

## **ADVANCED PROJECTS DESIGN TEAM**

**STUDY NAME: SUPER NOVA-ACC. PROBE (SNAP)**

**STUDY LEADER: ANDY GERBER**

**28, 29, 31 OCTOBER & 01 NOVEMBER 2002**

**Modified by H. Heetderks – 1 October, 2003**

- 1) Cost Information Removed**
- 2) Various modifications (highlighted in blue)**

1.0 EXECUTIVE SUMMARY	4
1.1 Mission Objectives	4
1.2 Design Description	4
1.3 Programmatic	9
2.0 MISSION REQUIREMENTS	10
2.1 Science Requirements	10
2.2 Instrument Requirements	12
3.0 FLIGHT SYSTEM	14
3.1 Systems	14
3.2 Propulsion	20
3.3 ACS	24
3.4 CDS	36
Mass and Power Estimate Summary	38
3.5 Power	42
3.6 Thermal	47
3.7 Structures	48
3.8 Telecom	52
3.9 SOFTWARE	56
4.0 PROGRAMMATIC	59
5.0 Issues and CONCERNS	65
Science	65
Mission Design	65
Ground Systems and Mission Operations	65
Instruments	65
Systems	65
Propulsion	65
ACS	65
CDS	65
Power	66
Thermal	66
Structures	66
Telecom	66
Programmatic	66
6.0 TECHNOLOGY	67

---

6.1 Power	67
6.2 Telecom	68
7.0 REFERENCES	69
Thermal	70
Telecom	72
APPENDIX—TECHNOLOGY READINESS LEVEL DEFINITIONS	77
APPENDIX—DNP/FBC ASSUMPTIONS	78
APPENDIX—LEVELS OF RADIATION-HARDENING	79
APPENDIX—ACRONYMS	80

## SNAP 2002-10 TEAM X ROSTER - MONDAY, OCTOBER 28, 2002

<b>Subsystem *</b> <b>=Lead</b>	<b>Name</b>	<b>Phone Number</b>	<b>E-Mail</b>
ACS*	Kinsey Bob	626-395-0460	<a href="mailto:Robert.J.Kinsey@aero.org">Robert.J.Kinsey@aero.org</a>
CDS*	Henry Mike	4-1553	<a href="mailto:Michael.D.Henry@jpl.nasa.gov">Michael.D.Henry@jpl.nasa.gov</a>
Configuration	Baez Enrique	3-6688	<a href="mailto:Enrique.Baez-Jr@jpl.nasa.gov">Enrique.Baez-Jr@jpl.nasa.gov</a>
Cost*	Roust Tamara	4-4614	<a href="mailto:Tamara.L.Roust@jpl.nasa.gov">Tamara.L.Roust@jpl.nasa.gov</a>
Cost	Fong Michael	3-2704	<a href="mailto:Michael.J.Fong@jpl.nasa.gov">Michael.J.Fong@jpl.nasa.gov</a>
Cost	Kwan Eric	3-6282	<a href="mailto:Eric.W.Kwan@jpl.nasa.gov">Eric.W.Kwan@jpl.nasa.gov</a>
Cost	Hihn Jairus	4-1248	<a href="mailto:Jairus.M.Hihn@jpl.nasa.gov">Jairus.M.Hihn@jpl.nasa.gov</a>
Cost	Nigg Daniel	310-336-2205	<a href="mailto:Daniel.A.Nigg@aero.org">Daniel.A.Nigg@aero.org</a>
Facilitator	Gerber Andrew	4-2172	<a href="mailto:Andrew.J.Gerber@jpl.nasa.gov">Andrew.J.Gerber@jpl.nasa.gov</a>
Ground Systems	Jones Michael	4-3918	<a href="mailto:Michael.K.Jones@jpl.nasa.gov">Michael.K.Jones@jpl.nasa.gov</a>
Instrument*	Danielson Ed	4-4121	<a href="mailto:George.E.Danielson-Jr@jpl.nasa.gov">George.E.Danielson-Jr@jpl.nasa.gov</a>
Mission Design	Strauss William	3-1482	<a href="mailto:William.D.Strauss@jpl.nasa.gov">William.D.Strauss@jpl.nasa.gov</a>
Power	Valdez Thomas	4-3797	<a href="mailto:Thomas.I.Valdez@jpl.nasa.gov">Thomas.I.Valdez@jpl.nasa.gov</a>
Product Czar*	Hamlin Louise	3-4491	<a href="mailto:Louise.A.Hamlin@jpl.nasa.gov">Louise.A.Hamlin@jpl.nasa.gov</a>
Programmatics	Cowan Debbie	4-4104	<a href="mailto:Deborah.A.Cowan@jpl.nasa.gov">Deborah.A.Cowan@jpl.nasa.gov</a>
Programmatics*	Heneghan Cate	4-1279	<a href="mailto:Cate.Heneghan@jpl.nasa.gov">Cate.Heneghan@jpl.nasa.gov</a>
Propulsion	Baker Raymond	3-2894	<a href="mailto:Raymond.S.Baker@jpl.nasa.gov">Raymond.S.Baker@jpl.nasa.gov</a>
Propulsion	Cowley Richard	3-0817	<a href="mailto:Richard.T.Cowley@jpl.nasa.gov">Richard.T.Cowley@jpl.nasa.gov</a>
Science*	Palluconi Frank	4-8362	<a href="mailto:Frank.D.Palluconi@jpl.nasa.gov">Frank.D.Palluconi@jpl.nasa.gov</a>
Software*	Tung Yu-Wen	4-2673	<a href="mailto:Yu-Wen.Tung@jpl.nasa.gov">Yu-Wen.Tung@jpl.nasa.gov</a>
Structure	Nicaise Fabien	3-6754	<a href="mailto:Fabien.Nicaise@jpl.nasa.gov">Fabien.Nicaise@jpl.nasa.gov</a>
Systems	Andringa Jason	4-1892	<a href="mailto:Jason.M.Andringa@jpl.nasa.gov">Jason.M.Andringa@jpl.nasa.gov</a>
Telecom*	Valas Sam	4-1413	<a href="mailto:Sam.Valas@jpl.nasa.gov">Sam.Valas@jpl.nasa.gov</a>
Telecom – H/W*	Dennis Matthew	3-4182	<a href="mailto:Matthew.Dennis@jpl.nasa.gov">Matthew.Dennis@jpl.nasa.gov</a>
Thermal	Shakkottai Partha	4-4808	<a href="mailto:P.Shakkottai@jpl.nasa.gov">P.Shakkottai@jpl.nasa.gov</a>

## 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 to 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

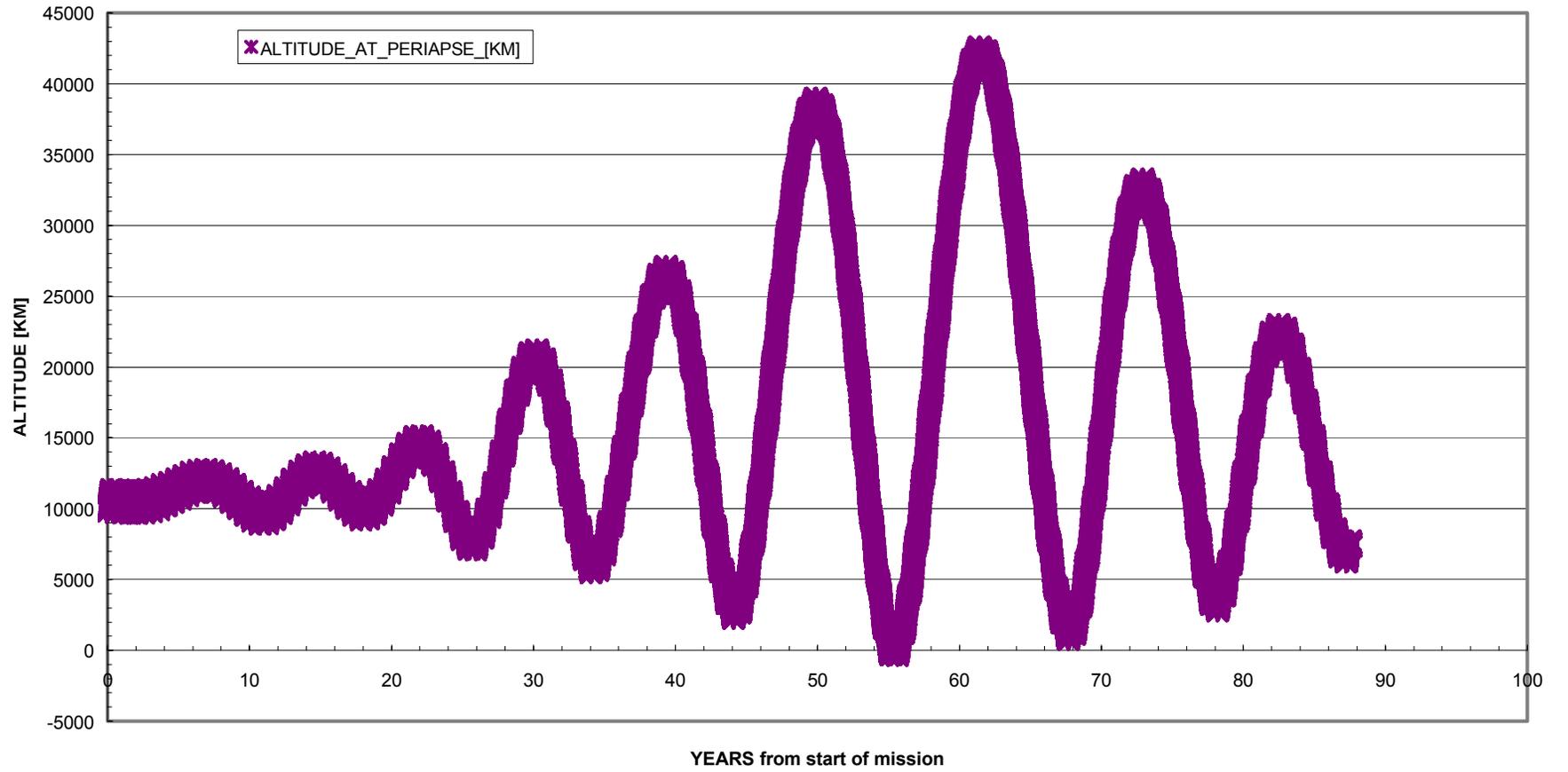
(semi-major axis [km], eccentricity , inclination [deg], longitude of the ascending node [deg], argument of periapse [deg], mean anomaly [deg]), Earth Centered Equator and Equinox of J2000: (87710.6, 0.8131, 26.3733, 349.938, 0.0, 0.0)

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. This inclination change may be small enough that correction is unnecessary. If correction is desired, a maneuver would be performed approximately every 30 days. Over a 5 year mission an orbit maintenance budget of approximately 180 m/s would be required. Further analysis would be required if this option is chosen.

#### 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 periapee 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
Reaction Wheel unloading	60
Orbit Maintenance	180 **
Total Delta-v required	290

\*\* if required to maintain inclination

### GROUND SYSTEMS AND MISSION OPERATIONS

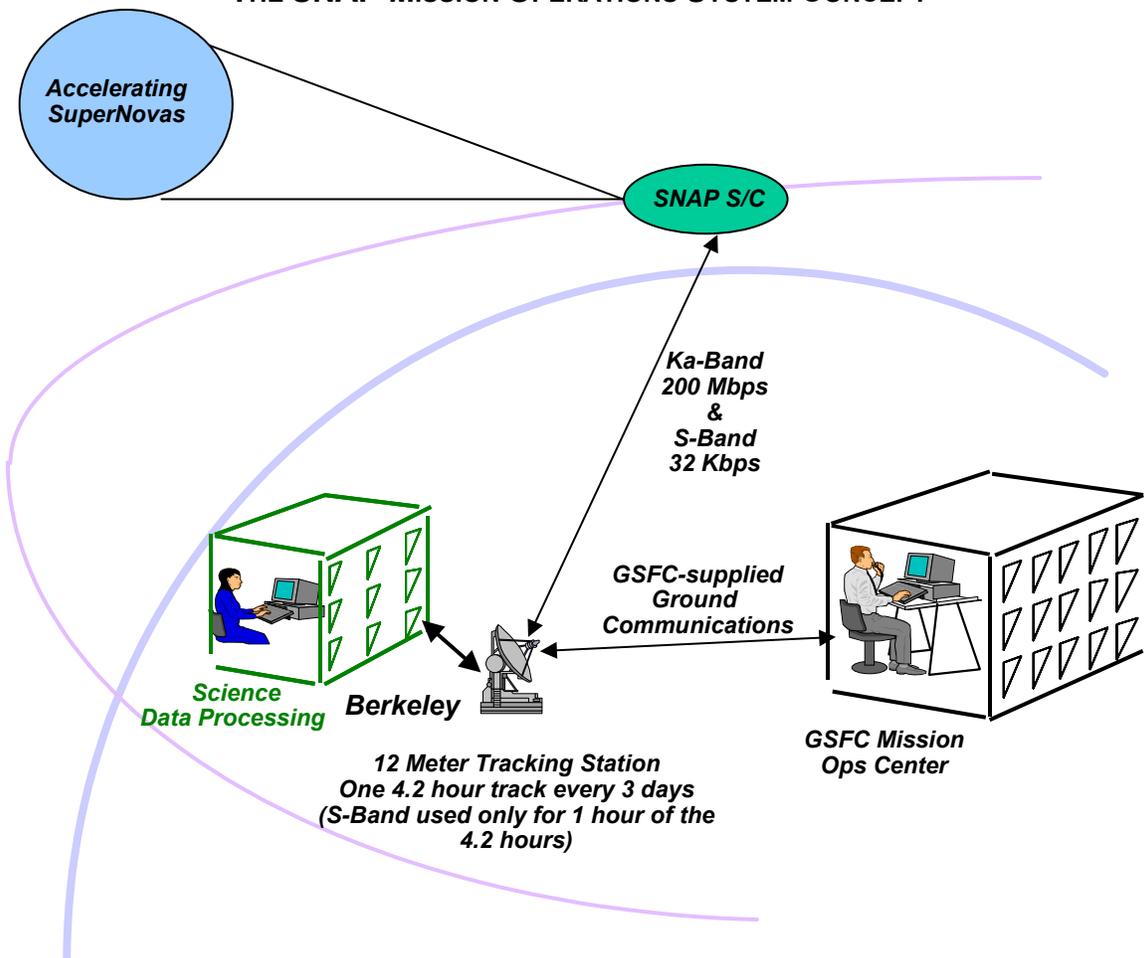
The major driving requirements on the Mission Operations System (MOS) and Ground System are described in this section.

- 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  $\pm 0.01$  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  $\pm 0.003$  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.

### POWER SUBSYSTEM

The power subsystem for the Super Nova Acc. Probe (SNAP) is required to operate for the mission duration of 5 years. The spacecraft will be placed 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<sub>2</sub> common pressure vessel (CPV) energy storage device, and power subsystem electronics. The components were 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 1m<sup>2</sup> and 5 standard 16 blade louvers of area equal to 1/4 m<sup>2</sup>, 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 ( $\lambda / \Delta \lambda$ ) 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 zodiacal 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  $\sim 100$  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 (200Å) Silicon (6150Å rest-frame) and Sulfur (5350Å 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  $\lambda/\Delta\lambda$  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 1600 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 2.0m<sup>2</sup> 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 0.006m<sup>2</sup> and average length of about 1.0m. [They could be heat pipes for better thermal conductance.]
- Radiator area is 2.0m<sup>2</sup>, 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 FPA nodes 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.
- Focal planes - high - The mosaics of detectors require special mechanical and electrical design that requires the signal and clock drivers to be derived from 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<sup>2</sup> 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, **Kepler, and SIRTf missions**. Total radiation during the mission is anticipated to be 30 krad behind 200 mils of aluminum with a radiation dose margin of 2 added.

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	
<i>Supernova/Acceleration Probe</i>	
<i>Orbiter</i>	
<i>Programmatic/Mission</i>	
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
<i>Spacecraft</i>	
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 maintenance) <span style="border: 1px solid black; padding: 2px;">See note on page 7</span>
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
<i>Costing</i>	
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

<i>Supernova/Acceleration Probe</i>		<i>Orbiter</i>		Legend						
<i>SYSTEMS WORKSHEET</i>		<i>Orbiter</i>								
Analyst:	Jason Andringa				Inputs from Sub					
Start Date:	#####				Inputs from Sys					
					Inputs from othe					
					Calculated					
Stabilization - cruise	3-Axis	Pointing Direction - cruise	N/A							
Stabilization - science	3-Axis	Pointing Direction - science	N/S ecliptic	Mission Duration	4.0					
				Max probe sun distance	1					
Pointing Control	0.030 arcsec	Radiation Total Dose, krad	30	Instrument Data Rate	1.5					
Pointing Knowledge	0.015 arcsec	Science BER	1.00E-06	Data Storage	3000					
Pointing Stability	0.030 relative pointing over 300 sec	Redundancy	Selected	Mission Data Volume	#####					
Determined by:	Science			Maximum Link Distance	60000					
		Technology Cutoff	2005/12	Return Data Rate	200 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) Steady (5 hours)	Mode 4 Power (W) TCM	Mode 5 Power (W) Launch + 6 hrs.	NASA TRL	To-day
<b>Payload</b>										
Instruments	61.9%	811.3	30%	1054.7	65.9	0.0	0.0	0.0	0	
<b>Payload Total</b>	61.9%	811.3	30%	1054.7	65.9	0.0	0.0	0.0		
<b>Bus</b>										
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	actually 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
<b>Bus Total</b>		499.7	22%	611.5	377.7	578.9	318.0	214.7	183.3	
<b>Spacecraft Total (Dry)</b>		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	
<b>Spacecraft with Contingency</b>		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
<b>Spacecraft Total (Wet)</b>		1967.2								
LV Adapter		0.0								
<b>Launch Mass</b>		1967.2								
<b>Launch Vehicle Capability</b>		3041.1	Delta 4040-12 *					Launch C3	-4.5	LV TRL
								Mission Unique LV Contingency	0%	
								Fairing type	standard	500
<b>Launch Vehicle Margin</b>		1073.9	35%					Fairing dia., m	?	Cost

## SYSTEM TABLE 3: MASS AND EQUIPMENT LIST

Mission: Supernova/Acceleration Probe  
 Element: Orbiter  
 System Level - Summary of Equipment List

## Instruments

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

## Attitude Determination and Control System

Component	Fit Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	T R L	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	6	0.003 deg/hr bias stability, 0.0001 deg/rt-hr rand walk, 4 150 Nms momentum storage, 0.1 to 0.2 Nm torque
Reaction Wheels #1	4	14.000	56.000	20%	67.200	105.0	22.0	Honeywell, HR16 Max	Various	6	Min step 0.0059 deg with HD 255:1 ratio, 113 Nm max 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	Assumed to control 4 motors per board
Gimbal Drive Electronics	2	0.590	1.180	10%	1.298	4.8		Spectrum Astro, DS-1 Gimbal Drive Electronics	DS-1	9	

## Command and Data System

Component	Fit Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	T R L	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 DTCL	2	0.500	1.000	30%	1.300	6.6	3.3	TBD		6	6U? 12 High Speed LVDS, 16 discrete inut, 16 Discrete 3U Command decoding 0 to
Mars05 ULDL	2	0.500	1.000	30%	1.300	3.9	2.0	TBD		6	100kbits/sec. BCH decoding 3U MIL-STD-1553B, 48 discrete
Mars05 GIF	2	0.350	0.700	30%	0.910	5.0	2.5	TBD		6	in, 48 discrete out. Analog signal. Analog Acquisition 3U non PCI, 48 voltage input, 16 passive
Mars05 AAC	4	0.350	1.400	30%	1.820	5.0	2.5	TBD		6	
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	Fit Units	Mass/ Unit (kg)	Total Mass (kg)	Contingency %	CBE + Contingency (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	T R L	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	T R L	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	T R L	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	8.855	8.855	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%	26.315			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	T R L	Description/Comments
X/Ka 0.5m diam High Gain Antenna	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/FF	2	2.500	5.000	20%	6.000	40.4	40.4	TBD		0	None
Ka-band (10Mbps) Xntr 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	Fit 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%	3.205	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	Fit 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.	TRL	Description/Comments
Instruments	1	811.3	30%	1054.7	0.0	65.9		0	
Attitude Determination and Control System	20	76.9	18%	91.1	166.8	71.0		6	
Command and Data System	18	45.3	29%	58.6	276.4	82.1		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	
<b>System Total</b>	<b>159</b>	<b>1309.9</b>	<b>28%</b>	<b>1682.6</b>	<b>589.5</b>	<b>322.4</b>		<b>6</b>	

### 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	50	60
R/W Dump (4 Yrs)	51	60
Maintain Inclination	180	153
Residuals		7

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.

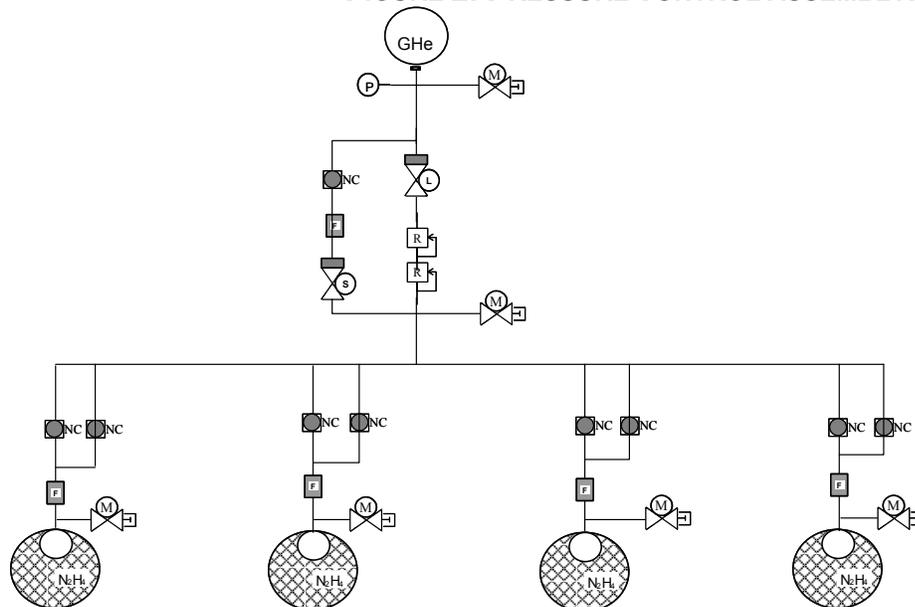
In order to minimize slosh, the monopropellant system utilizes 4 tanks with elastomeric diaphragms. Tanks are filled to near capacity at launch with liquid ultra pure anhydrous hydrazine ( $N_2H_4$ ). In operation, only one tank is used at a time. In this way, it is postulated that the net dynamic effect 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.



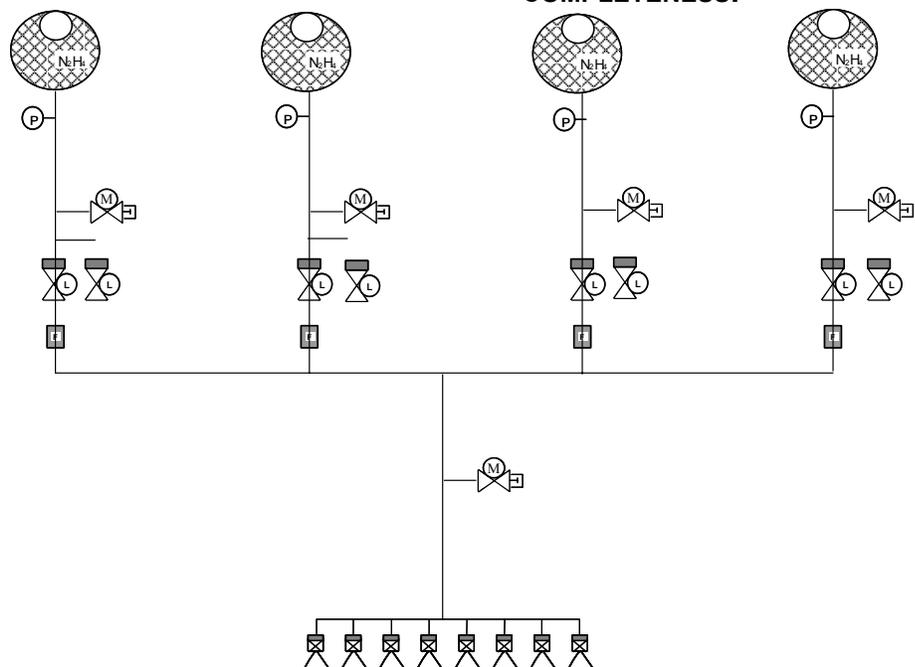
requires a typical opening power of 8W. These thrusters have extensive heritage including Intelsat, MPF, and MER.

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 nitrogen (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  $N_2H_4$  and usually  $N_2O_4$ . 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

#### SUBSYSTEM-LEVEL REQUIREMENTS

Pointing Stability: The goal for telescope pointing stability during imaging is within  $\pm 0.01$  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  $\pm 0.03$  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  $10.5 \mu\text{m}$  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  $\pm 0.27$  arcsec (1 sigma) over 1200 seconds. The requirement is stability to within  $\pm 0.67$  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  $\pm 1$  arcsec (3 sigma) in pitch and yaw and to within  $\pm 2$  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  $\pm 0.09$  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  $\pm 51$  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  $\pm 10$  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  $\pm 0.01$  arcsec (3 sigma) over 1200 seconds. This would support the goal of pitch and yaw pointing control to within  $\pm 0.03$  arcsec (3 sigma). The proposed requirement for roll (twist about the boresight) is accuracy within  $\pm 0.7$  arcsec (3 sigma)

over 1200 seconds. This would support the goal of pointing control to within  $\pm 0.81$  arcsec (3 sigma).

The proposed requirement for knowledge of the telescope boresight prior to calibration is accuracy within  $\pm 50$  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  $\pm 9$  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  $\pm 0.03$  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  $\pm 0.01$  arcsec (1 sigma) in pitch and yaw.

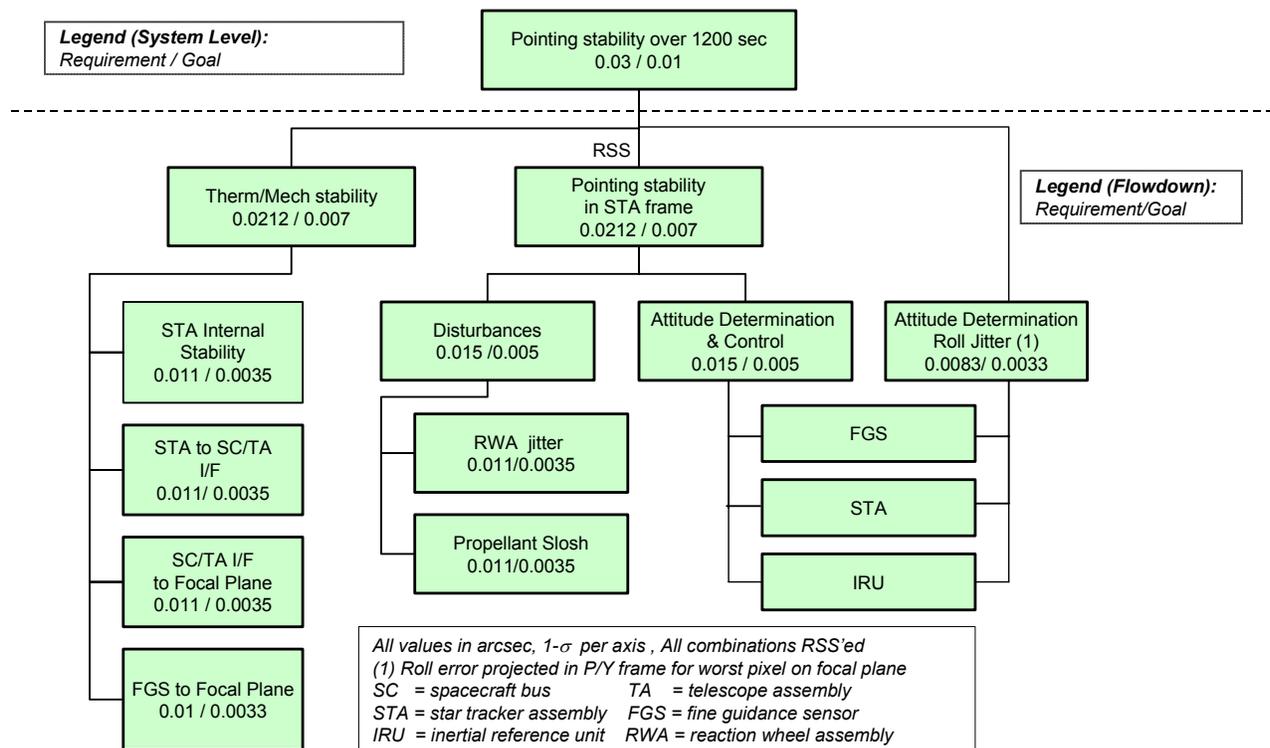
The values for thermal/mechanical stability requirements are in the ballpark of predictions for SIRTf. The values for disturbances are in the ballpark of predictions for SIRTf.

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 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  $\pm 0.03$  arcsec over 300 seconds, 3 sigma) is only about 40% looser than the requirement for the Hubble Space Telescope (HST).

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  $\mu\text{m}$  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  $\pm 10$  arcsec (3 sigma). During initial calibration of the bus attitude reference relative to the FGS's, looser pointing control to within  $\pm 50$  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. 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 pointing stability goal during imaging is to within  $\pm 0.03$  arcsec (3 sigma) in pitch and yaw over 1200 seconds for four images. The goal for roll is to within  $\pm 0.81$  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  $\pm 0.01$  arcsec (3 sigma). In that case, the allocation for all other errors would be  $\pm 0.02$  arcsec (3 sigma) if the intent were to meet the goal of pointing control within  $\pm 0.03$  arcsec (3 sigma). One of the largest error sources for HST was gyro noise, which induced errors of  $\pm 0.006$  to  $\pm 0.009$  arcsec (3 sigma) per axis in the closed loop control system. The expectation for SNAP is that gyro noise would likewise be a significant error source, but that the pointing control goal might nonetheless be met.

### **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  $\pm 50$  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  $\pm 1$  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).

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  $\pm 0.003$  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 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**

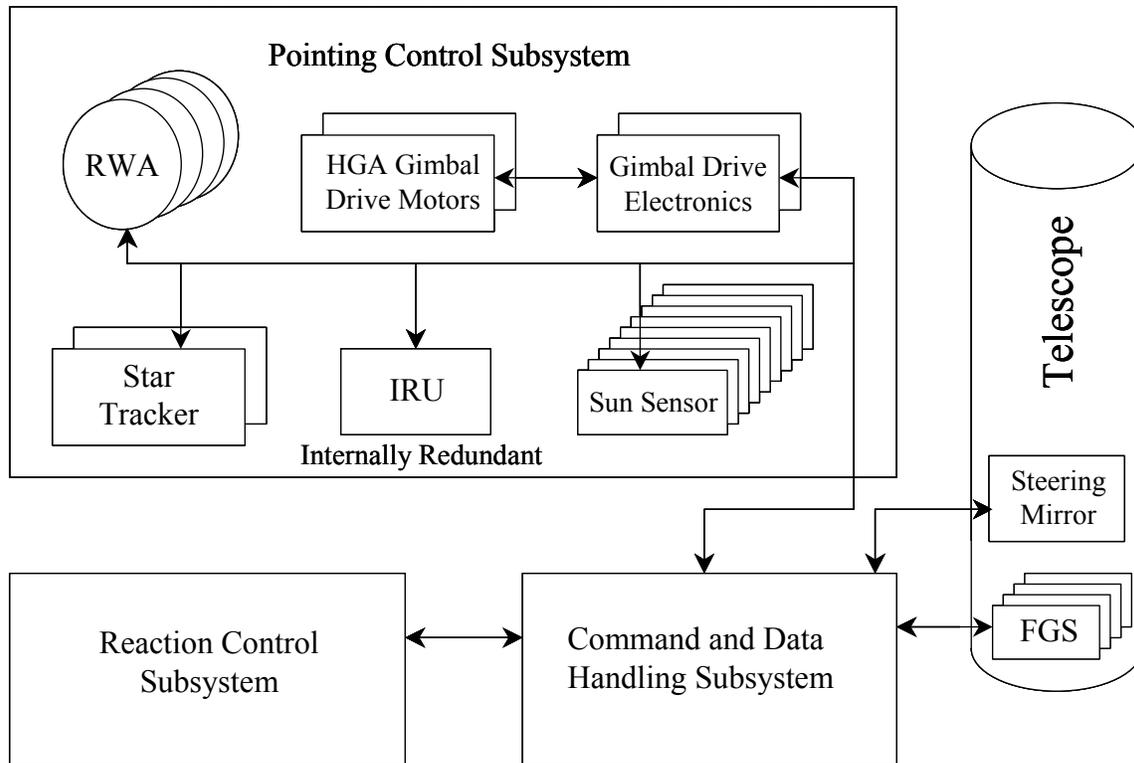
There is a rigid solar array fixed to the bus.

### **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  $9 \mu\text{N}/\text{m}^2$  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 axes 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  $< 1500 \text{ kg}\cdot\text{m}^2$ . 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 packetization. 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<sup>2</sup>.

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.

Cds snap Worksheet

Mission: Supernova/Acceleration Probe  
Element: Orbiter

**COMMAND AND DATA HANDLING WORKSHEET**

Estimated by: Mike Henry  
Last Update: 11/1/02

Number of CDS Strings:

Technology Cutoff: 2005/12

Use State Tool?

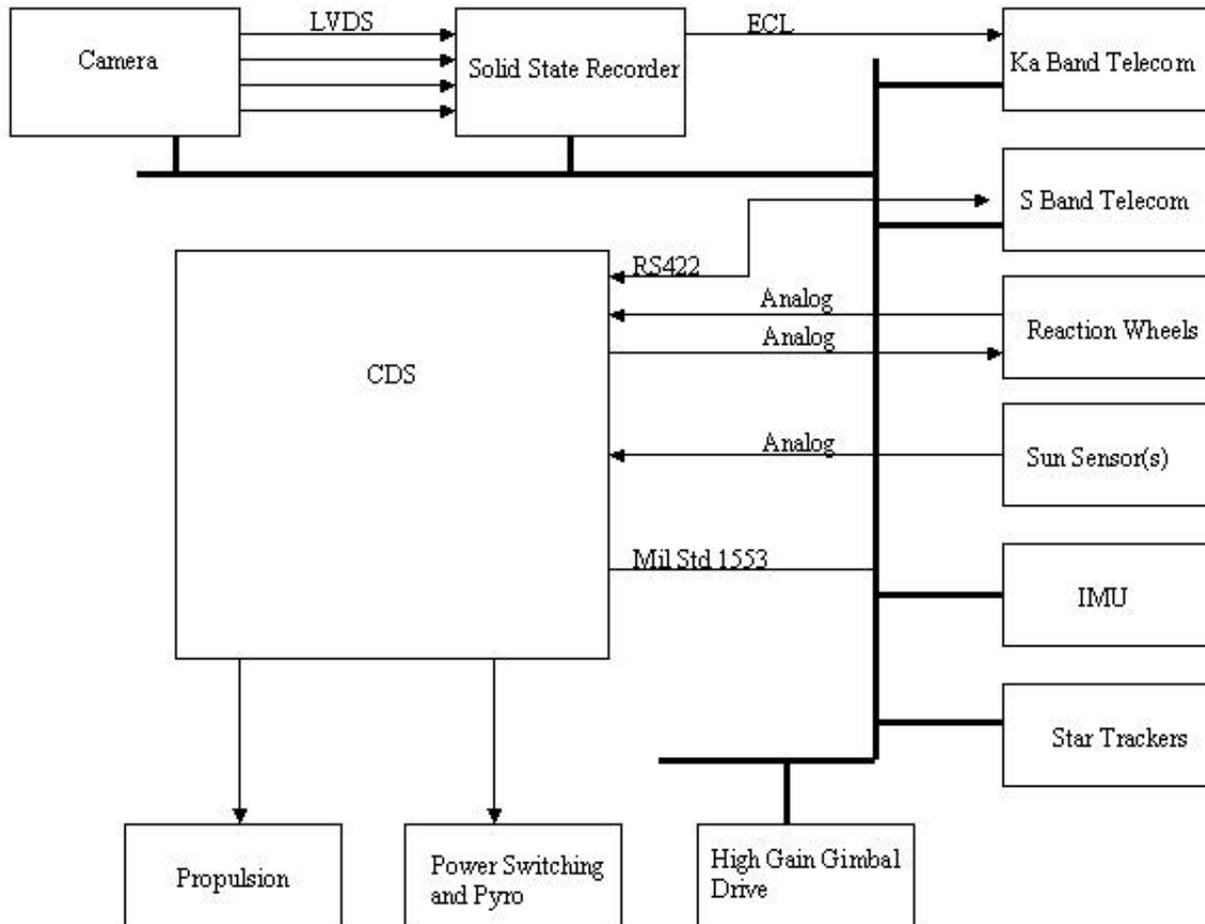
Unit	Mass [kg]	Contingency %	CBE+Cont. (kg)	Power per Unit (W)	Cost (\$K) Recurring	Cost(\$K) Development	Power (W)			Launch 8 hrs	NASA TRL	Comments
							(62 out of 72 hours)	Telecom (6 hours)	Standby (5 hours)			
<b>TOTAL</b>	<b>47.2</b>	<b>29%</b>	<b>61.2</b>	<b>276.4</b>	<b>8,350.0</b>	<b>no entry</b>	<b>163.9</b>	<b>276.4</b>	<b>96.4</b>	<b>36.4</b>	<b>6</b>	
SFC	2	1.04	15%	1.20	11.50	420.0	11.50	11.50	11.50	11.50	6	From JSC equipment sheet, RAD-750
NVM	2	0.64	15%	0.74	2.20	180.0	2.20	2.20	2.20	2.20	6	From equipment list - 2 Gbits - Latest Max
Mars05 CMIC-8	2	0.70	30%	0.91	2.20	200.0	2.20	2.20	2.20	2.20	6	3U Dual String Control card 6:02 Peer Re
Mars05 DTCI	2	1.00	30%	1.30	6.00	200.0	6.00	6.00	6.00	6.00	6	EU7 12 High Speed LVDS, 16 discrete in
Mars05 ULDL	2	1.00	30%	1.30	3.90	200.0	3.90	3.90	3.90	3.90	6	3u Command decoding 0 to 100kbits/sec. B
Mars05 GIF	2	0.70	30%	0.91	5.00	200.0	5.00	5.00	5.00	5.00	6	3U MI-STD-1553B, 48 discrete in, 48 disc
Mars05 AAC	4	1.40	30%	1.82	5.00	400.0	5.00	5.00	5.00	5.00	6	Analog Acquisition 3U non PCI, 48 voltage
Mars05 Back Plane	1	0.26	30%	0.34		50.0						
TCIU												
<b>TOTAL</b>	<b>1</b>	<b>36.60</b>	<b>30%</b>	<b>47.58</b>	<b>240.00</b>	<b>6,500.0</b>	<b>127.50</b>	<b>240.00</b>	<b>60.00</b>			

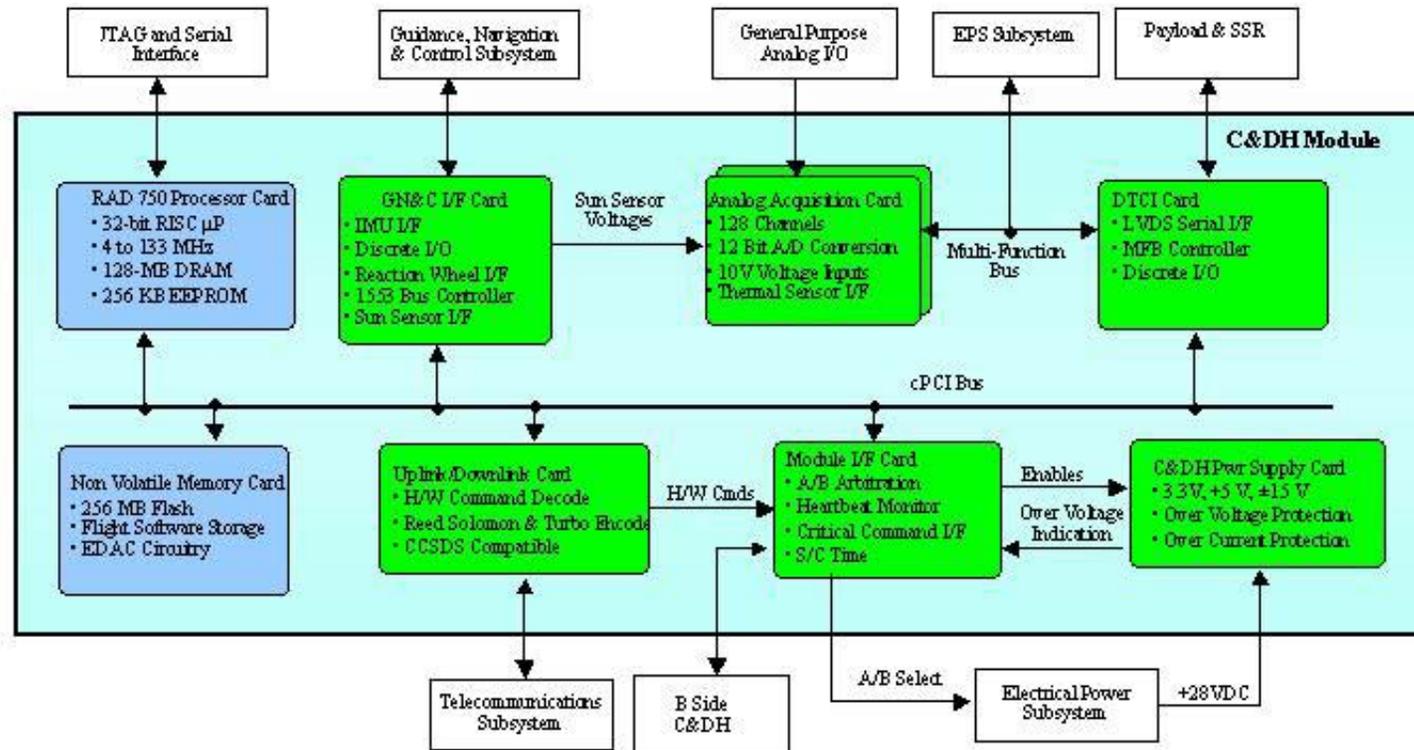
Shielding (kg):  30%

30.6 krad/s TID behind 100 mils of Aluminum

**BLOCK DIAGRAM**

SNAP CDS Interface Block Diagram





### **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 \$8M 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

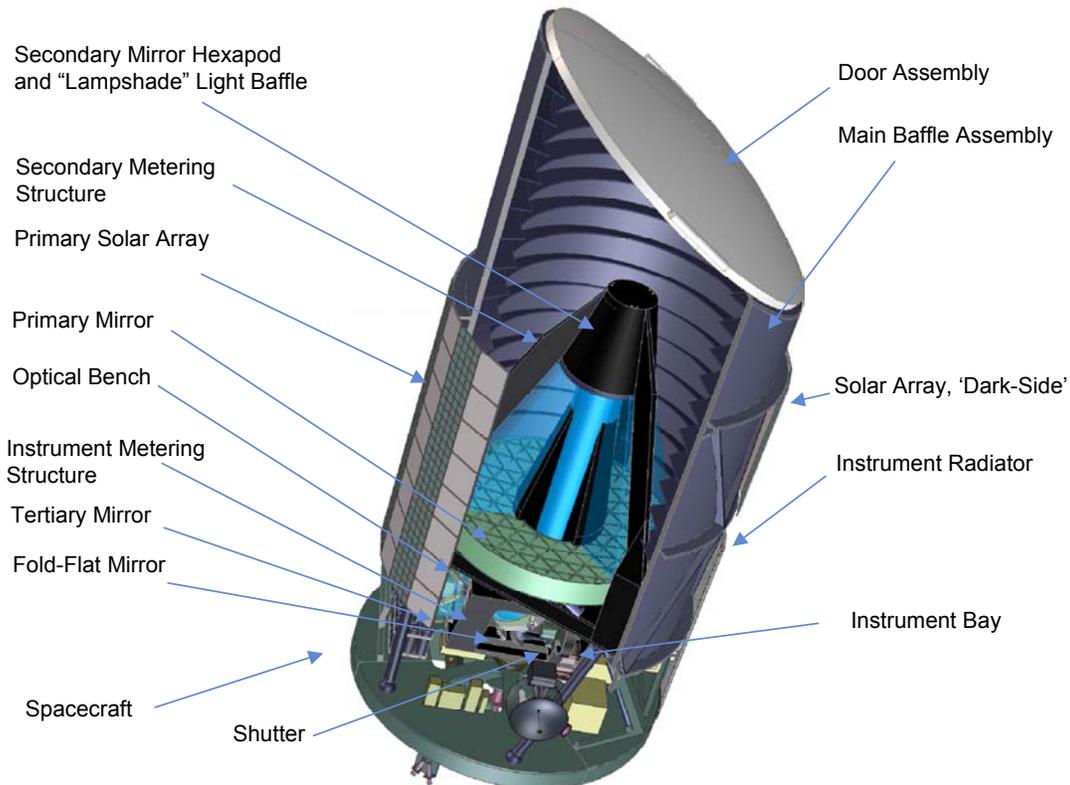
### 3.5 POWER

#### 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.

Array Design Summary						
Mass	8.89	kg	# Panels	2	Configuration	Fixed
Area	6.00	m <sup>2</sup>	Diameter	2.76	Target	Earth Orbit
Design	GaAs TJ Rigid		Power Density		Available	2001
Cost			<i>BOL</i>	215.77	TRL	9
			<i>EOL w/ cos</i>	118.86		

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<sup>2</sup> respectively.

The emergency solar array is designed to provide power to the spacecraft in the event that the spacecraft loses 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<sup>2</sup> respectively.

### ENERGY STORAGE DEVICE

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

<i>Design Summary and Application Notes</i>		
Capacity	90.0	Ampere-Hours
Mass	52.94	Kg
Battery Chemistry		Ni-H <sub>2</sub> (CPV)
TRL Level	9	
Calendar Life	12	Years
Min Temp	-15	Degrees C
Max Rate	63.0	Amps

Should also consider  
Li-Ion technology

A 12-cell 90 Ahr common pressure vessel (CPV) Ni-H<sub>2</sub> 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

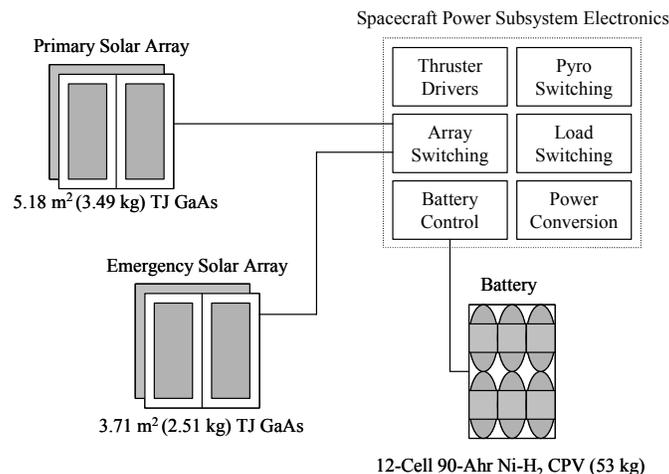
FIGURE 4. SPACECRAFT POWER SUBSYSTEM ELECTRONICS DESIGN SUMMARY.

Power Electronics Design Summary				Statistics	
				Boards	18
System Type:	Large Orbiter	Class A/B	Functional Redundant	Mass	14.4 Kg
SA w/ 2nd Batt - String Switcher	6U Boards			Power	756.6 Watts
Switching:	Dumb Solid Stat:Power Conversion COTS			Cost	\$ 5,300 K
Minimum TRL:	6				

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<sub>2</sub> 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. Dual strings are required. Li-Ion batteries are only considered for three-year missions.

### NI-H<sub>2</sub> IPV (INDIVIDUAL PRESSURE VESSEL)

A 23 cell 94 Ahr Ni-H<sub>2</sub> 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<sub>2</sub> 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 1m<sup>2</sup> and 5 standard 16 blade louvers of area equal to ¼ m<sup>2</sup>, 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 maneuvers or accidentally, the temperatures of the focal 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

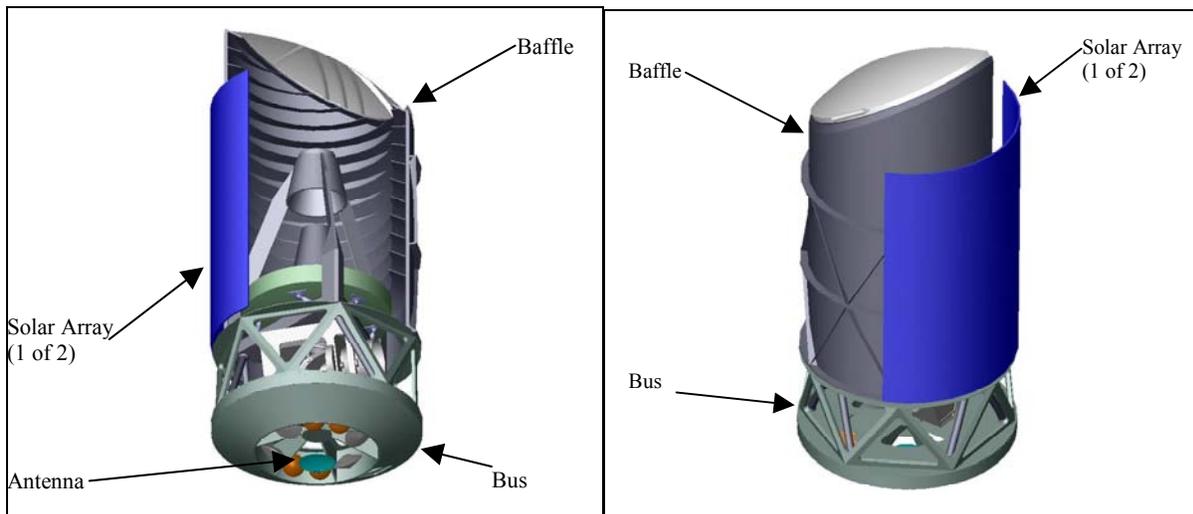
The primary purpose of the structures subsystem is to provide mounting and structural support for all spacecraft subsystems. In this case the principal structural supports are for the telescope and the baffle associated with it. In addition, the baffles need to support the solar arrays and the doors 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  $360^\circ$  about the line of sight of the telescope and  $\pm 30^\circ$  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.



#### 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**

Materials: Composites	Units	Mass (kg)		
		Best Est.	Contingency	CBE + Conting.
<b>TOTAL (less Cabling, LV adapters)</b>		<b>144.0</b>	<b>30%</b>	<b>187.2</b>
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
<b>Cabling</b>		<b>42.4</b>	<b>30%</b>	<b>55.1</b>

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**

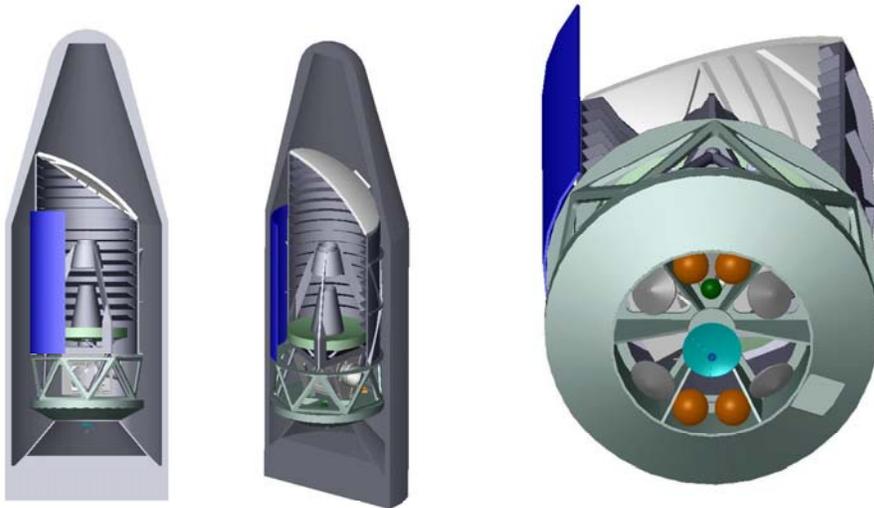
Structures technology risk is basically low per usual Structures design philosophy; assumed to employ current best technology, standard design procedures and testing, and appropriate stress margins.

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 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.  $\pm 5^\circ$  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 1/2 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
<b>Total</b>	<b>21</b>	<b>18.7</b>		

### 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)

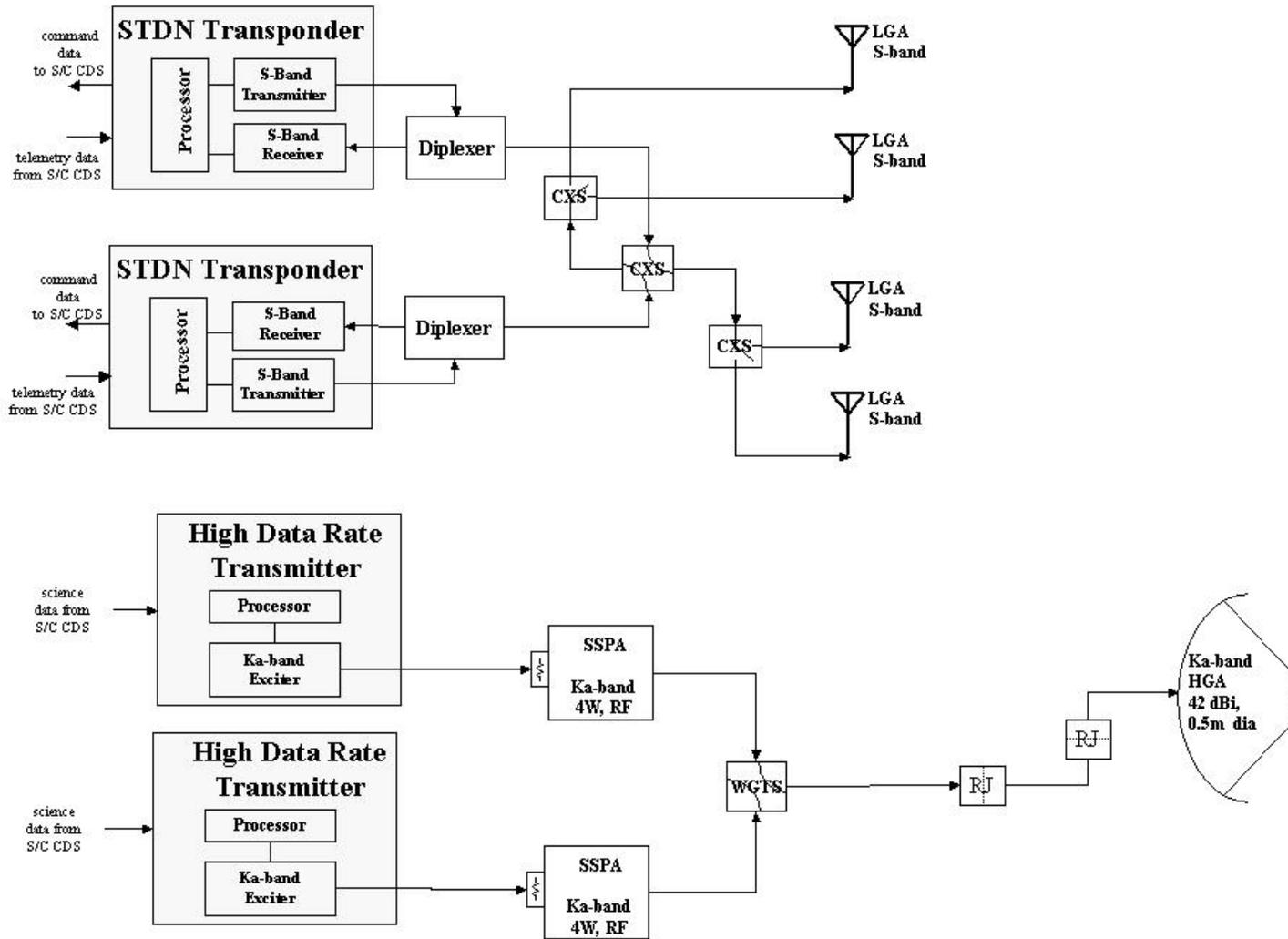
Power Mode #3 (Standby): S-band transponder is in receive mode = 5.4W  
Ka-band transmitter is off

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.

## **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., AACCS, 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 AACCS 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 < 0.03 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 3.2 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 detectors
  - CCDs
  - Electronics: ASIC development is required
  - Filters
  - Spectrograph Miscellaneous:
  - 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.
  - Development phase C/D: 4 years.
  - Mission operation phase E: 4 years.

○

## 4.0 PROGRAMMATICS

### 4.1 PROGRAMMATICS

#### SUBSYSTEM-LEVEL REQUIREMENTS:

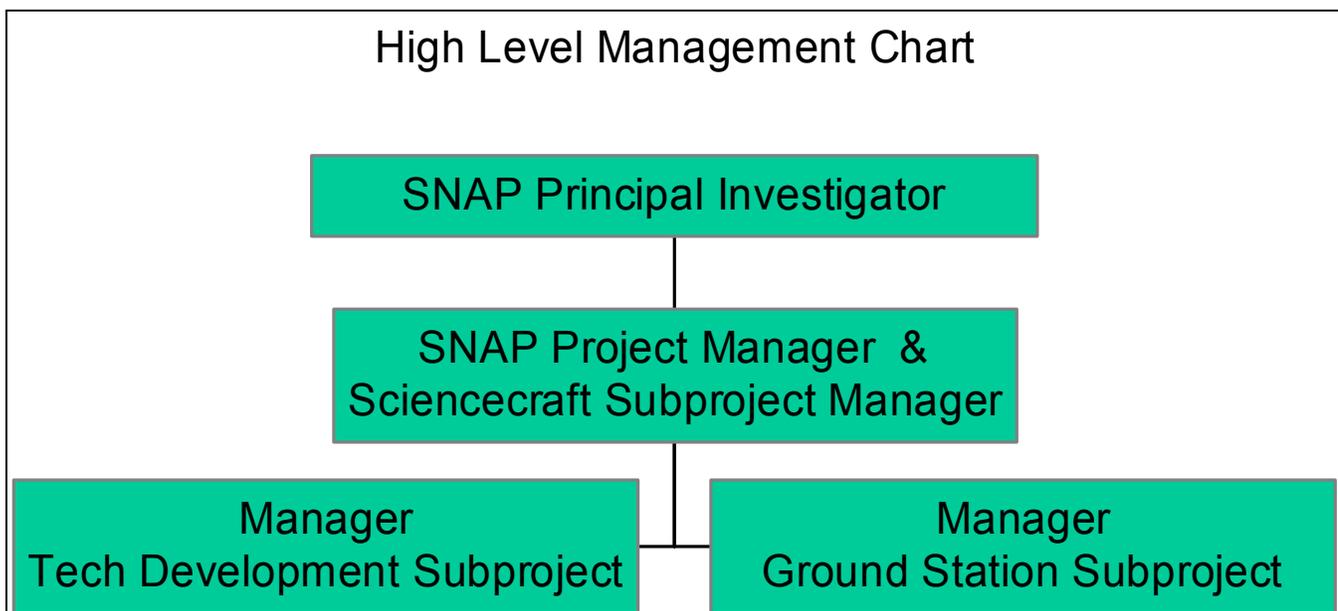
Programmatically this mission is 3 projects: A technology development subproject for the detectors, a sciencecraft subproject, and a ground station upgrade subproject. The overriding programmatic requirement is that each subproject be completed prior to launch.

#### 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 is being developed
- Focal planes near this size have been flown before, but none quite this large.
- The ACS must be very stable; it is estimated that it will take 1 – 1.5 years to develop.

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.

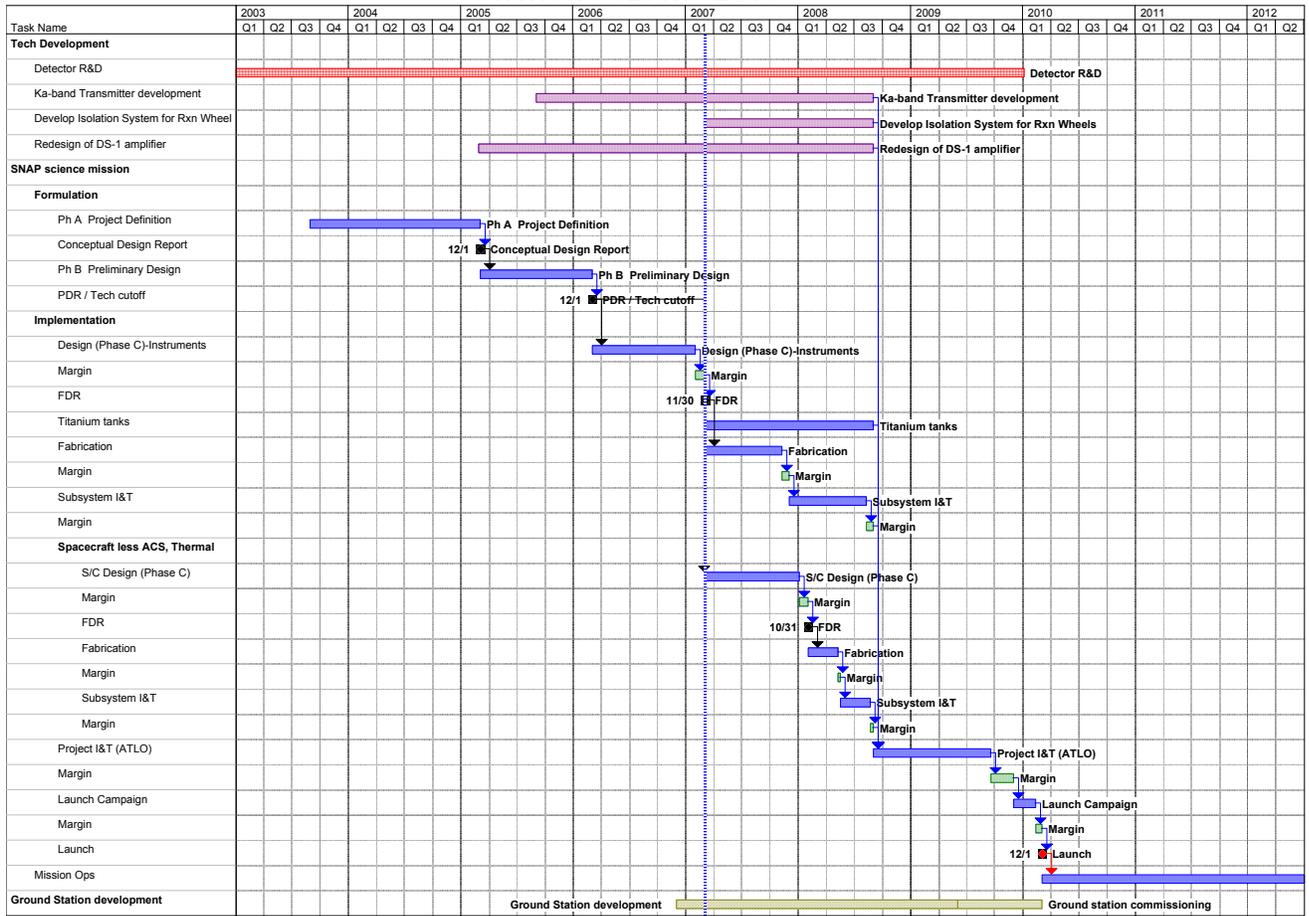
The overall implementation phases are shown as 48 months from PDR to launch. To save money, only the ACS and thermal parts, and the systems engineers start on time. The rest of the project, predominantly the bus, will start a year after the early

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

.

**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.

## 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 procure 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

- ACS requirements are subtle and it is important to insure that the S/C vendor fully understands them.

### 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 baffle 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<sub>2</sub> battery is the heaviest cell design. It has the highest TRL level of all Ni-H<sub>2</sub> designs and is suitable for use in single string configuration. Cell bypass circuitry is required.

##### CPV

The CPV Ni-H<sub>2</sub> 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<sub>2</sub> 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 2<sup>nd</sup> 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  $N_c = 000.189$

Tip to Base Temperature Ratio = 0.931

**FIN HEAT LOSS =12.27-W/M**

Fin Efficiency = 0.799. An area equal to **1.3m by 1.3m = 1.69 m<sup>2</sup>** will dissipate **16.5 W**

Temperature Profile Data is Given Below

x/L	T/T <sub>b</sub>	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								5.999E+04		Range, km
Ka-band/0.5 m HGA/57% eff./0.20 deg. pointing error								0.0004		Range, AU
New Ka-band Berkeley Ground Station								0.00		OWLT, hrs
Hot body noise = 0 K										
1 way								0.00		SEP, deg
Thru channel/RS-Convolutional (7,1/2)/PB=1.E-6		Carrier Loop Bandwidth = 150.0 Hz.						10		Elev. Angle, deg
		Bit Rate = 240000000 bps								
Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	S			
<b>TRANSMITTER PARAMETERS</b>										
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				
<b>PATH PARAMETERS</b>										
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	
<b>RECEIVER PARAMETERS</b>										
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			
<b>TOTAL POWER SUMMARY</b>										
# Total Rcvd Pwr (Pt)	dBm				-75.77	0.2236	G	diplexed	LNA Selection	
								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	dB-Hz				94.30	0.2236	G			
<b>CARRIER PERFORMANCE at Req. Pt/No</b>										
# Tim 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	
# Pct/No (39+14+15+16)	dB-Hz				80.02	0.2993	T			
# Carrier Loop Bandwidth, BI	dB-Hz	21.76	0.00	0.00	21.76	0.0000	T	150.0	Carrier BI, 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	
								0.000	p.n. solar, rad^2	
								0.000	phase noise variance, rad^2	
<b>SUBCARRIER PERFORMANCE at Req. Pt/No</b>										
# SubCar. L. SNR	dB				67.92					
# Required Loop SNR	dB				20.00			5000	SubCarr BI, mHz	
# SubCarrier Loop SNR Margin	dB				47.92			10	SubCarr window f.	
<b>SYMBOL LOOP PERFORMANCE at Req. Pt/No</b>										
# Sym. Loop SNR	dB				53.25			5000	Sym BI, mHz	
# Required Loop SNR	dB				15.00			10	Sym window f.	
# SubCarrier Loop SNR Margin	dB				38.25			Optimize		
<b>TELEMETRY PERFORMANCE at Required Pt/No</b>										
# Tim Data Supp	dB	-0.44	0.07	-0.07	-0.44	0.0008	T	72.0	tim 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		D	0.005	BER at BVR output	
# Required Pt/No	dB-Hz				90.24	0.2508	U			
# Performance Margin (39-13)	dB				4.06					

FIGURE A-2 S-BAND NOMINAL ENGINEERING DOWNLINK

3W SSPA								5.999E+04 Range, km	
S-band LGA - 5 deg. off-point								0.0004 Range, AU	
Berkeley S-band Ground Station								0.00 OWLT, hrs	
Hot body noise = 0 K								0.00 SEP, deg	
1 way								10 Elev. Angle, deg	
Tlm channel/RS-Convolutional (7,1/2)/PB=1.E-6								Carrier Loop Bandwidth = 13.7 Hz.	
								Bit Rate = 45052 bps	
Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	S		
<b>TRANSMITTER PARAMETERS</b>									
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			
<b>PATH PARAMETERS</b>									
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
<b>RECEIVER PARAMETERS</b>									
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		
<b>TOTAL POWER SUMMARY</b>									
# Total Rcvd Pwr (Pt)	dBm				-111.14	0.2048	G	diplexed	LNA Selection
(5+6+7+8+9+10)									
# Noise Spec Dens	dBm/Hz	-174.65	-0.02	0.03	-174.65	0.0001	G	Optimize	
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	dB-Hz				63.50	0.2049	G		
(11-12)									
<b>CARRIER PERFORMANCE at Req. Pt/No</b>									
# 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
								0.000	p.n., solar, rad^2
								0.002	phase noise variance, rad^2
<b>SUBCARRIER PERFORMANCE at Req. Pt/No</b>									
# SubCar. L. SNR	dB				48.18			1000	SubCarr Bl, mHz
# Required Loop SNR	dB				20.00			1	SubCarr window f.
# SubCarrier Loop SNR Margin	dB				28.18				
<b>SYMBOL LOOP PERFORMANCE at Req. Pt/No</b>									
# 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	
<b>TELEMETRY PERFORMANCE at Required Pt/No</b>									
# 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		D	0.005	BER at BYR 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

Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	Shape		
								5.999E+04	Range, km
								0.0004	Range, AU
								0.00	OWLT, hrs
								0.00	SEP, deg
								10	Elev. Angle
2000.0000									
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	8	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)	dBm				-103.91	0.3041	G		
(5+6+7+8+9+10+11)									
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	BI, 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								1.646E+05	Range, km
S-band LGA - 5 deg. off-point								0.0011	Range, AU
BGS 10m S-band station								0.00	OWLT, kus
Hot body noise = 0 K								0.00	SEP, deg
1 way								10	Elev. Angle, deg
Tlm channel/RS-Convolutional (7,1/2)/PB=1 E-6								Carrier Loop Bandwidth = 8.3 Hz.	
								Bit Rate = 10000 bps	
Link Parameter	Unit	Design Value	Fav Tol	Adv Tol	Mean Value	Var	S		
<b>TRANSMITTER PARAMETERS</b>									
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			
<b>PATH PARAMETERS</b>									
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	2299.07	Freq, MHz
<b>RECEIVER PARAMETERS</b>									
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	54	BGS Antenna
# Polarization Loss	dB	-0.05	0.00	0.00	-0.05	0.0000	U		
<b>TOTAL POWER SUMMARY</b>									
# Total Rcvd Pwr (Pt)	dBm				-119.91	0.2048	G		
(5+6+7+8+9+10)									
# 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	dB-Hz				54.74	0.2049	G		
(11-12)									
<b>CARRIER PERFORMANCE at Req. Pt/No</b>									
# 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, BI	dB-Hz	9.18	0.00	0.00	9.18	0.0000	T	8.3	Carrier BI, Hz
# Carrier Loop SNR	dB				25.34	0.4177	U	-20	Carrier Phase Noise
# Required Carrier Loop SNR	dB				10.00	0.0000	D		
# Carrier Loop SNR Margin	dB				15.34	0.4177	U		Type 2. SuperCritically Damp
<b>SUBCARRIER PERFORMANCE at Req. Pt/No</b>									
# SubCar. L. SNR	dB				41.64				p.n. var, tx contribution, rad^2
# Required Loop SNR	dB				20.00				p.n. var, loop snr, rad^2
# SubCarrier Loop SNR Margin	dB				21.64				p.n., solar, rad^2
<b>SYMBOL LOOP PERFORMANCE at Req. Pt/No</b>									
# Sym. Loop SNR	dB				32.84				phase noise variance, rad^2
# Required Loop SNR	dB				15.00				SubCarr BI, MHz
# SubCarrier Loop SNR Margin	dB				17.84				SubCarr window f.
<b>TELEMETRY PERFORMANCE at Required Pt/No</b>									
# 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	0.2855	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

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	Level 1	Basic principles observed and reported (G)
	Level 2	Technology concepts/applications formulated (G)
Feasibility Research	Level 3	Analytical and experimental critical function and/or characteristic proof-of-concept (G)
	Level 4	Component and/or breadboard validation in laboratory (G)
Technology Development	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)
Technology Validation	Level 7	System validation model demonstrated in space (S)
System/Subsystem Development	Level 8	Actual system completed and "flight qualified" through test and demonstration (G or S)
	Level 9	Actual system "flight proven" through successful mission operations (S)
System Test, Launch and Ops		

**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 in 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
<ul style="list-style-type: none"> <li>• Designed for specific hardness level</li> <li>• Total dose: &gt;200 krad to &gt;1 Mrad</li> <li>• SEU threshold LET: 80150 MeV/mg/cm<sup>2</sup></li> <li>• SEU error rate: 10<sup>-10</sup> to 10<sup>-12</sup> errors/bit-day</li> <li>• Latchup: Si on insulator technologies; no latchup problems</li> </ul>	<ul style="list-style-type: none"> <li>• Hardness offered as a by-product of the design</li> <li>• Total dose: 20 krad to &gt;50 krad</li> <li>• SEU threshold LET: 20 MeV/mg/cm<sup>2</sup> (typ)</li> <li>• SEU error rate: 10<sup>-7</sup> to 10<sup>-8</sup> errors/bit-day</li> <li>• Latchup: Customer evaluation and risk</li> </ul>	<ul style="list-style-type: none"> <li>• Hardness limited by inherent process and design; customer risk</li> <li>• Total dose: 2 krad to &gt;10 krad</li> <li>• SEU threshold LET: 5 MeV/mg/cm<sup>2</sup> (typ)</li> <li>• SEU error rate: 10<sup>-5</sup> errors/bit-day</li> <li>• Latchup: Customer evaluation and risk</li> </ul>

**APPENDIX—ACRONYMS**

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