant IMU (1 flight unit, 1 EM, spare gyro, spare electronics and power supply): $x.x M
  - Spare gyro, electronics, and power supply assumed to cost 50% of flight unit price
- Precision Balanced Reaction Wheels (4 flight units, 1 EM, 1 Spare): $x.x M
- Precision Drive Motors for the HGA (2 flight units, 1 EM, 1 Spare): $xxx K
- Gimbal Drive Electronics (2 flight units, 1 EM, 1 Reserved Spare): $xxx K
  - Spare reserved for 30% of flight unit price (non-refundable deposit)

It is assumed that this ACS design has some similarity to a previous design but requires major modifications. The assumed levels of effort relative to a completely new design are as follows:

- System Engineering: 90%
- Hardware Engineering: 80%
- Integration and Test: 70%
- Control Design and Analysis: 40 to 50%

The total ACS cost is estimated for the first vehicle as $xx.x M. This includes the cost of IV&V. Non-recurring costs are estimated to be $xx.x M, and recurring costs are estimated to be $xx.x M.

The estimated levels of effort for the industry supplier to build the spacecraft are shown below. This assumes that system engineering takes place over Phase A/B and Phase C/D and other efforts take place over Phase C/D only.

If JPL were to build the spacecraft, the total for the above levels of effort would be $x..x M higher at $xx.x M.

WBS 4.1.4: Power Subsystem

TABLE 1. **POWER SUBSYSTEM COST BREAKDOWN.**

The total cost, in 2003 dollars, is xx,xxx K as shown in table 1.

WBS 4.1.5: Propulsion Subsystem

The total cost of this system is listed in the table below. Component costs are estimated based on database values, number of units, development if required, etc. Labor estimates are scaled based on system type and complexity.

The team X tool has a margin of error of –20%/+40%. This results in a total possible range of $x.xM to $x.xM.

WBS 4.1.6: STRUCTURES

Although the customer desires to have a commercial partner build a bus to attach to the instrument, the cost was estimated assuming Spacecraft bus will be built in-house, to DNP schedule and costs. This is because the commercial partner will likely have a similar process costing a similar amount. Costing was done in FY'02 $$, adjustment to real-year $$ to be done by Cost subsystem.

This is a Class A/B mission, requiring an selected EM spacecraft components. Costs for EM hardware are carried in Dev.Test category. Structure did not carry EM units for the Solar Array panels or the Antenna gimbals, as they are likely to be standard components. The remaining structure is new and requires EM units.

The Structure subsystem (structures, mechanisms, and cabling harness) cost totals $Mxx.x for non-recurring, $Mx.x for recurring, = $Mxx.x total. The mechanical I+T cost is $Mx.x for non-recurring and $Mx.x for recurring, or $Mx.x total.

I&T costs do not include telescope payload integration cost as customer is carrying those costs.

Team X baseline costs were originally developed for a Class B-/C+ mission. In recognition of the extra engineering effort, documentation, reviews and testing required for a Class A/B mission, the baseline Team X costs have been adjusted per the mission Class. The Non-Recurring total (but not the individual line items in the table) includes a 1.5 factor for the effects of this being a Class A/B mission, and the Recurring total a 1.15 factors.

Note that the costs in the table(s) are totalled to the **top** and **left.** The Non-Recurring costs include most of the engineering, the development testing, some of the flight hardware, and spares. The Recurring cost consists of most of the flight hardware, some of the engineering, and the qual. testing.

The Mechanical I&T costs, in the last line, are separate and not included in the Structures totals. These are a portion of what have traditionally been called Systems ATLO (assembly, test, and launch operations) costs.

STRUCTURES COST TABLE 1: COST ESTIMATES FOR BUS STRUCTURE, PHASE C/D, IN $M (INCLUDES DESIGN, TESTING, AND FLIGHT HARDWARE)

WBS 4.1.7: THERMAL SUBSYSTEM

Cost in $M

WBS 4.1.8: TELECOMMUNICATIONS SUBSYSTEM

TELECOM HARDWARE COST

The folowing table contains an estimate of the cost breakdown for both the UHF and X-band telecom hardware.

COST ASSUMPTIONS

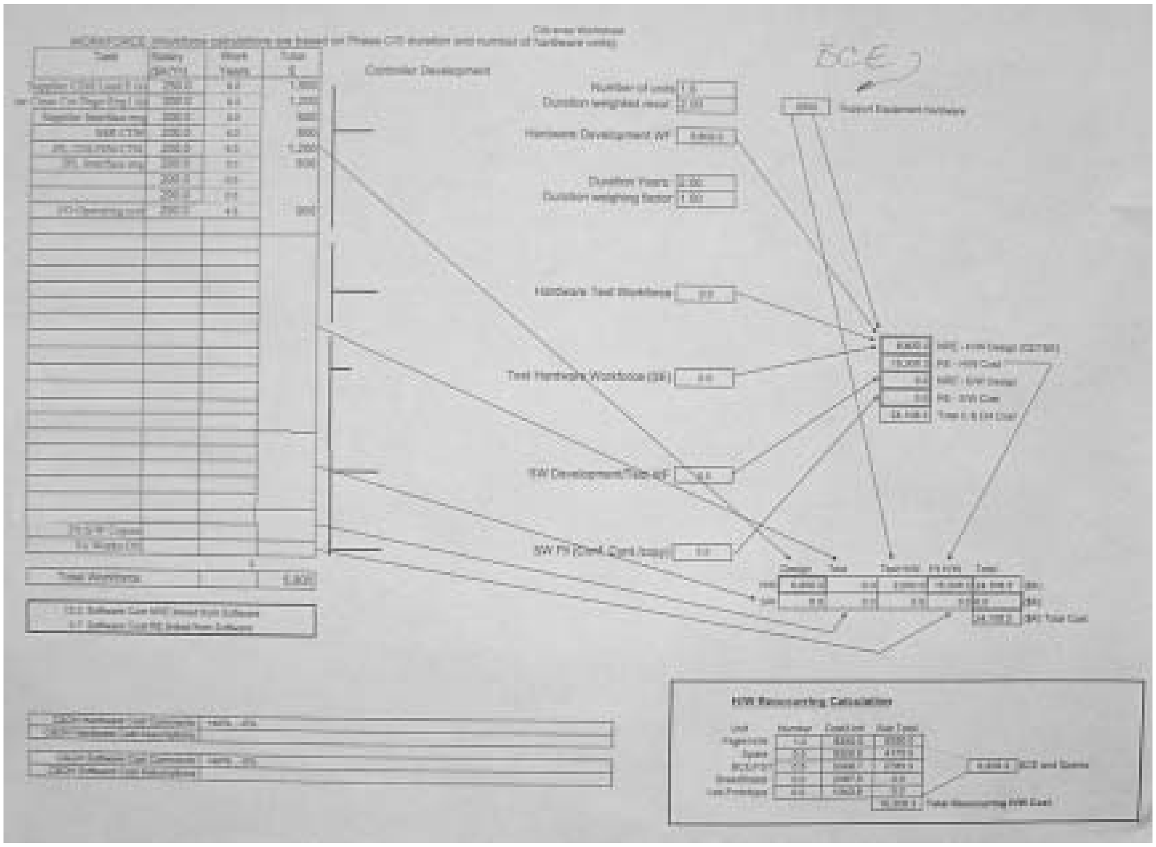
- STM NRE is paid by TMOD.
- No NRE for the UHF transceiver, TWTAs, LGAs, and UHF Antennas is assumed. It is assumed that the NRE is already been paid or will be paid by other projects.
- 2 sets of flight hardware for the X-band (STM transponder and X-TWTA) and 1 set of flight hardware for the UHF (Mars network transceiver) and 1 set of spares (except for the HGA which has no spares) are included.

WBS 4.1.9: COMMAND & DATA HANDLING AND 4.1.10: C&DH SOFTWARE

QQ XXX design cost summary, in $M:

Hardware spares	x.x
Non-recurring Engineering—Hardware Design (DDT&E)	x.x
Recurring Engineering Hardware	x.x
Non-recurring Engineering—Software Design (DDT&E)	x.x
Recurring Engineering Software	x.x
Total	

CDS COST ESTIMATE



WBS 4.1.11: Mechanical Build-Up

The Mechanical Build-Up cost is estimated by the Structures engineer, and is included in the tables in that section (above). This WBS element includes facilities, ground equipment, and personnel for the actual assembly of the spacecraft.

WBS 4.2: Instruments

The costs shown here are from a model that uses mass as a main driver from a data base of over 200 instruments.

283.06	283.06
283.06	283.06

The costs shown here are using our cost model and making the assumption that the hardware is build at Berkeley or a university

197.15	197.15
197.15	197.15

The costs shown here are without the 30% mass contingency

243.93	243.93
243.93	243.93

The costs shown below are for procuring the instrument at a university and also without the 30% mass cont.

169.90	169.90
169.90	169.90

The customer estimated costs are:

190

After discussing the details of the customer estimate, we are convinced their estimate is better than our model. But the model does bracket their cost.

WBS 4.3: Integration, Assembly, Test, and Launch Operations

The ATLO cost estimate is based on the Team X Parametric Cost Model. It is based primarily on the number of large pieces of the spacecraft that need to be integrated and the total cost of the spacecraft and instruments.

<Likely nothing of value coming from Systems, but you should double-check.>

WBS 5.0: Mission Operations (including MOS Development)

The mission operations development (Phases B/C/D) cost for SNAP is as follows.

- The dedicated ground station at Berkeley is costed as a new build even though the customer says a retrofit of an existing station is possible. We have no basis for a costing of the retrofit. The new build assumes the land and support buildings to house the data processing systems and operators are available for no cost. The physical ground station development cost is $x.xM (FY03$). This figure was taken from material supplied by the customer. The Ka-band receiver rough cost is $xM (FY03$). The data processing and local data storage development cost is $xM (FY03$).
- The remainder of the mission operation development is costed as if it were developed at GSFC. The cost is $xx.xM (FY03$).
- The total cost of mission operations development is $xx.xM (FY03$).

The mission operations (Phase E) cost for SNAP is as follows:

- The cost for operation of the ground station for 4 years is $x.xM (FY03$).
- The remainder of the mission operations is costed as if it were operated at GSFC. The cost for 4 years is $x.xM.
- The total cost for Phase E mission operations is $xx.xM (FY03$). This cost does not include science data processing or science data analysis.

WBS 6.0: Reserves

These are cost reserves that would normally be held by the project manager or principal investigator, to be released when unexpected costs appear. These reserves are assumed to be 5% of Phase A costs, 10% of Phase B, 20% of Phase C/D, and 10% of Phase E.

Launch Vehicle

The launch vehicle costs for this study have been taken from the most recent announcement of opportunity by the Office of Space Science for missions generally of this cost, and should be replaced after discussion with the NASA launch services procurement people. The Delta II 7925 costs are from the most recent Discovery AO (98-OSS-04).

The launch vehicle costs for this study are based on costs for commercial launches published in the International Space Industry Report, 6 July 1998. The commercial costs published there are adjusted upward slightly to account for the higher prices NASA projects typically have to pay, due to overheads in procuring the launch vehicle, and also due to inefficiencies in the procurement process. This additional cost typically runs several mission dollars, often 10%-20% of the launch vehicle cost.

The cost for the Star 48 upper stage has been estimated based on the Discovery AO and ISIR cost differences between Delta II 7x25 and 7x20 launch vehicles (the 7x25's are 7x20's with Star 48 upper stages).

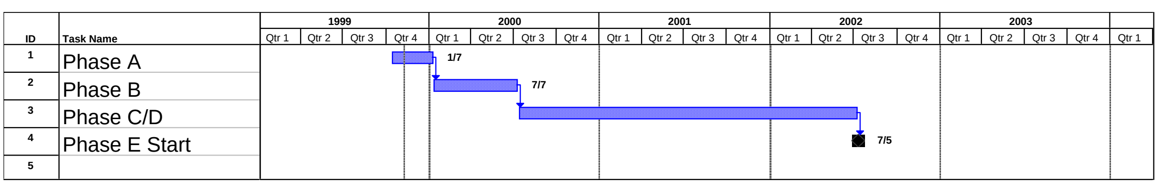
Technology Development Cost Estimates

Several subsystems are planning on using technologies that still need significant development before being ready to be used on a mission such as this. The affected subsystems have developed estimates as to how much funding will be required to develop the technology to the extent needed by this project. None of these costs have been included in the total project cost estimates presented above.

Project Schedule

[Delete if programmatics will be providing a write-up]

The DNP standard schedule is appropriate for this project. It uses a 3-month phase A, 6-month phase B, and 2-year phase C/D. A top level Gantt chart for this schedule is shown below.



5.0 ISSUES AND CONCERNS

SCIENCE
None

MISSION DESIGN
None

GROUND SYSTEMS AND MISSION OPERATIONS
Issues are discussed under the "Trade Considerations" sections. There are no concerns.

INSTRUMENTS
- We recommend the customer procures a complete set of spares for all the optical elements
- We recommend the customer design a backup FPA using existing commercial detectors.
- We agree that the detectors and optical fabrication will have

SYSTEMS
Issues are discussed under the "Trade Considerations" sections. There are no concerns.

PROPULSION
Issues are discussed under the "Trade Considerations" sections. There are no concerns.

ACS

REACTION WHEEL SIZING NEEDS REVISITING
- Need to characterize reaction wheel disturbance spectrum to determine acceptable operating speed range.
- If wheel speeds must be severely limited to avoid exciting jitter, may need more or larger wheels to have an adequate momentum storage capability.

POINTING CONTROL REQUIREMENT DURING IMAGING
- Customer information provided to Team X before the study indicates a requirement for pitch and yaw pointing control to within ±1 arcsec (3 sigma) over 1000 seconds.
- The real requirement appears to be pitch and yaw pointing control to within ±0.09 arcsec (3 sigma) over 1200 seconds based on taking four successive 300-second images in a dithering sequence.
- Need follow-up to confirm what the real requirement is.

ACS COST ESTIMATE
- The Team X ACS cost estimate is considerably higher than the customer expected, and there were questions raised as to what justifies the cost.

- Team X ACS costs are based on parametric models for levels of effort and grass-roots estimates for hardware. The parametric models were calibrated using flight actuals for a number of past JPL missions.
- There is an open question as to how much cheaper an industry supplier would be compared to JPL.
- The opinion of cost experts at JPL is that industry would not be substantially cheaper for a mission of this type since the pointing requirements are extremely tight and necessitate thorough design and analysis and integration and test.

CDS
This study assumes a substantial cost reduction by procuring the bus elements from a supplier of a spacecraft of very similar ACS, CDS and Power such as Mars 05. Validation of > 60% inheritance would help validate the cost assumptions.

POWER
The spacecraft cannot operate continuously in the "Telecom" mode. The solar array size was limited to the area on the spacecraft baffel assembly as shown in figure 1. This limits the amount of time for telecom passes. The operation constraint is the inability to do continuous memory read outs (MROs) or other troubleshooting. The effect on safe hold must be considered and perhaps a deployable array can be incorporate to mitigate this concern.

THERMAL
The technology is well in hand and there are no special concerns.

STRUCTURES
Structures has no serious Issues and Concerns with this study.

TELECOM
MNT readiness: The Mars Network Transceiver (MNT) is a new development for the Mars Infrastructure program. The development is just beginning. Expected first use of the MNT is for 2003. If the MNT is not available for the Mars Recon satellite, the likely backups would be the Cincinnati Electronics Mars'01 UHF transceiver or an MCAS1 UHF transceiver.

PROGRAMMATICS
Programmatics has no serious Issues and Concerns with this study.

6.0 TECHNOLOGY

6.1 Power

Solar Array Technology

Triple junction solar array cell chemistry is *state-of-art* and flight proven. The temperature and solar flux environment in this mission is acceptable. The design is a typical commercial build.

Battery Technology

IPV

The IPV Ni-H_2 battery is the heaviest cell design. It has the highest TRL level of all Ni-H_2 designs and is suitable for use in single string configuration. Cell bypass circuitry is required.

CPV

The CPV Ni-H_2 battery is lighter than the IPV. It will sustain a larger voltage drop with 2 cells out per failure. Fewer cells to failure means that it is less suitable for single string designs. Cell bypass circuitry is required.

Li-Ion

Batteries based on Li-Ion chemistry are currently limited to 3-year missions. Single string operation is not an option dual strings are required. Single string Li-Ion configuration is required to out compete Ni-H_2 designs. Cell bypass design is not used for Li-Ion direct energy transfer configurations (DET).

Future Li-Ion

Solid-state Li-Ion architecture with hi-voltage buss and cell bypass/ voltage boost converter configuration allows Li-Ion batteries to operate in single string mode. This design is beyond current *state-of-art* but may become available in the future.

Electronics Technology

High TRL level technology. Commercially available

6.2 TELECOM

TABLE: TELECOM SUBSYSTEM TECHNOLOGY—ITEMS

	STM transponder	Mars Network Transceiver
Metrics	1.1 kg	1.6 kg
Benefits	Lower mass	Software configurable, provides relay data return and navigation function
Heritage	New design	New design
TRL	4	1
Criticality	Upgrade from SDST	Provides more functions and channels than the MCAS1 transceiver
Cost		
Schedule	EM in FY00	EM in FY00
Comments	Being developed by Section 336. Backup: SDST	Includes a 10W power amplifier. Being developed by Section 335 for the Mars Infrastructure program; Backup: MCAS1 UHF transceiver and CE Mars'01UHF transceiver

7.0 REFERENCES

POWER

REFERENCES

1. Team X Power Subsystem Tool
2. David Linden, Handbook of Batteries 2nd Edition

THERMAL

COLD RADIATOR TEMPERATURE DISTRIBUTION:

Finite Difference Solution Results For A Radiating Fin Of Uniform Area

Solution Obtained Using The Tri-Diagonal Solver

Coupled With An Iterative Procedure Based On Quasilinearization

----
-

The Data You Input to Form Are Here:

Base Temperature = **120.15- Kelvin**

Irradiation From Background = 0000.00- W/Sq.m

Thermal Conductivity = 220-W/m-Deg C

Emissivity of Surface of Fin = 0.5 . 0.5 on two sides is equivalent to 1 on one side of the radiator.

Absorptivity of Surface of Fin = 0

Length of Fin = 1.3-m

Fin Thickness = **0.004-m**

Equivalent Sink Temperature = 000.0-Kelvin

Ratio of Eq.Sink Temp to Base Temp. =0.000

The Program Converged After 11 Iterations

The Calculated Data Are Here:

Non-dimensional Radiating Fin Parameter Nc =000.189

Tip to Base Temperature Ratio = 0.931

FIN HEAT LOSS =12.27-W/M

Fin Efficiecy = 0.799. An area equal to **1.3m by 1.3m** = 1.69 m2 will dissipate **16.5 W**

Temperature Profile Data is Given Below

x/L	T/Tb	x - m	T - Deg C
0.000	1.000	0.000	-153.0
0.071	0.990	0.092	-154.2
0.143	0.980	0.186	-155.4
0.214	0.972	0.278	-156.4
0.286	0.964	0.372	-157.3
0.357	0.957	0.464	-158.2

0.429	0.951	0.558	-158.9
0.500	0.946	0.650	-159.5
0.571	0.942	0.742	-160.0
0.643	0.938	0.836	-160.4
0.714	0.935	0.928	-160.8
0.786	0.933	1.022	-161.1
0.857	0.932	1.114	-161.2
0.929	0.931	1.208	-161.3
1.000	0.931	1.300	-161.3

-

That Completes The Output of The FD Analysis.

TELECOM

FIGURE A-1 KA-BAND HIGH RATE DOWNLINK

4W SSPA

Ka-band/0.5 m HGA/57% eff./0.20 deg. pointing error

New Ka-band Berkeley Ground Station
Hot body noise = 0 K
1 way

Tlm channel/RS-Convolutional (7,1/2)/PB=1 E-6

Carrier Loop Bandwidth = 150.0 Hz.
Bit Rate = 240000000 bps

5.999E+04	Range, km
0.0004	Range, AU
0.00	OWLT, hrs
0.00	SEP, deg
10	Elev. Angle, deg

#	Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	S		
	TRANSMITTER PARAMETERS									
1	S/C RF Power Output	dBm	36.02	0.00	0.00	36.02	0.0000	T	4.0	Xmtr Pwr, W
2	Total Circuit Loss	dB	-4.00	FALSE	#####	-4.00	0.0000	U		Ka Carr Mode
3	Antenna Gain	dBi	42.05	0.49	-0.49	42.05	0.0401	T	1.3	3 dB beamwidth, deg
4	Ant Pointing Loss	dB	-0.26	0.71	-0.71	-0.26	0.1684	U	HGA	S/C Antenna
5	EIRP (1+2+3+4)	dBm				73.80	0.2085			
	PATH PARAMETERS									
6	Space Loss	dB	-218.11	0.00	0.00	-218.11	0.0000	D	Ka	RF band
7	Atmospheric Attn	dB	-1.99	0.00	0.00	-1.99	0.0000	D	32000.00	Freq, MHz
	RECEIVER PARAMETERS									
8	BGS Antenna Gain	dBi	70.80	0.30	-0.30	70.80	0.0150	T	90	Weather %
9	Ant Pointing Loss	dB	-0.20	0.00	0.00	-0.20	0.0000	U	~~54~~	BGS
#	Polarization Loss	dB	-0.07	0.01	-0.01	-0.07	0.0001	U		
									N/A	LNA Selection
	TOTAL POWER SUMMARY								diplexed	
#	Total Rcvd Pwr (Pt) (5+6+7+8+9+10)	dBm				-75.77	0.2236	G	Optimize	
#	Noise Spec Dens	dBm/Hz	-170.07	0.00	0.00	-170.07	0.0000	G		
	System Noise Temp	K	713.00	0.00	0.00			G		
	Vacuum	K	32.00	0.00	0.00			T	1	WAY
	Elevation	K	2.55	0.00	0.00			G		
	Clouds	K	101.88	0.00	0.00			G		
	Hot Body Noise	K	0.00	0.00	0.00			G	G/T	40.08
#	**Received Pt/No** (11-12)	**dB-Hz**				**94.30**	**0.2236**	**G**		
	CARRIER PERFORMANCE at Req. Pt/No									
#	Tlm Carrier Supp	dB	-10.20	0.64	-0.70	-10.22	0.0757	T	TRUE	TLM MOD
#	Rng Carrier Supp	dB	0.00	0.00	0.00	0.00	0.0000	T	FALSE	RNG MOD
#	DOR Carrier Supp	dB	0.00	0.00	0.00	0.00	0.0000	T	FALSE	DOR MOD
#	Pc/No (39+14+15+16)	dB-Hz				80.02	0.2993	T		
#	Carrier Loop Bandwidth, Bl	dB-Hz	21.76	0.00	0.00	21.76	0.0000	T	150.0	Carrier Bl, Hz
#	Carrier Loop SNR	dB				58.26	0.2993	U	-10	Carrier Phase Noise
#	Required Carrier Loop SNR	dB				10.00	0.0000	D	Type 2, SuperCritically Damp	
#	Carrier Loop SNR Margin	dB				48.26	0.2993	U	0.000	p.n. var., tx contribution, rad^2
									0.000	p.n. var., loop snr, rad^2
	SUBCARRIER PERFORMANCE at Req. Pt/No								0.000	p.n. solar, rad^2
#	SubCar. L. SNR	dB				67.92			0.000	phase noise variance, rad^2
#	Required Loop SNR	dB				20.00			5000	SubCarr Bl, mHz
#	SubCarrier Loop SNR Margin	dB				47.92			1.0	SubCarr window f.
	SYMBOL LOOP PERFORMANCE at Req. Pt/No									
#	Sym. Loop SNR	dB				53.25			5000	Sym Bl, mHz
#	Required Loop SNR	dB				15.00			1.0	Sym window f.
#	SubCarrier Loop SNR Margin	dB				38.25				
									Optimize	
	TELEMETRY PERFORMANCE at Required Pt/No									
#	Tlm Data Supp	dB	-0.44	0.07	-0.07	-0.44	0.0008	T	72.0	tlm MI, deg
#	Rng Data Supp	dB	0.00	0.00	0.00	0.00	0.0000	T	0.00	peak rng MI, deg
#	DOR Data Supp	dB	0.00	0.00	0.00	0.00	0.0000	T	480000000	Symbol Rate, bps
#	Data Rate	dB	83.80	0.00	0.00	83.80	0.0000	D	########	bit rate, bps
#	Radio Loss	dB	0.00			0.00		T	1.00	radio loss, dimensionless
#	SubCarrier Demod. Loss	dB	0.00			0.00		T	1.00	subcarr. loss, dimensionless
#	Symbol Sync. Loss	dB	-0.26			-0.26		T	1.06	sym. sync. loss, dimensionless
#	Waveform Distortion Loss	dB	-0.61			-0.61		T		
#	Array Gain (incl. -0.1 db loss)	dB	0.00			0.00		T	FALSE	Array
#	Threshold Eb/No	dB				2.31		D	Convolutional (7,1/2)	Coding
#	Required Eb/No	dB				6.00			0.005	BER at BVR output
#	**Required Pt/No**	**dB-Hz**				**90.24**	**0.2500**	**U**		
#	**Performance Margin** (39-13)	**dB**				4.06				

FIGURE A-2 S-BAND NOMINAL ENGINEERING DOWNLINK

3W SSPA

S-band LGA - 5 deg. off-point

Berkeley S-band Ground Station
Hot body noise = 0 K
1 way

Tlm channel/RS-Convolutional (7,1/2)/PB=1.E-6

Carrier Loop Bandwidth = 13.7 Hz
Bit Rate = 45052 bps

5.999E+04	Range, km
0.0004	Range, AU
0.00	OWLT, hr
0.00	SEP, deg
10	Elev. Angle, deg

#	Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	S		
	TRANSMITTER PARAMETERS									
1	S/C RF Power Output	dBm	34.77	1.30	-0.60	35.00	0.1574	T	3.0	Xmtr Pwr, W
2	Total Circuit Loss	dB	-2.00	FALSE	#####	-2.00	0.0000	U		
3	Antenna Gain	dBi	5.00	0.51	-0.51	5.00	0.0436	T	5.0	Off-Boresight Angle, deg
4	Ant Pointing Loss	dB	0.00	0.00	0.00	0.00	0.0000	U	LGA	S/C Antenna
5	EIRP (1+2+3+4)	dBm				38.00	0.2009			
	PATH PARAMETERS									
6	Space Loss	dB	-195.24	0.00	0.00	-195.24	0.0000	D	S	RF band
7	Atmospheric Attn	dB	-0.19	0.00	0.00	-0.19	0.0000	D	2299.07	Freq, MHz
	RECEIVER PARAMETERS									
8	Antenna Gain	dBi	46.40	0.10	-0.20	46.37	0.0039	T	90	Weather %
9	Ant Pointing Loss	dB	-0.03	0.00	0.00	-0.03	0.0000	U	~~54~~	BGS
#	Polarization Loss	dB	-0.05	0.00	0.00	-0.05	0.0000	U		
									N/A	LNA Selection
	TOTAL POWER SUMMARY								diplexed	
#	Total Rcvd Pwr (Pt) (5+6+7+8+9+10)	dBm				-111.14	0.2048	G	Optimize	
#	Noise Spec Dens	dBm/Hz	-174.65	-0.02	0.03	-174.65	0.0001	G		
	System Noise Temp	K	248.00	-1.00	2.00			G		
	Vacuum	K	37.48	-1.00	2.00			T	1	WAY
	Elevation	K	0.00	0.00	0.00			G		
	Clouds	K	11.88	0.00	0.00			G		
	Hot Body Noise	K	0.00	0.00	0.00			G	G/T	22.24
#	**Received Pt/No** (11-12)	**dB-Hz**				**63.50**	**0.2049**	**G**		
	CARRIER PERFORMANCE at Req. Pt/No									
#	Tlm Carrier Supp	dB	-10.20	0.64	-0.70	-10.22	0.0757	T	TRUE	TLM MOD
#	Rng Carrier Supp	dB	-2.13	0.43	-0.48	-2.15	0.0347	T	HIGH	RNG MOD
#	DOR Carrier Supp	dB	0.00	0.00	0.00	0.00	0.0000	T	FALSE	DOR MOD
#	Pc/No (39+14+15+16)	dB-Hz				41.06	0.4177	T		
#	Carrier Loop Bandwidth, Bl	dB-Hz	11.36	0.00	0.00	11.36	0.0000	T	13.7	Carrier Bl, Hz
#	Carrier Loop SNR	dB				29.70	0.4177	U	-20	Carrier Phase Noise
#	Required Carrier Loop SNR	dB				10.00	0.0000	D	Type 2 - SuperCritically Damp	
#	Carrier Loop SNR Margin	dB				19.70	0.4177	U	0.001	p.n. var.,tx contribution, rad^2
									0.001	p.n. var., loop snr, rad^2
	SUBCARRIER PERFORMANCE at Req. Pt/No								0.000	p.n.,solar, rad^2
#	SubCar. L. SNR	dB				48.18			0.002	phase noise variance, rad^2
#	Required Loop SNR	dB				20.00			1000	SubCarr Bl, mHz
#	SubCarrier Loop SNR Margin	dB				28.18			1	SubCarr window f.
	SYMBOL LOOP PERFORMANCE at Req. Pt/No									
#	Sym. Loop SNR	dB				39.38			1000	Sym Bl, mHz
#	Required Loop SNR	dB				15.00			1	Sym window f.
#	SubCarrier Loop SNR Margin	dB				24.38				
									Optimize	
	TELEMETRY PERFORMANCE at Required Pt/No									
#	Tlm Data Supp	dB	-0.44	0.07	-0.07	-0.44	0.0008	T	72.0	tlm MI, deg
#	Rng Data Supp	dB	-2.13	0.43	-0.48	-2.15	0.0347	T	55.00	peak rng MI, deg
#	DOR Data Supp	dB	0.00	0.00	0.00	0.00	0.0000	T	90104	Symbol Rate, bps
#	Data Rate	dB	46.54	0.00	0.00	46.54	0.0000	D	45052	bit rate, bps
#	Radio Loss	dB	-0.01			-0.01		T	1.00	radio loss, dimensionless
#	SubCarrier Demod. Loss	dB	-0.02			-0.02		T	1.00	subcarr. loss, dimensionless
#	Symbol Sync. Loss	dB	-0.02			-0.02		T	1.00	sym. sync. loss, dimensionless
#	Waveform Distortion Loss	dB	-0.13			-0.13		T		
#	Array Gain (incl. -0.1 db loss)	dB	0.00			0.00		T	FALSE	Array
#	Threshold Eb/No	dB				2.31		D	Convolutional (7,1/2)	coding
#	Required Eb/No	dB				4.30			0.005	BER at BVR output
#	**Required Pt/No**	**dB-Hz**				**53.42**	**0.2855**	**U**		
#	**Performance Margin** (39-13)	**dB**				**10.08**				

FIGURE A-3 S-BAND NOMINAL COMMAND UPLINK

LGA Uplink
BGS 0.04 kW/10m S-band station

Hot Body Noise: ~ 100K

2000.0000

5.999E+04	Range, km
0.0004	Range, AU
0.00	OWLT, hrs
0.00	SEP, deg
10	Elev. Angle

	Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	Shape		
	TRANSMITTER PARAMETERS									
1	Total Xmitter Pwr	dBm	46.02	0.00	-1.00	45.69	0.0556	T	0.04	Xmtr Pwr, kW
2	Xmitter Waveguide Loss	dB	-1.50	0.10	-0.10	-1.50	0.0017	T		
3	Antenna Gain	dBi	45.00	0.20	-0.30	44.97	0.0106	T	~~54~~	
4	Ant Pointing Loss	dB	-0.10	0.00	0.00	-0.10	0.0000	U		
5	EIRP (1+2+3+4)	dBm				89.05	0.0679	U		
									S	RF band
	PATH PARAMETERS									
6	Space Loss	dB	-194.52			-194.52		D	2117.06	Freq, MHz
7	Atmospheric Attn	dB	-0.19			-0.19		D	90	Weather %
	RECEIVER PARAMETERS									
8	Polarization Loss	dB	-0.05	0.00	0.00	-0.05	0.0000	U		
9	Ant Pointing Loss	dB	0.00	0.76	-0.76	0.00	0.1930	U		
10	S/C Antenna Gain	dBi	4.00	0.51	-0.51	4.00	0.0432	T	5.0	Boresight Angle, deg
11	Lumped Ckt/Ant Loss	dB	-2.20	FALSE	FALSE	-2.20	0.0000	U	LGA	S/C Antenna
	TOTAL POWER SUMMARY									
12	Total Rcvd Pwr (Pt) (5+6+7+8+9+10+11)	dBm				-103.91	0.3041	G		
13	Noise Spec Dens	dBm/Hz	-169.56	5.46	5.46	-164.10	0.0000	G		
	System Noise Temp	K	802.00	#VALUE!	######					
	Rcvr Noise Temp	K	627.00	#VALUE!	######					
	Rcvr Noise Figure	dB	5.00	#N/A	#N/A					
	Loss Noise Contr.	K	115.26	0.00	0.00					
	Ant Noise Contr.	K	60.00	0.00	0.00					
14	Rcvd Pt/No	dB-Hz				60.19	0.3041	G	(12-13)	
	CARRIER PERFORMANCE									
15	Cmd Carrier Supp	dB	-3.46	0.20	-0.20	-3.46	0.0067	T	TRUE	CMD.MOD
16	Rng Carrier Supp	dB	-3.00	0.10	-0.10	-3.00	0.0017	T	TRUE	RNG.MOD
17	Rcvd Carr Power (Pc)	dBm				-110.37	0.3191	T	(12+15+16)	
17a	Pc/No	dB-Hz				53.73				
18	Carr Noise BW, BL	dB-Hz	20.37	0.08	-0.08	20.37	0.0021	U	108.9	Bl, Hz
19	Required Carrier Loop SNR	dB	12.00			12.00		D		
20	Excess Carrier Loop SNR	dB				21.36	0.3212	U	(17-13-18-19)	
	CHANNEL PERFORMANCE									
21	Cmd Modulation Loss	dB	-3.04	0.17	-0.18	-3.04	0.0051	T	1.2	cmd MI, rad
22	Rng Data Supp	dB	-3.00	0.10	-0.10	-3.00	0.0017	T	44.9	rng MI, deg
23	Data Pwr to Rcvr (Pd)	dBm				-109.95	0.3167	T	(12+21+22)	
24	Data Rate	dB	33.01	0.00	0.00	33.01	0.0000	D	2000	data rate
25	Eb/No	dB				21.14	0.3167	T	(14+21+22-24)	
26	System Loss	dB	-1.00	0.00	-1.00	-1.33	0.0556	T	(includes radio loss)	
27	Threshold Eb/No	dB	9.60			9.60		D	BER = 1e-5, uncoded	
28	Performance Margin	dB				10.21	0.3167	T	(25+26-27)	

3W SSPA

S-band LGA - 5 deg. off-point
BGS 10m S-band station

Hot body noise = 0 K
1 way

Tlm channel/BS-Convolutional (7,1/2)/PB=1.E-6

Carrier Loop Bandwidth = 0.3 Hz
Bit Rate = 10000 bps

1.646E+05	Range, km
0.0011	Range, AU
0.00	OWLT, hrs
0.00	SEP, deg
10	Elev. Angle, deg

#	Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	S		
	TRANSMITTER PARAMETERS									
1	S/C RF Power Output	dBm	34.77	1.30	-0.60	35.00	0.1574	T	3.0	Xmtr Pwr, W
2	Total Circuit Loss	dB	-2.00	FALSE	#####	-2.00	0.0000	U		
3	Antenna Gain	dBi	5.00	0.51	-0.51	5.00	0.0435	T	5.0	Off-Boresight Angle, deg
4	Ant Pointing Loss	dB	0.00	0.00	0.00	0.00	0.0000	U	LGA	S/C Antenna
5	EIRP (1+2+3+4)	dBm				38.00	0.2009			
	PATH PARAMETERS									
6	Space Loss	dB	-204.01	0.00	0.00	-204.01	0.0000	D	S	RF band
7	Atmospheric Attn	dB	-0.19	0.00	0.00	-0.19	0.0000	D	2298.07	Freq, MHz
	RECEIVER PARAMETERS									
8	BGS Antenna Gain	dBi	46.40	0.10	-0.20	46.37	0.0039	T	90	Weather %
9	Ant Pointing Loss	dB	-0.03	0.00	0.00	-0.03	0.0000	U	~~34~~	BGS Antenna
#	Polarization Loss	dB	-0.05	0.00	0.00	-0.05	0.0000	U		
									N/A	LNA Selection
	TOTAL POWER SUMMARY								diplexed	
#	Total Rcvd Pwr (Pt) (5+6+7+8+9+10)	dBm				-119.91	0.2040	G	Optimize	
#	Noise Spec Dens	dBm/Hz	-174.65	-0.02	0.03	-174.65	0.0001	G		
	System Noise Temp	K	248.00	-1.00	2.00			G		
	Vacuum	K	37.48	-1.00	2.00			T	1	WAY
	Elevation	K	0.00	0.00	0.00			G		
	Clouds	K	11.88	0.00	0.00			G		
	Hot Body Noise	K	0.00	0.00	0.00			G	G/T	22.24
#	**Received Pt/No** (11-12)	**dB-Hz**				**54.74**	**0.2049**	**G**		
	CARRIER PERFORMANCE at Req. Pt/No									
#	Tlm Carrier Supp	dB	-10.20	0.64	-0.70	-10.22	0.0757	T	TRUE	TLM.MOD
#	Rng Carrier Supp	dB	-2.13	0.43	-0.48	-2.15	0.0347	T	HIGH	RNG.MOD
#	DOR Carrier Supp	dB	0.00	0.00	0.00	0.00	0.0000	T	FALSE	DOR.MOD
#	Pc/No (39+14+15+16)	dB-Hz				34.52	0.4177	T		
#	Carrier Loop Bandwidth, Bl	dB-Hz	9.10	0.00	0.00	9.10	0.0000	T	0.3	Carrier Bl, Hz
#	Carrier Loop SNR	dB				25.34	0.4177	U	-20	Carrier Phase Noise
#	Required Carrier Loop SNR	dB				10.00	0.0000	D	Type 2. SuperCritically Damp	
#	Carrier Loop SNR Margin	dB				15.34	0.4177	U	0.002	p.n. var, tx contribution, rad^2
									0.003	p.n. var, loop snr, rad^2
	SUBCARRIER PERFORMANCE at Req. Pt/No								0.000	p.n., solar, rad^2
#	SubCar. L. SNR	dB				41.64			0.005	phase noise variance, rad^2
#	Required Loop SNR	dB				20.00			1000	SubCarr Bl, mHz
#	SubCarrier Loop SNR Margin	dB				21.64			1	SubCarr window f.
	SYMBOL LOOP PERFORMANCE at Req. Pt/No									
#	Sym. Loop SNR	dB				32.84			1000	Sym Bl, mHz
#	Required Loop SNR	dB				15.00			1	Sym window f.
#	SubCarrier Loop SNR Margin	dB				17.84				
									Optimize	
	TELEMETRY PERFORMANCE at Required Pt/No									
#	Tlm Data Supp	dB	-0.44	0.07	-0.07	-0.44	0.0008	T	72.0	tlm MI, deg
#	Rng Data Supp	dB	-2.13	0.43	-0.48	-2.15	0.0347	T	55.00	peak rng MI, deg
#	DOR Data Supp	dB	0.00	0.00	0.00	0.00	0.0000	T	20000	Symbol Rate, bps
#	Data Rate	dB	40.00	0.00	0.00	40.00	0.0000	D	10000	bit rate, bps
#	Radio Loss	dB	-0.04			-0.04		T	1.01	radio loss, dimensionless
#	SubCarrier Demod. Loss	dB	-0.05			-0.05		T	1.01	subcarr. loss, dimensionless
#	Symbol Sync. Loss	dB	-0.05			-0.05		T	1.01	sym. sync. loss, dimensionless
#	Waveform Distortion Loss	dB	0.00			0.00		T		
#	Array Gain (incl. -0.1 db loss)	dB	0.00			0.00		T	FALSE	Array
#	Threshold Eb/No	dB				2.31		D	Convolutional (7,1/2)	Coding
#	Required Eb/No	dB				4.30			0.005	BER at BVR output
#	**Required Pt/No**	**dB-Hz**				**46.89**	**0.2856**	**U**		
#	**Performance Margin** (39-13)	**dB**				**7.85**				

FIGURE A-4 S-BAND SAFEMODE DOWNLINK

Figure A-5 S-band Safemode Command Uplink

LGA Uplink
BGS 0.04 kW/10 m ground station

Hot Body Noise: ~ 100K

1000.0000

1.646E+05	Range, km
0.0011	Range, AU
0.00	OWLT, hrs
0.00	SEP, deg
10	Elev. Angle

	Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	Shape		
	TRANSMITTER PARAMETERS									
1	Total Xmitter Pwr	dBm	46.02	0.00	-1.00	45.69	0.0556	T	0.04	Xmtr Pwr, kW
2	Xmitter Waveguide Loss	dB	-1.50	0.10	-0.10	-1.50	0.0017	T		
3	BGS Antenna Gain	dBi	45.00	0.20	-0.30	44.97	0.0106	T	~~54~~	
4	Ant Pointing Loss	dB	-0.10	0.00	0.00	-0.10	0.0000	U		
5	EIRP (1+2+3+4)	dBm				89.05	0.0679	U		
									S	RF band
	PATH PARAMETERS									
6	Space Loss	dB	-203.29			-203.29		D	2117.06	Freq, MHz
7	Atmospheric Attn	dB	-0.19			-0.19		D	90	Weather %
	RECEIVER PARAMETERS									
8	Polarization Loss	dB	-0.05	0.00	0.00	-0.05	0.0000	U		
9	Ant Pointing Loss	dB	0.00	0.76	-0.76	0.00	0.1930	U		
10	S/C Antenna Gain	dBi	4.00	0.51	-0.51	4.00	0.0432	T	5.0	Boresight Angle, deg
11	Lumped Ckt/Ant Loss	dB	-2.20	FALSE	FALSE	-2.20	0.0000	U	LGA	S/C Antenna
	TOTAL POWER SUMMARY									
12	Total Rcvd Pwr (Pt) (5+6+7+8+9+10+11)	dBm				-112.68	0.3041	G		
13	Noise Spec Dens	dBm/Hz	-169.56	5.46	5.46	-164.10	0.0000	G		
	System Noise Temp	K	802.00	#VALUE!	######					
	Rcvr Noise Temp	K	627.00	#VALUE!	######					
	Rcvr Noise Figure	dB	2.70	#N/A	#N/A					
	Loss Noise Contr.	K	115.26	0.00	0.00					
	Ant Noise Contr.	K	60.00	0.00	0.00					
14	Rcvd Pt/No	dB-Hz				51.42	0.3041	G	(12-13)	
	CARRIER PERFORMANCE									
15	Cmd Carrier Supp	dB	-3.46	0.20	-0.20	-3.46	0.0067	T	TRUE	CMD.MOD
16	Rng Carrier Supp	dB	-3.00	0.10	-0.10	-3.00	0.0017	T	TRUE	RNG.MOD
17	Rcvd Carr Power (Pc)	dBm				-119.14	0.3191	T	(12+15+16)	
17a	Pc/No	dB-Hz				44.96				
18	Carr Noise BW, BL	dB-Hz	19.85	0.09	-0.09	19.85	0.0027	U	96.7	Bl, Hz
19	Required Carrier Loop SNR	dB	12.00			12.00		D		
20	Excess Carrier Loop SNR	dB				13.11	0.3218	U	(17-13-18-19)	
	CHANNEL PERFORMANCE									
21	Cmd Modulation Loss	dB	-3.04	0.17	-0.18	-3.04	0.0051	T	1.2	cmd MI, rad
22	Rng Data Supp	dB	-3.00	0.10	-0.10	-3.00	0.0017	T	44.9	rng MI, deg
23	Data Pwr to Rcvr (Pd)	dBm				-118.72	0.3167	T	(12+21+22)	
24	Data Rate	dB	30.00	0.00	0.00	30.00	0.0000	D	1000	data rate
25	Eb/No	dB				15.39	0.3167	T	(14+21+22-24)	
26	System Loss	dB	-1.00	0.00	-1.00	-1.33	0.0556	T	(includes radio loss)	
27	Threshold Eb/No	dB	9.60			9.60		D	BER = 1e-5, uncoded	
28	Performance Margin	dB				4.45	0.3167	T	(25+26-27)	

APPENDIX—TECHNOLOGY READINESS LEVEL DEFINITIONS

TABLE D-1: TRL DEFINITIONS

Basic Research
Feasibility Research
Technology Development
Technology Validation
System/Subsystem Development
System Test, Launch and Ops

Level 1	Basic principles observed and reported (G)
Level 2	Technology concepts/applications formulated (G)
Level 3	Analytical and experimental critical function and/or characteristic proof-of-concept (G)
Level 4	Component and/or breadboard validation in laboratory (G)
Level 5	Component and/or breadboard demonstration in relevant environment (G or S)
Level 6	System validation model demonstration in relevant/simulated environment (G or S)
Level 7	System validation model demonstrated in space (S)
Level 8	Actual system completed and "flight qualified" through test and demonstration (G or S)
Level 9	Actual system "flight proven" through sucessful mission operations (S)

APPENDIX—DNP/FBC ASSUMPTIONS

- Development cost will include phases A through D—the formulation phase. The formulation phase is assumed to be approximately 33 months long (9 months for the old phase A/B and 24 months for the old phase C/D). Phase E is to be known as the implementation phase. Implementation costs are not typically impacted by FBC.
- A fast hardware build and delivery time line will not exceed 18 months. Long lead items may result in fixed designs for some system components.
- Spacecraft subsystems will assume that behavioral and cross-cutting models will be developed. This will result in a reduction is work force and substantially reduced requirements documentation. The model process will replace the previous specification process.
- The engineering staff will be highly experienced.
- Previously designed flight hardware will be used as much as possible.
- Unless otherwise indicated, inheritance will be based on X2000 hardware.
- New technology development for hardware will be paid for prior to the start of the project being analyzed.
- The development level for the project will be at least at a Technology Readiness Level of 6 before the start of Phase A (Formulation). Families of missions will use a fixed technology base and be sized for unique performance requirements only.
- Off-the-shelf components will be used.
- Component verification and qualification will be done at the spacecraft contractor whenever possible.
- Mass purchase prices for hardware will be assumed (a purchase of at least 5 units).
- Minimal (level of effort) quality assurance will be assumed.
- Ground Support Equipment (GSE) will be provided by the Flight System Test Bed (FST) where continuous real and virtual hardware testing is done. All flight software will also be tested in the FST. FST costs will be funded by JPL and not the project.
- All design hub development and operations costs will be funded by JPL and not the project.
- When a second spacecraft is to be built, it will be part of the original plan and identical to the first spacecraft in all functional aspects, including flight and mission parameters (not just 95% the same), in order for the second unit cost estimate to be valid.
- Due to earlier delivery requirements of the ground system and increase of required I&T support, phase C/D ground system costs will incur a small increase.

APPENDIX—LEVELS OF RADIATION-HARDENING

TABLE F-1: WHAT RAD-HARD MEANS

Rad-hard	Rad-tolerant	Commercial
• Designed for specific hardness level • Total dose: >200 krad to >1 Mrad • SEU threshold LET: 80150 MeV/mg/cm^2 • SEU error rate: 10^{-10} to 10^{-12} errors/bit-day • Latchup: Si on insulator technologies; no latchup problems	• Hardness offered as a by-product of the design • Total dose: 20 krad to >50 krad • SEU threshold LET: 20 MeV/mg/cm^2 (typ) • SEU error rate: 10^{-7} to 10^{-8} errors/bit-day • Latchup: Customer evaluation and risk	• Hardness limited by inherent process and design; customer risk • Total dose: 2 krad to >10 krad • SEU threshold LET: 5 MeV/mg/cm^2 (typ) • SEU error rate: 10^{-5} errors/bit-day • Latchup: Customer evaluation and risk

APPENDIX—ACRONYMS

Check the Team X Acronym List at http://pdc.jpl.nasa.gov/users/teamx/Acronyms.html