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Advanced X-Ray Timing Array Mission: Conceptual Spacecraft Design Study

R.C. Hopkins, L. Johnson, H.D. Thomas, and C.A. Wilson-Hodge
Marshall Space Flight Center, Huntsville, Alabama

M. Baysinger and C.D. Maples
Qualis Corporation, Huntsville, Alabama

P.D. Capizzo
Raytheon Company, Huntsville, Alabama

L.L. Fabisinski
International Space Systems, Inc., Huntsville, Alabama

L. Hornsby and K.S. Thompson
Jacobs ESTS Group, Huntsville, Alabama

J.H. Miernik
ERC, Inc./Jacobs ESTS Group, Huntsville, Alabama

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National Aeronautics and
Space Administration

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LIST OF ACRONYMS AND SYMBOLS

ACS	attitude control system
AIAA	American Institute of Aeronautics and Astronautics
AITECH	AITech International
Al	aluminum
ATK	Alliant Techsystems, Inc.
AXTAR	Advanced X-ray Timing Array
BAT	Burst Alert Telescope
BH	black hole
CCAFS	Cape Canaveral Air Force Station
CG	center of gravity
CMG	control moment gyros
COMM	communication
COTS	commercial off the shelf
Cu	copper
DAS	Debris Assessment Software
FCI	Fluid Components International
FEMAP	Finite Element Modeling Analysis and Postprocessing
GaAs	gallium arsenide
GSFC	Goddard Space Flight Center
IDS	Instrument Data System

LIST OF ACRONYMS AND SYMBOLS (Continued)

IMU	Inertial Measurement Unit
ITJ	improved triple junction
LATA	Large Area Timing Array
Li	lithium
MEL	master equipment list
MGA	mass growth allowance
MIDEX	medium-class Explorer
MLI	multilayer insulation
MOS	margin of safety
MPS	main propulsion system
MSFC	Marshall Space Flight Center
NS	neutron star
ORS Sat-1	operationally responsive space satellite 1
pyro	pyrotechnic
RCS	reaction control system
RSDO	Rapid Spacecraft Development Office
RW	reaction wheel
RXTE	Rossi X-Ray Timing Explorer
SAA	South Atlantic Anomaly
Si	silicon

LIST OF ACRONYMS AND SYMBOLS (Continued)

SM	Sky Monitor
STK	Satellite Took Kit
TDRSS	Tracking and Data Relay Satellite System
TM	Technical Memorandum
TRL	Technology Readiness Level
UV	ultraviolet
VME	Versa Module Europa
ZnO	zinc oxide

NOMENCLATURE

α	absorptivity
g	acceleration due to gravity at standard sea level
ΔV	delta velocity (change in velocity of an orbital maneuver)
m_0	initial mass
ε	emissivity
ε^*	effective emissivity

TECHNICAL MEMORANDUM

ADVANCED X-RAY TIMING ARRAY MISSION: CONCEPTUAL SPACECRAFT DESIGN STUDY

1. INTRODUCTION

In the spring of 2010, the Advanced Concepts Office of the NASA Marshall Space Flight Center (MSFC) completed a conceptual spacecraft design to meet vehicle requirements for the Advanced X-Ray Timing Array (AXTAR) science mission. The goal was to design a spacecraft that provides power and data handling for the instruments, communication (COMM), pointing, station keeping, momentum unloading, thermal control, and end-of-life disposal. Since the science team was interested in both a minimum-mass design and one that carried the maximum possible number of instruments, two separate conceptual designs were created. The approach was to first complete a conceptual spacecraft design that could accommodate a minimum-required instrument suite as specified by the science team and select a possible launch vehicle (or family of vehicles), and then complete at least one iteration of a design exercise in which the maximum number of science instruments were accommodated by a spacecraft launched on the Falcon 9 launch vehicle. The smaller configuration, which was the primary product of the study, is a closed design, while the more massive configuration was not iterated to closure due to time constraints. Nevertheless, the larger configuration is representative of a science package designed to maximize the capability of a Falcon 9 or similar launch vehicle.

An introduction to the science mission, the science instruments, and the details of the spacecraft requirements and design are included in the sections of this Technical Memorandum (TM).

2. SCIENCE MISSION SUMMARY

This TM presents a brief summary of AXTAR's science mission and requirements. See reference 1 for more details.

AXTAR's primary science goals focus on extracting fundamental physics from neutron stars (NSs) and stellar mass black holes (BHs) (the end products of stellar evolution for massive stars). An NS is composed of ultradense matter with the mass of the Sun compressed into a star the size of a city. Stellar mass BHs are thought to contain the mass of several to tens of suns compressed into a singularity (a single point) surrounded by an event horizon, where the escape velocity from the BH equals the speed of light. Orbital periods near the BH event horizon are on millisecond timescales and are set by the BH's mass, angular momentum, and the laws of relativistic gravity. Maximum rotation periods of NSs are also on millisecond timescales and are set by the equation of state, the relationship between the star's mass and radius, and of ultradense matter in its interior.

With its large area and high time resolution, AXTAR's Large Area Timing Array (LATA) will address the following key science questions raised in the 2010 Astronomy and Astrophysics Decadal Survey:

(1) How do BHs work and influence their surroundings?

(2) How do rotation and magnetic fields affect stars?

(3) What controls the masses, spins, and radii of compact stellar remnants?

The Sky Monitor (SM) instrument on-board AXTAR will serve as a trigger for LATA observations and will monitor hundreds of x-ray sources for a wide variety of primary science investigations. Partnering with wide-angle astronomy observatories at other wavelengths, the SM will support time-domain astronomy including surveying the sky at many wavelengths (stressed as important science in the 2010 Astronomy and Astrophysics Decadal Survey).

Reference 1 gives the science objectives that set AXTAR's mission requirements in detail. A few key points are summarized in this TM. AXTAR's required effective area is set by two key science objectives: (1) Achieving a 5–10% NS radius measurement and (2) achieving a 0.1% rms fractional amplitude detection threshold for high-frequency, quasi-periodic oscillations in BHs. Its targets will be bright galactic sources, requiring that the detector and data system handle high data rates without impacts from dead time, pileup, or data losses. Many of the target observations need to be triggered when a source is in a particular state, and a sensitive SM with maximal coverage of the accessible sky is required because x-ray transients may occur anywhere in the sky. The Rossi X-Ray Timing Explorer (RXTE) has shown that AXTAR's targeted events are often short lived, requiring the capability to reschedule in a matter of hours and slew to new coordinates expeditiously.

3. SCIENCE INSTRUMENTS

While some basic information on the science instruments is included for completeness, this TM focuses on the spacecraft design. Refer to reference 1 for details regarding the science instruments.

The two science instruments for AXTAR are the LATA and the SM (which is a collection of cameras). The function of the SM is to view as much of the sky as possible, detect x-ray sources, and trigger the spacecraft to slew and point the LATA at the x-ray source for detailed observations. The function of the Instrument Data System (IDS) is to process the data from the science instruments. A short summary of the instrument parameters is found in section 4.

The LATA is the primary AXTAR science instrument and is composed of several coaligned super modules (see fig. 1). From top to bottom, the components in figure 1 are as follows: The collimator, a light shield, silicon (Si) detectors, an interposer board, a digital board, and a mounting plate that provides support and shielding. The 5×5 array of Si detectors (each 10×10 cm) results in a super module length and width of approximately 60 cm due to space between the detectors and border material. With the collimator installed, the height of each super module is approximately 20 cm. As stated above, all super modules are coaligned to within 1 arc-min, although they do not have to be coplanar. The science team desired a spacecraft configuration that contained at least 20 super modules.

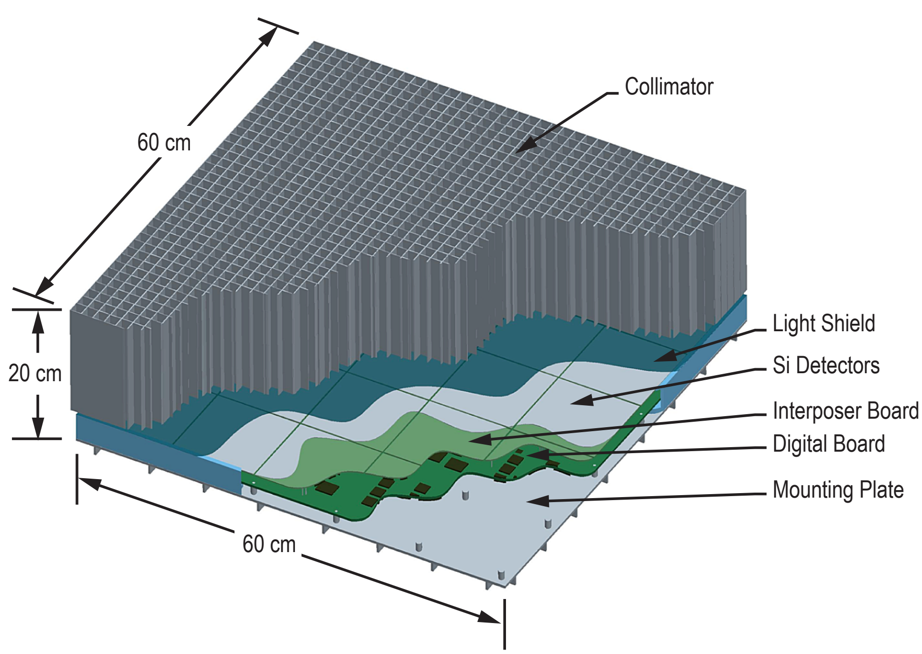

Figure 1. Cutaway rendering of a LATA super module.

The SM observes a broad portion of the sky and provides the data necessary to repoint the LATA at an active x-ray source. The SM is composed of a maximum of 32 cameras, each pointing to a different part of the sky except for a small area of overlap. Figure 2 shows an illustration of a single SM camera and a cluster of seven cameras. Each SM camera contains a four-element detector plane (see fig. 2(a)) consisting of the same Si pixel detectors used for the LATA and is topped with a two-dimensional coded mask. The base of each camera is 25×25 cm, and the top is 30-cm square, with a total height of 25 cm. The spacecraft carries a maximum of 32 SM cameras in an arrangement similar to a soccer ball. Each camera looks in a different direction, resulting in total sky coverage.

The SM cameras do not have to be collocated on the spacecraft, but can be placed on various locations for convenience and packaging. The science team desired a minimum of seven SM cameras, with the central field of view of the cluster being coaligned with the LATA. If another cluster of seven could be added, the next priority was to point the cluster in the opposite direction of the first. This is because the x-ray sources predominately lay along the galactic plane. If room and power were available, additional SM cameras could then be placed on the spacecraft, up to a maximum of 32.

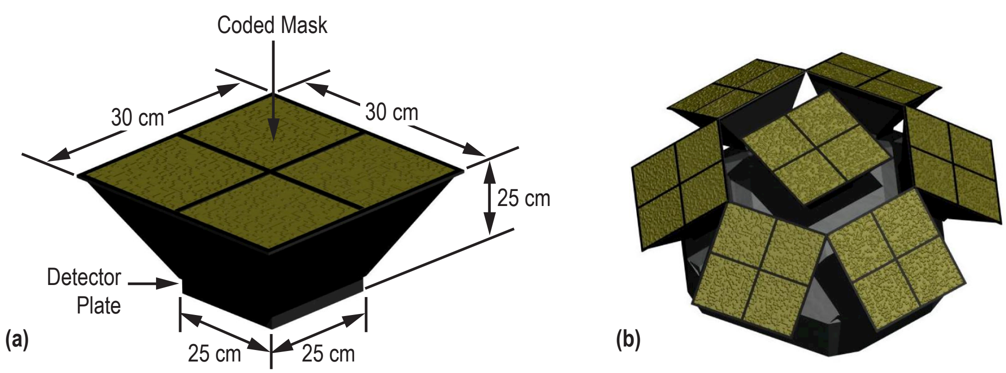

Figure 2. Rendering of (a) an SM camera and (b) a cluster of seven cameras.

4. MISSION AND SPACECRAFT REQUIREMENTS

The top-level requirements for the AXTAR design study are listed in table 1. Major parameters for the science instruments (the LATA, SM, and IDS) are also listed for completeness. The IDS is not actually an instrument, but it is included as part of the science payload because it processes the data from the SM and LATA. The spacecraft structure and subsystems were designed to meet these requirements and included a 30% mass growth allowance (MGA) margin on both dry mass and power for the spacecraft and instruments.

Table 1. Spacecraft and mission requirements and instrument data.

Parameter	Required (Desired or Bounding) Value
Spacecraft and Mission	
Anticipated launch year	2019 (used as a guide for orbit lifetime estimates)
Orbit altitude	Approximately 600 km (study output), circular
Orbit inclination	28.5° or less (as low as possible)
Spacecraft lifetime	3 yr
Consumables	5 yr
Orbit lifetime	10 yr
Pointing accuracy	1 arc-min (or better)
Pointing knowledge	5 arc-s (or better)
Maximum slew rate	180° in 30 min
SM Camera (each)	
Mass	2 kg + 2 kg per telemetry hub
Power	4 W + 9 W per telemetry hub
Dimensions	30 × 30 × 25 cm
Quantity	7 minimum (32 maximum)
Thermal requirement	–400 °C to +100 °C (detector plane)
Alignment	32 faces and vertices of a dodecahedron.
LATA Super Module (each)	
Mass	30 kg
Power	30 W
Dimensions	60 × 60 × 20 cm
Quantity	20 (as many as possible)
Thermal requirement	–400 °C to + 100 °C (detector plane)
Alignment	All coaligned within 1 arc-min
IDS	
Mass	20 kg
Power	40 W
Dimensions	30 × 30 × 20 cm
Quantity	1
Contingency Philosophy	
Mass	30% for spacecraft subsystems and instruments.
Power	30% for spacecraft and instruments.

5. MISSION ANALYSIS

The mission analysis tasks included generating an initial spacecraft estimate, gathering launch vehicle performance data and recommending one or more vehicles to define the dynamic envelope, estimating the orbit lifetime, determining the amount of propellant required for a controlled deorbit, and investigating ground station contact times for various orbit inclinations. Several tools (e.g., NASA Debris Assessment Software (DAS), Analytical Graphics' Satellite Tool Kit (STK), and in-house tools) were used to complete these assessments.

5.1 Initial Spacecraft Mass Estimate

To determine an initial spacecraft mass estimate, the team looked at historical spacecraft bus data from the Goddard Space Flight Center (GSFC) Rapid Spacecraft Development Office (RSDO) catalog, version 2, known as Rapid II. Rapid II was chosen since the new Rapid III catalog had not yet been released. Plotting the payload mass capability of the buses versus their dry masses resulted in what appeared to be two groups of spacecraft (see fig. 3). The team briefly attempted to determine the reason for the two groups, which consisted of a 'light' group where the payload outweighed the dry bus and a 'heavy' group where the dry bus outweighed the payload. However, after looking at power, propulsion, and other subsystem data, the team could not find any trends that could explain the two apparent groups. Therefore, the mission analysis team decided to use the average of the light and heavy estimates as the basis for the initial dry bus mass estimate. In order to determine the bus mass estimate, the payload (science instrument) mass must be found.

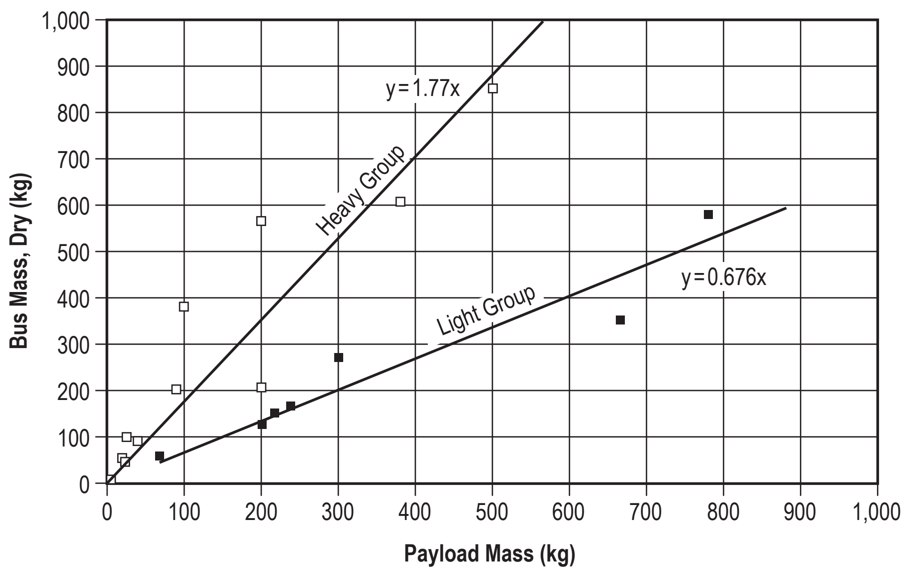

Figure 3. Plot of spacecraft payload mass as a function of dry bus mass from the Rapid II catalog.

The payload consists of two science instrument suites, the LATA and a collection of several SM cameras. The LATA is composed of 60-×60-cm super modules, each with a mass of 30 kg. The SM cameras have a mass of only 2 kg each. Since the science team desired a minimum of 20 LATA super modules, 7 SM cameras, and a 20-kg IDS, the estimated total payload mass was 634 kg (see table 2). Given this mass and the curve fits of figure 3 and adding preliminary propellant capabilities for deorbit, the team arrived at an initial total observatory mass (spacecraft, science instruments, and propellant) of approximately 2,000 kg (see table 3). The final observatory mass resulting from the design study was almost equal to that predicted by the 'heavy' spacecraft bus curve. Nevertheless, the initial estimate was an effective starting point for analysts to begin sizing subsystems.

Table 2. Initial payload mass estimate for AXTAR.

Instrument	Mass Each	Qty	Mass Total
LATA	30	20	600
SM	2	7	14
IDS	20	1	20
		Total instrument mass	634

Table 3. Initial estimate for the spacecraft mass.

Component	Mass (Low Estimate)	Mass (High Estimate)
Payload	634	634
S/C dry mass	450	1,200
Contingency (30%)	325	550
Subtotal dry mass	1,409	2,384
Propellant	90	150
Approximate total mass	1,500	2,500
Split the difference between the low and high estimates ≈2,000 kg		

5.2 Launch Vehicle Selection and Performance

Given the basic dimensions of the LATA super modules (60×60 cm) and the desire to have a minimum of 20 LATA super modules, the team created a preliminary layout of the LATA to compare its dimensions with several launch vehicle shrouds, two of which are shown in figure 4. Since it appeared that the LATA in the Minotaur IV shroud left too little room for the spacecraft (and was a tight fit by itself) and virtually no mass margin, the team decided to baseline the Taurus II as the launch vehicle. Given this decision, the smaller configuration with 20 LATA super modules could be launched on either a Taurus II or Falcon 9 (or larger) vehicle, while the larger configuration with the maximum number of super modules would require a Falcon 9 or larger vehicle.

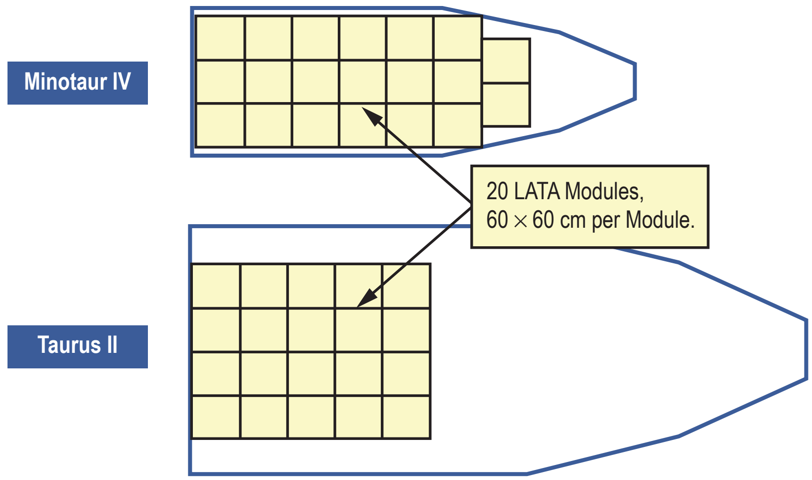

Figure 4. Conceptual layout of the 20 LATA super modules (each 60×60 cm) in the Minotaur IV and Taurus II shrouds.

At the completion of the spacecraft design cycle, this decision was revisited to make sure that the spacecraft mass and volume were still within the limits of the selected launch vehicles. Section 6.1 provides more details.

Given a rough idea of the launch vehicle size needed, the team gathered payload mass capability data for a representative cross section of vehicles as a function of final inclination. A plot of several vehicle performance curves is shown in figure 5. Even though it is desirable to minimize passage through the South Atlantic Anomaly (SAA) by going to as low an inclination as possible, the science team was willing to trade inclination for more LATA super modules. Therefore, performance to various inclinations was needed. The Atlas V 401 performance was included for comparison purposes, and the Taurus II enhanced performance is very preliminary.

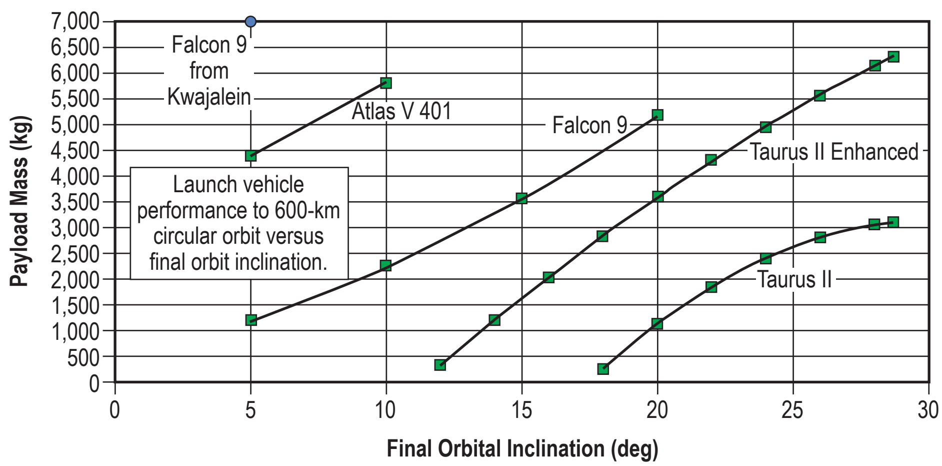

Figure 5. Plot of launch vehicle performance for several vehicles. Final inclination could be traded for spacecraft mass if needed for the AXTAR mission. All launches are from Cape Canaveral Air Force station (CCAFS) unless otherwise noted.

5.3 Orbital Lifetime

The NASA DAS (version 2.0.1) is used to determine the orbital lifetimes at several area-to-mass values and initial circular orbit altitudes. Figure 6 shows the results of the trade analysis. The desire is to find the lowest starting altitude with an orbital lifetime of at least 10 years but no more than 25 years. The AXTAR area-to-mass is expected to vary between approximately 0.007 and 0.011 m^2/kg (bounds the low and high values for both the Taurus II and Falcon 9 configurations). Taking this into consideration, an initial circular orbital altitude of 585 km (preliminary) is selected.

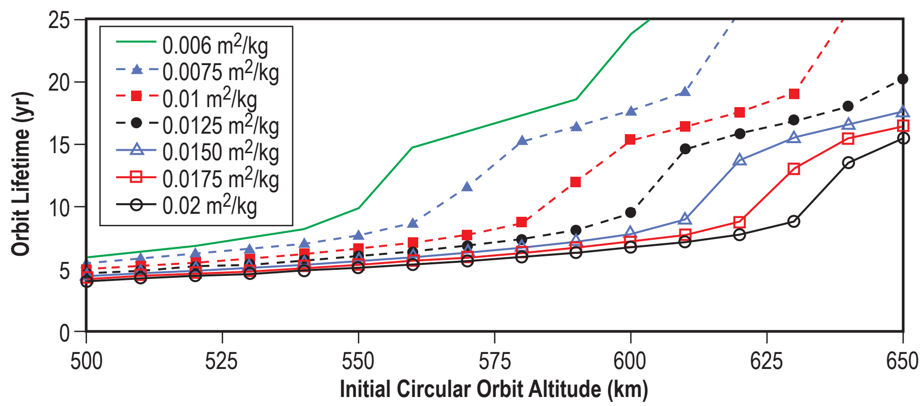

Figure 6. AXTAR orbital lifetime trade.

5.4 End of Life Disposal

The same strategy planned for deorbiting the Compton Gamma Ray Observatory is used for the AXTAR end-of-life disposal.[2] The deorbit is performed over a number of maneuvers that lower the perigee from the initial 585-km circular orbit. The final burn begins with an apogee of 585 km and a perigee of 150 km and targets a controlled reentry with a flight path angle of –1.2° at an altitude of 60 km. The number of maneuvers (five in this case) is chosen such that the ΔVs are roughly equal. The first four burns have an impulsive ΔV of 30.74 m/s and the final ΔV is 37.89 m/s.

Given the impulsive ΔVs above, a finite-burn analysis is performed to determine the total ΔV for each maneuver, including gravity losses, for various thrust levels. The ratio of finite burn to impulsive ΔVs as a function of thrust acceleration for the final deorbit maneuver is shown in figure 7. The first four maneuvers are close in magnitude and start from the same altitude as the final burn; therefore, they are assumed to have the same variation. These results are used to compute the total deorbit propellant requirements at desired thrust levels.

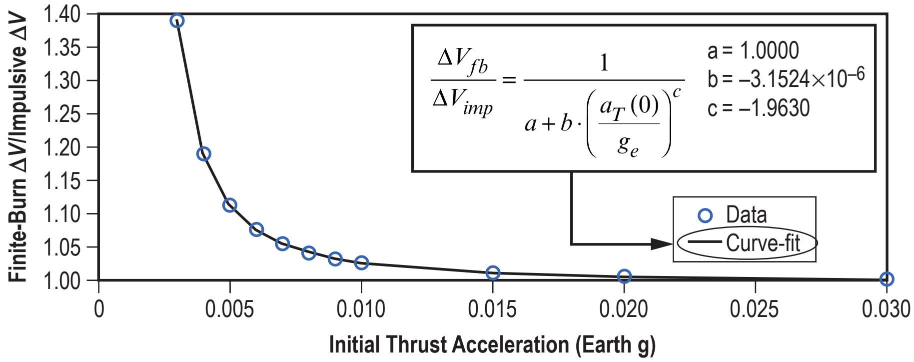

Figure 7. Ratio of finite burn to impulsive ΔVs as a function of thrust acceleration.

Applying the above maneuver strategy and finite-burn results, the total maneuver propellant is computed for the Taurus II and Falcon 9 spacecraft configurations. The Taurus II propellants are summarized in table 4, which gives a total required maneuver propellant of 194.8 kg (includes the main and attitude control amounts). Note that the thrust level drops as the pressure decreases in the tank (from 420 to 110 psia, see section 6.3). The maneuver propellants are computed using the minimum specific impulses of the main and attitude control thrusters (i.e., 223 s and 229 s, respectively). Due to the preliminary nature of this analysis, the attitude control propellant is computed assuming that 5 lbf of thrust operates over the maneuver times. Table 5 gives the results for the Falcon 9 option, which requires 363.48 kg of propellant. Off-the-shelf tanks are used for both spacecraft options. These tanks hold maneuver propellant amounts of 207.94 kg and 405.25 kg for the Taurus II and Falcon 9 configurations, respectively (see sections 6.3 and 7.3).

Table 4. Deorbit propellant requirements for the Taurus II option.

Event	m_0 (kg)	ΔV_ideal (m/s)	MPS Thrust (lbf)	ΔV_real (m/s)	Maneuver Time (s)	Manuever Propellent (kg)
Deorbit Maneuver 1	2,650.30	30.74	257.40	30.77	70.7	37.73
Deorbit Maneuver 2	2,613.26	30.74	216.05	30.79	83.1	37.35
Deorbit Maneuver 3	2,576.73	30.74	174.70	30.82	101.5	37.26
Deorbit Maneuver 4	2,540.67	30.74	133.35	30.88	131.3	36.92
Deorbit Maneuver 5	2,505.05	37.89	92.00	38.24	232.1	45.73

Table 5. Deorbit propellant requirements for the Falcon 9 option.

Event	m_0 (kg)	ΔV_ideal (m/s)	MPS Thrust (lbf)	ΔV_real (m/s)	Manuever Time (s)	Manuever Propellent (kg)
Deorbit Maneuver 1	4894.73	30.74	257.40	30.88	131.1	69.92
Deorbit Maneuver 2	4826.11	30.74	216.05	30.93	154.2	69.31
Deorbit Maneuver 3	4758.33	30.74	174.70	31.03	188.7	68.91
Deorbit Maneuver 4	4691.28	30.74	133.35	31.23	245.2	68.95
Deorbit Maneuver 5	4624.76	37.89	92.00	39.15	438.5	86.39

5.5 Miscellaneous Tasks

In addition to the above analyses, analysts also computed the beta angle history for 5-yr, eclipse durations over a 1-year period, estimated passage durations through the SAA, and ground station contact times for the first 2 months. The baseline launch date for the calculations was June 1, 2017. The 2017 launch date was the initial value used for the study. Although this date was later revised to 2019, there was insufficient time to revise the lifetime analysis. However, minimal impact is expected as the altitude can be raised to negate any shortening of the lifetime due to the solar cycle. The baseline orbits for these assessments were circular (5° and 28.5° inclinations) with an altitude of 585 km. These results were provided as inputs to thermal, power, and COMM experts during the spacecraft design process.

6. SPACECRAFT DESIGN FOR THE TAURUS II LAUNCH VEHICLE

This section details the spacecraft design for the smaller of the two AXTAR configurations with each spacecraft subsystem being described in an appropriate subsection. Carrying the minimum 20 LATA super modules, this design is compatible with the Taurus II payload mass and volume capabilities.

6.1 Configuration

The driving factor in the spacecraft's configuration was the launch vehicle size, which limited the number of the primary science instruments (LATA super modules). The LATA coplanar and pointing requirements drove the design layout to be divided into two main sections: A spacecraft system bus and a science bus (see fig. 8). The science bus/LATA array was set forward to provide viewing separation from the spacecraft's solar arrays and other systems, and the science bus structure was designed to allow the LATA super modules to be nestled inside the primary structure to provide additional shielding. In addition to the primary vehicle structure, a separate Sun-shade secondary structure was placed above the LATA to help meet the Sun avoidance-angle plane requirements. This gave the required shielding height necessary for the rows of LATA super modules.

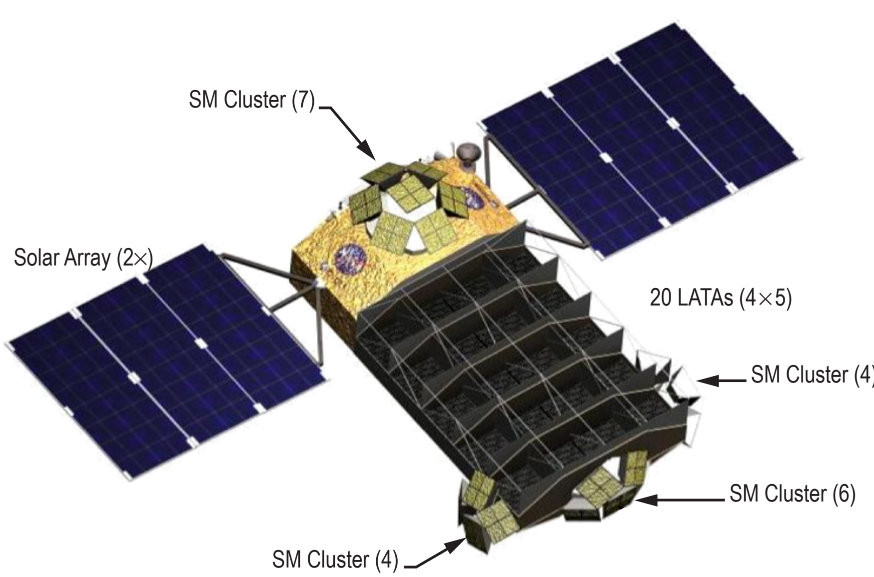

Figure 8. AXTAR configuration for the Taurus II launch vehicle option.

The other design factor was the number and placement of the SM cameras. A 32-face dodecahedron would allow full-sky coverage, but 27 were chosen for the baseline configuration due to volume and operational constraints. These were broken up into several clusters and placed on the spacecraft to optimize view angles while minimizing size. These camera clusters were fairly low mass; therefore, no additional provisions were needed to accommodate them.

The spacecraft system bus is located aft near the launch vehicle interface to carry launch vehicle loads and to allow for the spacecraft's systems to be enclosed in a more compact volume. The central core of the spacecraft bus houses the propulsion tanks and the momentum wheels in a pyramid configuration. The outer bays/walls allow for mounting of the other avionics and spacecraft components. The design has continuous load paths and structures that allow for minimum mass to be obtained. Locating the spacecraft bus aft allows for desirable center of gravity (CG) characteristics. The overall configuration is conservative and does allow room for component growth and for extra subsystem components to be added that were not analyzed in this study. While based on the Taurus II launch vehicle, the design could be used on comparable (not smaller) size launch vehicles. If a larger launch vehicle were selected, a different overall configuration and layout would need to be studied to achieve the optimal configuration.

6.2 Mass Properties

A collaborative engineering environment with discipline subsystem experts was used to size spacecraft subsystems and propellant loads. Two design concepts were analyzed: One to be launched on a Taurus II or similar vehicle and one to be launched on a Falcon 9 or similar vehicle. Mass properties for both Falcon 9 and the Taurus II designs used 30% as the accepted contingency. Science instrument masses also included a 30% contingency for both designs.

The mass breakdown for the overall vehicle, spacecraft subsystems, and science instruments for the Taurus II design is shown in table 6. The Taurus II design resulted in a total vehicle gross mass of 2,690 kg. Gross mass is the total of vehicle dry mass, inert mass, and propellant. Dry mass is defined as spacecraft subsystems mass minus the useable propellant, propellant residuals (see item 7.0 in table 6), and science instruments. The dry mass total, including the 30% contingency, resulted in a mass of 1,567 kg. Inert mass includes propellant residuals and science instruments. The inert mass totaled 916 kg for the Taurus II design. The total mass less propellant (dry mass plus inert mass) resulted in 2,482 kg for the Taurus II design.

Table 6. AXTAR mass summary for the Taurus II configuration.

AXTAR Taurus II Design MEL	Total Mass (kg)
1.0 Structure	770.00
2.0 Propulsion	65.69
3.0 Power	142.04
4.0 Avionics/control	188.53
5.0 Thermal control	38.80
6.0 Contingency	361.52
Dry Mass	**1,566.57**
7.0 Nonpropellant fluids	4.45
8.0 Payload/science instruments	911.20
Inert Mass	**915.65**
Total Less Propellant	**2,482.22**
9.0 Propellant (hydrazine)	207.94
Gross Mass	**2,690.16**

6.3 Propulsion

The propulsion system's primary function is to deorbit the spacecraft at the end of the mission (see sec. 5.4) and to provide attitude control during each maneuver. A simple monopropellant blowdown system, with maximum use of off-the-shelf components, is selected for this task. The propulsion systems for both the Taurus II and Falcon 9 configurations are very similar, the only differences being the selected propellant tanks and the structural support mass. The preliminary system schematic is shown in figure 9. Both propulsion options consist of three diaphragm tanks that are loaded with hydrazine propellant and nitrogen pressurant. The thruster configuration includes four pods, each containing one Aerojet MR-104A/C (100 lbf) engine and two MR-106L (5 lbf) engines. This system provides redundancy with two of the four larger thrusters designated as backup. The chosen engines are a compromise between maintaining low-gravity losses during the deorbit burns and minimizing the engine and feed system sizes.

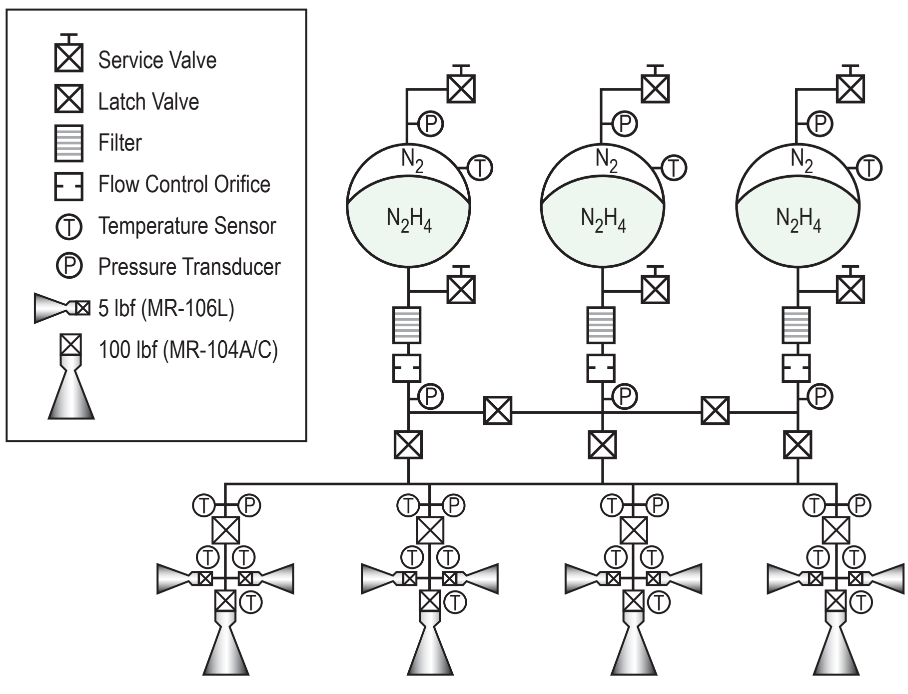

Figure 9. AXTAR propulsion system schematic.

Table 7 lists the mass breakdown for the Taurus II option propulsion system. After applying a conservative 30% growth allowance, the total system dry mass is 85.39 kg. An off-the-shelf tank is selected from Alliant Techsystems, Inc. (ATK) (model number 80488-1); the three tanks hold 210.04 kg of hydrazine and 2.35 kg of gaseous nitrogen. The available maneuver propellant is 207.94 kg, which gives a propellant margin of 6.7% above the required 194.80 kg (see sec. 5.4). The assumed tank pressure range is from 420 psia to 110 psia.

Table 7. Propulsion system mass summary for the Taurus II option.

Qty	Item	Unit Mass (kg)	Total Mass (kg)	Info
4	Axial thrusters	1.86	7.44	Aerojet MR-104A/C
8	Lateral thrusters	0.59	4.72	Aerojet MR-106L
3	Propellant tanks	11.02	33.05	ATK (80488-1)
3	Pressurant fill/drain valve	0.21	0.63	Moog (50-856)
10	Pressure transducers	0.28	2.80	Lunar prospector
19	Temperature sensors	0.10	1.98	FCI (AS-TT)
3	Propellant filters	0.30	0.90	VACCO (F1D10559-01)
3	Flow control orifice	0.02	0.06	AIAA 2003-4470
9	Latch valves	0.50	4.50	Moog (51-134)
3	Propellant fill/drain valve	0.21	0.63	Moog (50-856)
1	Lines and fittings	4.40	4.40	Estimate
1	Structural mounts	4.58	4.58	Estimate
		Total dry mass	65.69	
		Total mass after 30% contingency:	85.39	

6.4 Communications

At present, the downlink data rates for this mission are judged not to be in excess of fixed antenna capabilities. A COMM link to ground using fixed antennas is desirable over an active pointing design due to the elimination of gimbaled mechanisms. This results in a more reliable and lower risk design. Analysis shows that omnidirectional antennas are sufficient for the given data rates and assumed mission scenarios by using an X-band system for the science data downlink. The use of the Tracking and Data Relay Satellite System (TDRSS) for normal operations and data link is not desirable from a cost perspective. However, a TDRSS link during launch and startup operations is desirable. The COMM system designs are single fault tolerant.

For this mission, low-orbit inclinations are better for science data collection. A ground link analysis based on link times and daily accesses was performed to determine the best selection of ground stations at both 5° and 28°. South Point, Hawaii and Kourou, Guiana were selected as the primary and secondary ground link stations respectively for 28° orbits out of 11 possible ground stations analyzed. These same two stations are also selected at a 5° orbit, where Kourou becomes the primary station and South Point becomes the secondary station.

At these two stations, at least an 8-min primary link is possible seven times a day. Under a worst-case condition, a 6.7-min link five times a day is possible for the 5° inclination at the secondary South Point station. Using these two conditions as bounds, a link budget was performed assuming at least four links per day for 8 min each. For these link times, a 20-W transmitter can achieve a 90-Mbps data downlink rate. At this rate, almost 44 Gbits can be transmitted per station pass, or 172 Gbits per day. Assuming a continuous average data rate of 700 kbps for the entire LATA system and 72 kbps for an eight-unit SM cluster, enough link capability is left to download over six 15-min peak (19-Mbps) LATA events per day. Major SM events can be alternated with LATA event data, assuming the SM detects events first and then the LATA is pointed to the event. The suggested 20-W L3 X-band transmitter is presently at Technical Readiness Level (TRL) 6.

An estimate for the total spacecraft telemetry COMM data rate using S-band is 60-kbps downlink and 4-kbps uplink. Using a 5-W S-band transmitter, a link to the TDRSS for launch and startup operations can be accomplished at these data rates using omniantennas. The 5-W transmitter can also link with ground for normal telemetry with plenty of margin available. The suggested AeroAstro 5-W S-band transmitter is at TRL 8.

6.5 Avionics, Guidance, Navigation, and Control

The mass summary of the avionics system components is listed in table 8. Two fully redundant Proton 200 flight computers from SpaceMicro are the core of the avionics system. The computers are used for spacecraft operations and data management. They receive processed science data from the IDS and transfer the data to either the onboard data recorders or the X-band transmitters for downloading. The two data recorders from Surry Satellite Technology can store up to 256 Gbits of data at a rate of 150 Mbps. One day of required data storage is estimated to be about 173 Gbits, providing about 50% in memory margin. The flight computers are radiation hardened to a 100-krad total ionizing dose and a 70 MeV-cm^2/mg single-event latch up. The Proton 200, currently at TRL 6, is scheduled to launch during 2011 on board the Operationally Responsive Space Satellite 1 (ORS Sat-1). The Surrey data recorders are at TRL 8.

Table 8. Avionics mass results for the Taurus II configuration.

Avionics System Components	Total Mass (kg)	Comments
Attitude control system	95.38	Includes reaction wheels and torque rods.
Command and data systems	14.00	Includes computers and recorders.
Instrumentation	15.00	Includes sensors and cabling.
Communications systems	30.15	Includes X-band and S-band.
Avionics cabling	34.00	Power cabling not included.
Total dry mass	188.53	
Total mass after 30% contigency:	245.09	

Attitude knowledge is achieved using a redundant pair of Ball Aerospace star trackers and Northrop Grumman inertial measurement units (IMUs). The star trackers provide 4 arc-s of accuracy, meeting the 5 arc-s mission requirement. Both the IMUs and the star trackers are at TRL 8 or above.

While this satellite has large surface areas resulting in significant disturbance torques, the slewing and pointing requirements are modest. Off-the-shelf reaction wheels should be sufficient for attitude control, keeping the cost down. In low-Earth orbit, magnetic torque rods are good candidates for attitude control assist and desaturation of reaction wheels. By using magnetic torquers, the spacecraft reaction control system (RCS) will not normally be required for attitude control and is considered for contingency purposes and disposal only. A set of three dual-coil Microcosm MT400-2 magnetic torquers is suggested for this mission. These torquers are at or above TRL 8.

The trade space for reaction wheels included four different wheel types at four slewing speeds. This trade resulted in a Teldix RSI 68-170 reaction wheel being selected. These wheels are mounted in a four-wheel pyramid configuration for best performance while maintaining one-fault tolerance. They have sufficient momentum and torque capability to exceed the fast slew requirement of 180° in 30 min while providing the desired pointing accuracy of 1 arc-min. The Teldix wheels are at or above TRL 8.

Inertial pointing will be the typical pointing mode for science observations. In this mode, the magnetic torque rods provide good torque authority. Using the torquers to offset the disturbances, the 28-hour continuous pointing goal can be achieved with unlimited pointing times possible. The disturbances will usually be small in a zenith-pointing mode, which is a possible scanning mode. In this pointing mode, desaturation of the reaction wheels can be accomplished using the magnetic torque rods. Atmospheric and gravity torques are at a continuous maximum in a worst-case torque scenario. The reaction wheels will eventually saturate if left in this mode for extended periods; however, extended pointing times in this mode are unlikely since there are no apparent science advantages. If wheel saturation does occur, a return to zenith pointing for quick desaturation is possible.

6.6 Power

PowerPower required for the AXTAR spacecraft subsystems and science instrument payload is listed in table 9.

Table 9. Power budget for the AXTAR spacecraft.

Subsystem / Element	Power (W)
Attitude control system	152.4
Command, data, instrumentation	125.2
Communication power	179.2
Large area timing array	600.0
SM	121.0
Instrument data system	40.0
30% power margin	365.3
Total	1,583.1

The power system was designed to meet the 5-yr mission duration assumption and operate in a circular orbit of at least 500-km altitude (resulting in a maximum eclipse period of 43.9 min, including lunar eclipse and a minimum light period of 58.8 min). Regarding the environment, a solar power density of 1,370 W/m^2, solar panel operating temperature of 76 °C, a thermal sink temperature of 279 K, and a maximum ambient electronics temperature of 30 °C was assumed. The secondary batteries were selected based on a minimum of 25,000 charge/discharge cycles, a 40% maximum depth of discharge, a maximum operating temperature of 30 °C, and a packing factor of 1.35 (i.e., battery mass is 1.35 times the total mass of all the cells). The solar arrays were sized assuming a 5% knockdown for cell mismatch and interconnection, a 3% per year degradation (due to UV radiation, thermal cycling, and contamination), and 90% cell coverage for each panel. The assumed bus voltage was 28 V, with a charge efficiency and discharge efficiency of 90%.

Because of the relatively long science mission duration and the high levels of sunlight available, a solar-based power system was chosen for this spacecraft. Figure 10 illustrates the basic elements of this design. The power system mass summary is listed in table 10.

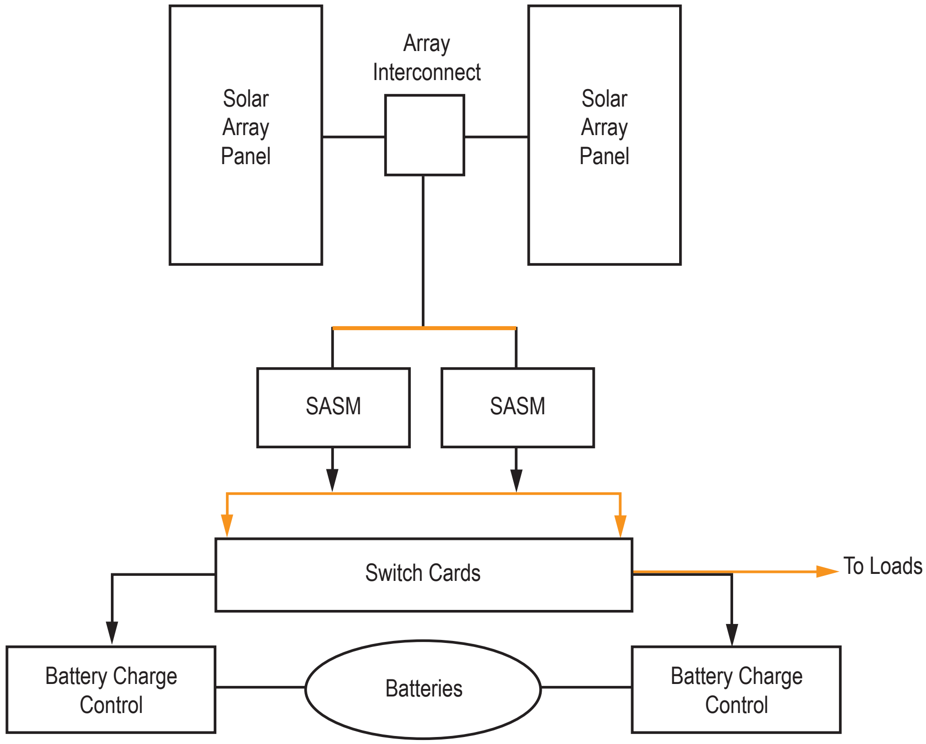

Figure 10. Solar-based power system.

Table 10. Power system mass summary.

Qty	Power System Components	Unit Mass (kg)	Total Mass (kg)	Comments
2	Rigid solar array wing	16.60	33.20	8.27 m^2 area
2	Solar array structure and yoke	17.20	34.40	Includes articulation mechanism.
2	Solar array switch module	1.14	2.28	Orion/Mars recon orbiter
4	Distribution switch	1.14	4.56	Orion
2	Pyro controller	1.10	2.20	Orion
1	Umbilical controller	1.14	1.14	Orion
2	Battery charge/discharge controller	1.15	2.30	Mars recon orbiter
1	Power system electronics enclosure	16.70	16.70	Scaled from COTS AITECH.
1	Cabling and harness	8.00	8.00	Estimate
3	Secondary battery	12.42	37.26	Sized from Saft VL 48E cells.
		Total dry mass	142.04	
		Total mass after 30% contingency	184.65	

The required power generation capacity of the solar arrays is the sum of the overall power requirement as detailed in table 9. The overall required power amounts to 3,708 W, which consists of the power required to charge the secondary batteries that supply this power requirement while the craft is in the eclipse (dark) portion of its orbit and the losses incurred in the power system. In order to minimize risk, a standard, rigid panel solar array that uses spring tension to unfold during deployment was chosen. The total area of the arrays, including hinges and mechanisms, is 16.6 m^2

($8.3\ m^2$ per wing) with an areal density of $2.16\ kg/m^2$ and a specific power (end of life) of $241.6\ W/m^2$. The cells are GaAs triple junction—Spectrolab ITJ or equivalent—with a 24% efficiency at 76 °C. The arrays are constructed with a ZnO front contact and a Cu back contact on M55J Al-honeycomb substrate.

The power electronics is comprised of a set of 11 VME circuit boards caged in a rugged VME enclosure that is scaled from an existing space-qualified design. Instead of specifying a particular manufacturer's board for each function, the system is sized from representative boards using typical masses, power requirements, and specifications from previous missions. Each of the two solar array switchboards (sequential shunt) handles both array wings for full fault tolerance. Two pyrotechnic (pyro) event controllers manage pyro events, each handling up to eight events. For power distribution, each of the 4 distribution switches can control up to 16 circuits. The two battery charge controllers can charge two secondary batteries each, and the single umbilical switch can switch up to eight external high-current circuits. The VME enclosure is sized with fully redundant power supplies.

Because the spacecraft and science instrumentation must be powered during those periods of the orbit in which the solar arrays get no sunlight, energy must be stored during the lighted periods. Li-ion secondary batteries were chosen for this purpose because of their favorable energy density and high reliability. The battery cells are Saft Li-ion VL48E cells (3.6 V, 48 Ah). Each battery contains cell-balancing electronics and consists of eight cells for a total charge capacity of 28.8 V. The total number of battery units is three, each having a mass of 12.42 kg. The maximum depth of discharge is 40%.

6.7 Structures

Structural requirements of the AXTAR spacecraft bus are driven by four main factors: (1) Launch platform limitations; (2) primary science instruments' shape, size, and mass; (3) the requirement to mount 20 LATA super modules in a coplanar array; and (4) radiation shielding. The Taurus II 3.45-m diameter launch platform limits LATA array width to four super modules, necessitating a 4×5 LATA array. Radiation shielding against cosmic and solar radiation is incorporated into the spacecraft bus. Since the LATA array is quite massive and requires heavy metal cosmic radiation shielding on three sides, the Taurus II CG height limit of 2 m from the payload interface plane limits the height of the spacecraft bus. The LATA also requires solar radiation shielding for viewing angles up to 45° from the Sun, which is installed in parallel planes perpendicular to the five rows of the LATA array. The resulting AXTAR-Taurus spacecraft configuration meets the Taurus II payload static envelope, CG, and payload mass-to-orbit estimate for the mission profile, as well as the science instruments' performance requirements with positive structural margin of safety. See figure 8 for configuration features and details.

The spacecraft concept uses lightweight 2024–T351 Al panels, tubing struts, and T- and I-beam structural supports for component and science instrument mounting. The panels conduct heat and double as radiators for thermal management. Midway fore and aft on opposite sides of the spacecraft bus, two flight-proven telescoping booms support the fold-out solar arrays and are stowed against the spacecraft bus for launch. These were sized similarly to those used on the Hubble Space Telescope but proportionally less massive.[3]

A short, octagonal main spacecraft bus provides the needed structure and surfaces to secure propulsion, avionics, and other spacecraft components on two interior panels of a four parallel-panel structure. The top and bottom exterior panels provide mounting for antennas and other sensors, as well as 13 of the 27 SMs. The aft bulkhead panels are used for thermal management and support the Taurus II launch adapter ring. The four parallel panels and two perpendicular walls that support and separate them are one-eighth- to one-quarter-in (0.32–0.64-cm) thick to support all the mounted equipment and enhance thermal management. Most exterior surfaces are much thinner and are primarily thermal closeouts, rather than structural.

The science bus structure needed to support the LATA array and 14 SMs is forward of the main spacecraft bus. Graded-Z galactic radiation shielding utilizing tin/tantalum/Al is modeled as nonstructural mass in the science bus structure, conceptually similar to the shielding flown on the Swift Burst Alert Telescope (BAT).[4] The availability of tin (foil) may not support the fabrication of this shielding. Copper replaced tin for the Swift BAT fabrication. This change should be negligible to structural mass estimates in this study. A lightweight solar shade truss structure of 0.04–0.1 gauge (0.09–0.25-cm thick) round tubing struts bears six lead/Al Sun shades, tall enough to shade each super module.

A Finite Element Modeling Analysis and Postprocessing (FEMAP) model was created based on the Pro/E configuration, loaded with components and instrument masses, and analyzed with NX NASTRAN. Eighteen load cases were run with a load set of combined Taurus II/Falcon 9 launch vehicle loads, +6/–2 g (acceleration due to gravity at standard sea level) axial and ±2 g lateral, at 30° and 45° intervals. A 1.4-safety factor for isotopic strength and a 0.65-buckling factor were used in the analysis. Once the structural analysis was successful, the model was optimized using Hypersizer® for minimum mass. Structural members were grouped, results were analyzed, and material thicknesses were adjusted to reflect common raw material stock and other manufacturability criteria. The results were iterated through FEMAP until a minimum mass, positive margin of safety (MOS), and total deformation of less than 3 cm was converged upon using a compressive yield stress of 40,000 psi (2.76×10^8 Pa) for Al. The FEMAP model used to analyze and size the spacecraft is depicted in figure 11. Plates, bars, and nonstructural and rigid-mass elements are used to represent all masses estimated by all the disciplines developing the spacecraft.

Table 11 summarizes the structural mass of the spacecraft bus, science bus, shielding,and secondary structure. The AXTAR-Taurus concept structure has a high MOS (i.e., low stresses) because some panels were thickened to limit excessive bending displacements and maximize thermal conduction and radiation to space. Total mass after 30% contingency is 1,001 kg.

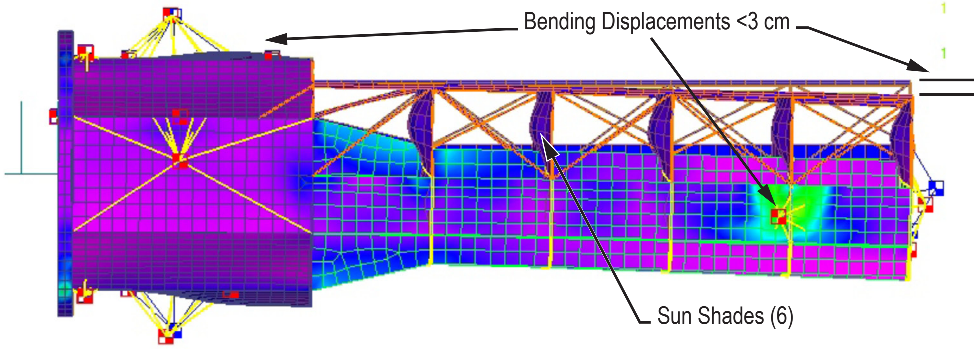

Figure 11. FEMAP Taurus II model deformed side view showing rows of Sun shades and exaggerated deformations.

Table 11. AXTAR structures mass summary for the Taurus II configuration.

Structural Components	Mass (kg)	Comments
Spacecraft bus	215.5	Aft end housing spacecraft systems and launch adapter.
Science/LATA bus	164.0	Forward end of spacecraft supporting 20 LATAs.
Galactic radiation shielding	306.0	Nonstructural mass shielding three sides of all LATA super modules.
Solar shade supports	30.5	Nonstructural components supporting solar shielding.
Solar radiation shielding	25.0	Nonstructural mass shielding all LATA super modules in the spacecraft X-Z plane.
Secondary structure	29.0	SM mounting structures.
Total dry mass	770.0	
Total mass after 30% contingency	1,001.0	

6.8 Thermal

A passive-thermal design concept was developed for the AXTAR spacecraft, passive meaning that no actively pumped fluid systems are required. Thermal control of the AXTAR spacecraft will use components including multilayer insulation, high emissivity paint and coatings, heaters, etc. to maintain spacecraft subsystem components within acceptable temperature ranges. There are no dedicated radiators; spacecraft structural panels act to dissipate avionics heat by conduction and also act as radiative surfaces. The bus outer surfaces are covered in low absorptivity materials in order to cold bias the spacecraft and minimize temperature fluctuations due to orbital position. Propellant tanks are wrapped in multilayer insulation (MLI) and propellant tank heaters are sized for worst-case cold orientation. RCS thrusters, antennas, solar arrays, and solar array mechanisms are not part of the prephase A analysis.

A system-level thermal model of the spacecraft bus and LATA structure was developed using Cullimore and Ring Technologies' Thermal Desktop® (<http://www.crtech.com>) to assess spacecraft bus and LATA support structure interface temperatures. The model geometry is shown in figure 12 and represents the Taurus II configuration. The structure is modeled as Al, the panel thickness being consistent with the structural design. Environmental heat loads were calculated for an Earth orbit altitude of 585 km. Steady-state spacecraft structure temperatures for both a hot and cold orientation were generated in order to determine a range of temperatures to be expected during the on-orbit operations. Full subsystem and experiment heat loads were applied. The hot orientation is defined as LATA to a Sun angle of 30° with a beta angle of 50°, and the cold orientation is defined as LATA to a Sun angle of 90° with a beta angle of 0°. Both hot and cold orientations are shown in figure 13.

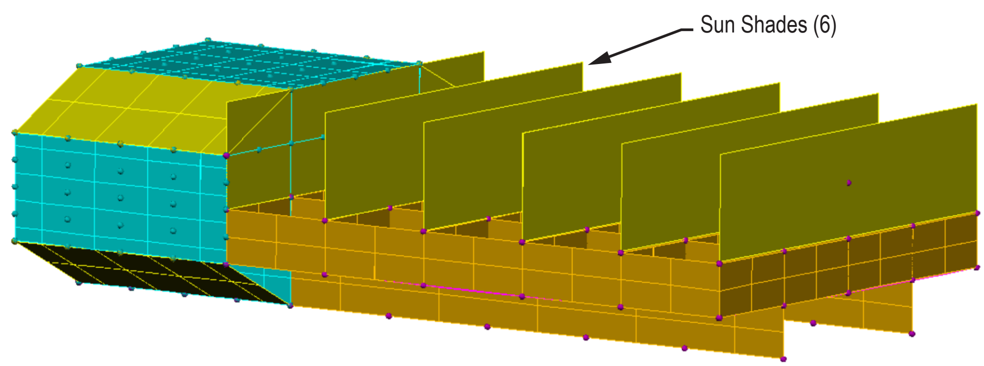

Figure 12. Thermal model geometry for the Taurus II configuration.

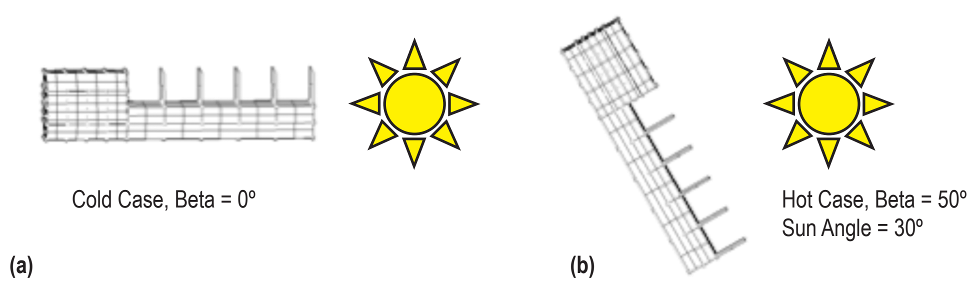

Figure 13. Orientation of spacecraft showing the (a) cold case and (b) hot case used in the thermal analysis.

The spacecraft bus internal surfaces are assumed to be black anodized to optimize radiative exchange within the enclosure. White paint is used on the exterior surfaces to provide a low absorptivity (α) to emissitivity (ε) ratio (α/ε) that serves to minimize structure temperatures. The backside of the LATA support structure is covered with a 12-layer MLI blanket, with the outer layer being beta cloth painted white. LATA supports and Sun shades are also painted white. Optical properties used for the thermal model surfaces are itemized in table 12 and were taken from various sources including the Spacecraft Thermal Control Handbook[5] and NASA Spacecraft Thermal Coatings Reference.[6] MLI ε^* values represent effective emissivity for the blanket.

Table 12. Thermal model surface optical properties.

	Material	Absorptivity	Emissivity
Spacecraft bus internal surfaces	Black anodized	0.90	0.90
Spacecraft bus external surfaces	White paint	0.25	0.87
Spacecraft bus closeouts	White paint on beta cloth	0.17	0.92
	Inner layer, black kapton® (MLI=5 layers) ε*= 0.02	0.92	0.88
LATA support structure	White paint on beta cloth (12 layers) ε*= 0.004	0.17	0.92
LATA supports	White paint	0.25	0.87
LATA sun shades	White paint	0.25	0.87
RCS tanks	MLI (5 Layers, AlK), ε*= 0.02	0.60	0.09

Subsystems equipment and experiment heat loads assumed for the thermal analysis are shown in table 13 for the Taurus II configuration. A total of 1,566 W of power/heat dissipation was considered in the thermal analysis. All heat loads are imposed directly on the structure and modeled as area averaged heat loads. Heat loads were distributed on the spacecraft panels according to specific box locations. Temperature predictions reflect structure interface temperatures and are documented in table 13. Experiment temperature prediction is not included as part of the subsystem thermal analysis. Each of the three propellant tanks will require a maximum of 3-W heater power, as sized by the cold orientation. All predicted interface temperatures are within the acceptable range, as shown in table 13.

Table 13. Heat dissipation, operating, and predicted temperatures for the Taurus II configuration.

	Total Heat Dissipation (W)	Operating Temperature Range (°C)	Predicted Interface Temperature Range (°C)
ACS control system	43.4	–40 to 85	–10 to 15
Reaction wheels (4)	97.6	–20 to 60	–10 to –7
Magnetic torquers (3)	11.4	–43 to 66	–10 to –3
Command and data system	125.2	–40 to 85	–10 to 15
COMM system	179.1	–20 to 70	–10 to 15
Power systems enclosure	167.0	–40 to 85	–10 to 23
Batteries (3)	185.4	0 to 45	7 to 23
RCS tank heaters (3)	9.0	10 to 30	19 to 25
IDS	40.0	–40 to 5	–20 to –11
LATA (20)	600.0	–40 to 10	–8 to 0
SM (27) + telemetry hub	117.0	–40 to 10	–30 to 3
Total	1,566.1*	–	–

* This total does not include RCS tank heaters since this is an intermittent requirement.

Steady state analysis results for the spacecraft bus and LATA support structure cold case are shown in figures 14 and 15, respectively. Structure temperatures range from 25 °C to 35 °C for the bus and –8 °C to –67 °C for the LATA structure. The steady state analysis results for the spacecraft bus and LATA support structure hot case are shown in figures 16 and 17, respectively. Structure temperatures range from –15 °C to 23 °C for the bus and 4 °C to –32 °C for the LATA structure. The spacecraft bus and LATA support structure were analyzed as a single unit, although temperature results are shown separately in figures 14 through 17 for clarity. Environmental heat loads are averaged about an orbital period. All predicted interface temperatures are within acceptable operating ranges for as-defined hot and cold orientations.

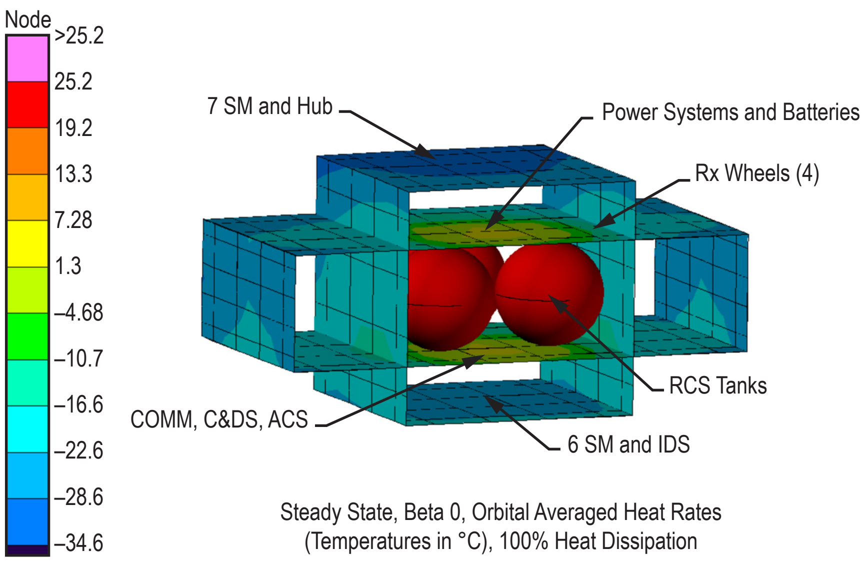

Figure 14. Analysis results for the Taurus II configuration: Spacecraft bus (cold case).

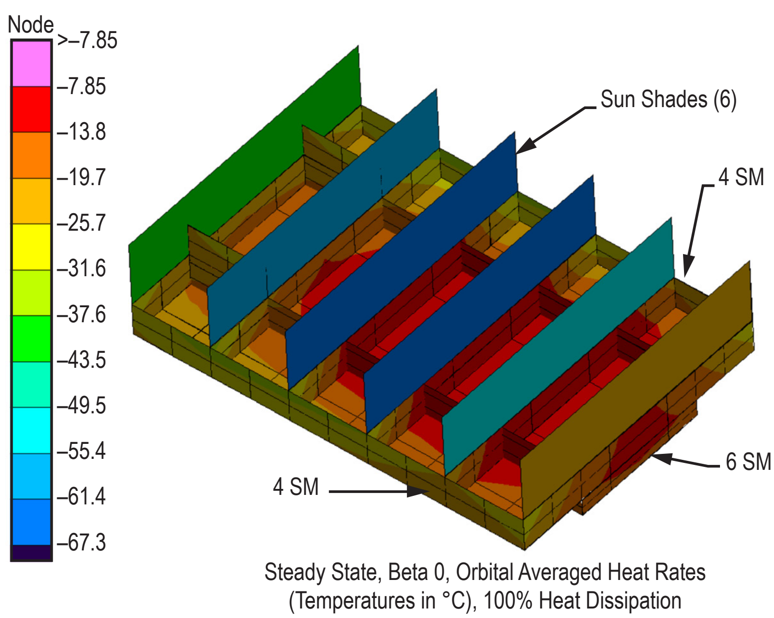

Figure 15. Analysis results for the Taurus II configuration: LATA structure (cold case).

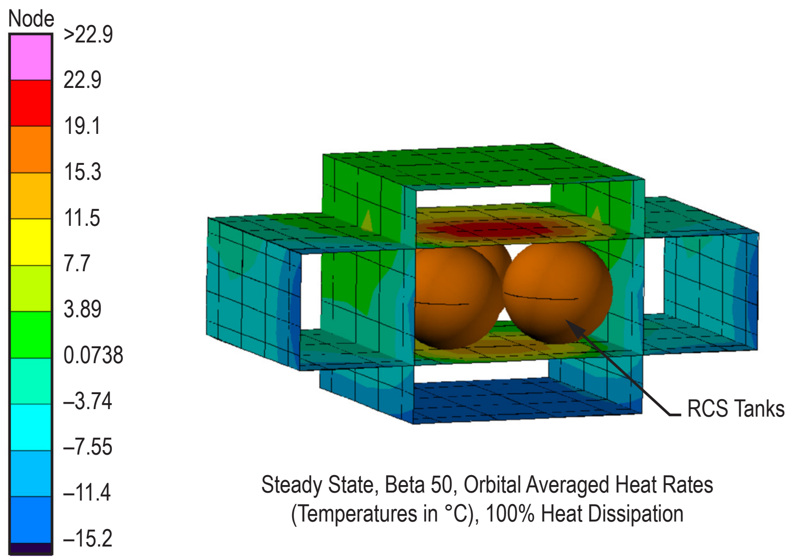

Figure 16. Analysis results for the Taurus II configuration: Spacecraft bus (hot case).

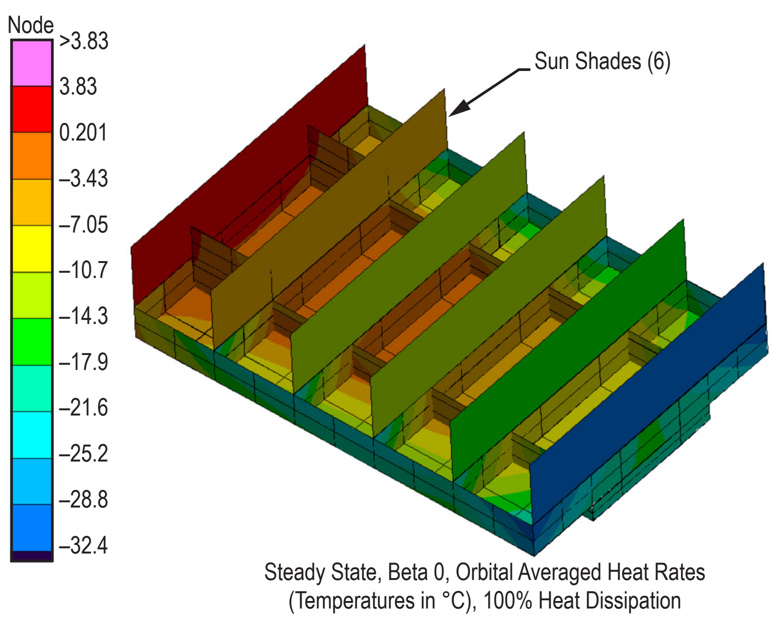

Figure 17. Analysis results for the Taurus II configuration: LATA structure (hot case).

Thermal management is accomplished with typical, flight proven components, so no technology development is required. Total thermal control mass for the Taurus II configuration is estimated at 50.4 kg, which includes a 30% margin as shown in table 14.

Table 14. Thermal subsystem mass summary for the Taurus II configuration.

Thermal System Components	Total Mass (kg)	Comments
Multilayer insulation/thermal tape	31.0	10–12 Layer blanket
Thermal filler	1.2	Chotherm
Paint/thermal coatings	6.1	White paint
Heaters/thermostats	0.5	–
Total dry mass	38.8	
Total mass after 30% contingency	50.4	

7. PRELIMINARY SPACECRAFT DESIGN FOR THE FALCON 9 LAUNCH VEHICLE

The science team was interested in knowing the maximum number of LATA super modules that could be accommodated on a spacecraft designed to use the higher performance of a larger launch vehicle such as the Falcon 9. Therefore, the spacecraft design team created an alternative configuration for that vehicle. While time constraints during the study prevented the team from iterating until the design closed, the mass budget and subsystem designs are representative of what would have been the final design had the study continued.

7.1 Configuration

The driving factor in the spacecraft's configuration was the number of the primary science instruments (LATA super modules) desired. The Falcon 9 shroud size would allow for more super modules to be carried than were needed to accomplish the science goals. The LATA's coplanar and pointing requirements drove the design layout to be divided into two main sections, a spacecraft system bus and a science bus (see fig. 18). The science bus/LATA array was set forward to provide viewing separation from the spacecraft's solar arrays and other systems. The science bus structure was designed to allow the LATA super modules to be nestled inside the primary structure to provide additional shielding. With a relatively large shroud diameter, it was possible to place six super modules in the spacecraft bus section for a more integrated compact vehicle. To help meet the Sun avoidance-angle plane requirements, a separate Sun-shade secondary structure was placed above the LATA in addition to the primary vehicle structure. This gave the required shielding height necessary for the rows of LATA super modules.

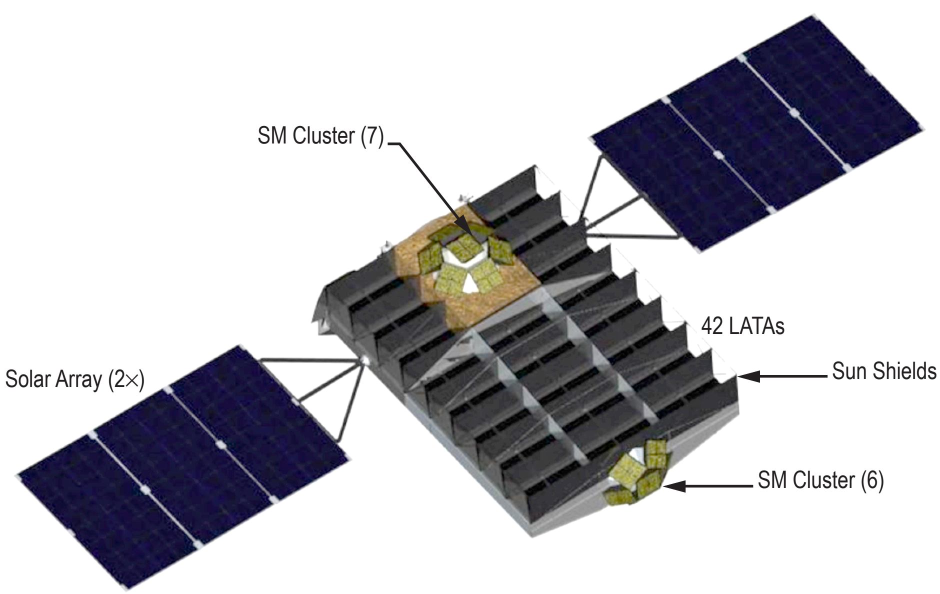

Figure 18. Configuration of the AXTAR spacecraft: Falcon 9 option.

Another design factor was the number and placement of the SM cameras. A 32-face dodecahedron would allow full-sky coverage. These were broken up into several clusters and placed on the spacecraft to optimize view angles while minimizing size. These camera clusters were fairly low mass, so no additional provisions were needed to accommodate them. Final placement and grouping would need to be optimized in a further study.

The spacecraft system bus is located aft near the launch vehicle interface to carry launch vehicle loads and to allow for the spacecraft's systems to be enclosed in a more compact volume. The central core of the spacecraft bus houses the propulsion tanks and the control moment gyros (CMGs). Discussion of the use of CMGs is in section 7.4. The outer bays/walls allow for mounting of the other avionics and spacecraft components. The design has continuous load paths and structures, allowing for minimum mass to be obtained. Locating the spacecraft bus aft also allows for desirable CG characteristics. The overall configuration is conservative and does allow room for component growth and for extra subsystem components to be added that were not analyzed in this study. The design based on the Falcon 9 launch vehicle could be used on comparable (not smaller) size launch vehicles.

7.2 Mass Properties

The mass breakdown for the overall vehicle, spacecraft subsystems, and science instruments for the Falcon 9 design is presented in table 15. The Falcon 9 design resulted in a total vehicle gross mass of 4,972 kg. Gross mass is the combined total of vehicle dry mass, inert mass, and propellant. Dry mass is defined as spacecraft subsystems mass minus the useable propellant, propellant residuals (see item 7.0 in table 15), and science instruments. The dry mass total, including the 30% contingency, resulted in a mass of 2,766 kg for the Falcon 9 design. Inert mass includes propellant residuals and science instruments. The inert mass for the Falcon 9 design totaled 1,801 kg. The total less propellant (dry mass plus inert mass) resulted in 4,567 kg for the Falcon 9 design.

Table 15. AXTAR mass summary for the Falcon 9 configuration.

AXTAR Falcon 9 Design MEL	**Total Mass (kg)**
1.0 Structure	1,330.00
2.0 Propulsion	94.66
3.0 Power	226.31
4.0 Avionics/control	422.53
5.0 Thermal control	53.90
6.0 Contingency	638.22
Dry Mass	**2,765.61**
7.0 Nonpropellant fluids	4.09
8.0 Payload/science instruments	1,797.20
Inert Mass	**1,801.29**
Total Less Propellant	**4,566.90**
9.0 Propellant (hydrazine)	405.25
Gross Mass	**4,972.15**

7.3 Propulsion

The mass breakdown for the Falcon 9 option propulsion system is listed in table 16. This system is identical to the one for the Taurus II configuration except for the propellant tanks and support structure (see sec. 6.3). The total predicted dry mass for this option is 123.05 kg. An ATK tank (model 80325-1) is also chosen for this configuration. The totals for loaded propellant and pressurant are 409.34 kg and 4.82 kg, respectively. The available maneuver propellant is 405.25 kg (11.5% margin). The assumed tank pressure range is from 420 psia to 110 psia.

Table 16. Propulsion system mass summary for the Falcon 9 option.

Qty	Item	Unit Mass (kg)	Total Mass (kg)	Info
4	Axial thrusters	1.86	7.44	Aerojet MR-104A/C
8	Lateral thrusters	0.59	4.72	Aerojet MR-106L
3	Propellant tanks	20.00	60.00	ATK (80325-1)
3	Pressurant fill/drain valve	0.21	0.63	Moog (50-856)
10	Pressure transducers	0.28	2.80	Lunar Prospector
19	Temperature sensors	0.10	1.98	FCI (AS-TT)
3	Propellant filters	0.30	0.90	VACCO (F1D10559-01)
3	Flow-control orifice	0.02	0.06	AIAA 2003-4470
9	Latch valves	0.50	4.50	Moog (51-134)
3	Propellant fill/drain valve	0.21	0.63	Moog (50-856)
1	Lines and fittings	4.40	4.40	Estimate
1	Structural mounts	6.60	6.60	Estimate
		Total dry mass	94.66	
		Total mass after 30% contingency:	123.05	

7.4 Communications, Avionics, Guidance, Navigation, and Control

For the larger Falcon-9 design, slewing is much more challenging, and CMGs are considered. Using CMGs in place of reaction wheels to accommodate the higher moments of inertia and disturbance torques provides the required performance with margin. Although the mass of CMGs is moderately higher than reaction wheels, the power demand is greatly reduced.

Using a pointing dish antenna to accommodate the higher data rates for an additional 22 LATA modules (basically doubling the data rate) greatly reduces the power requirement for transmission. However, this adds a pointing mechanism to the design. This pointing mechanism is estimated to be at TRL 8. For the additional memory requirement and torque authority needed, the number of data recorders and magnetic torquers were doubled. This resulted in a relatively small increase in mass and power. The mass summary is listed in table 17.

Table 17. COMM, avionics, and GN&C mass summary for the Falcon 9 configuration.

Avionics System Components	Total Mass (kg)	Comments
Attitude control system	308.98	Includes reaction wheels and torque rods
Command and data systems	20.00	Includes computers and recorders
Instrumentation	15.00	Includes sensors and cabling
Communications systems	38.55	Includes X-band and S-band
Avionics cabling	40.00	Power cabling not included
Total dry mass	422.53	
Total mass after 30% contingency	549.29	

7.5 Power

The extended configuration for the Falcon 9 had power requirements as shown in table 18.

Table 18. Power budget for AXTAR for the Falcon 9 configuration.

Subsystem / Element	Power (W)
Attitude control system	174.2
Command, data, instrumentation	178.8
Communication power	120.3
Large area timing array	1,260.0
SM	129.0
Instrument data system	40.0
30% power margin	570.7
Total	2,473.0

Using the same power system design, the team resized the components for the new power requirement. Table 19 summarizes the result.

Table 19. Power system mass summary for the Falcon 9 configuration.

Qty	Power System Components	Unit Mass (kg)	Total Mass (kg)	Comments
2	Rigid solar array wing	24.96	49.92	11.6 m^2 area
2	Solar array structure and yoke	25.89	51.78	Includes articulation mechanism
3	Solar array switch module	1.14	3.42	Orion/Mars recon orbiter
4	Distribution switch	1.14	4.56	Orion
2	Pyro controller	1.10	2.20	Orion
1	Umbilical controller	1.14	1.14	Orion
3	Battery charge/discharge ctl	1.15	3.45	Mars recon orbiter
1	Power system electronics enclosure	18.32	18.32	Scaled from COTS AITECH
1	Cabling and harness	17.00	17.00	Estimate
6	Secondary battery	12.42	74.52	Sized from SAFT VL 48E cells
	Total dry mass		226.31	
	Total mass after 30% contingency		294.20	

7.6 Structures

The Falcon 9 larger, 4.6-m diameter launch platform can launch a much larger spacecraft with greater science capability. Over twice as many LATA super modules were accommodated while maintaining a low CG. Spacecraft structural mass was scaled from the AXTAR-Taurus configuration (which also meets all Falcon 9 launch platform requirements) to accommodate the greater mass of 42 LATA super modules and other spacecraft components. An estimated 70% increase in structural mass is necessary to accommodate the larger and more capable AXTAR-Falcon concept. The FEMAP model used to analyze and size the spacecraft is depicted in figure 19.

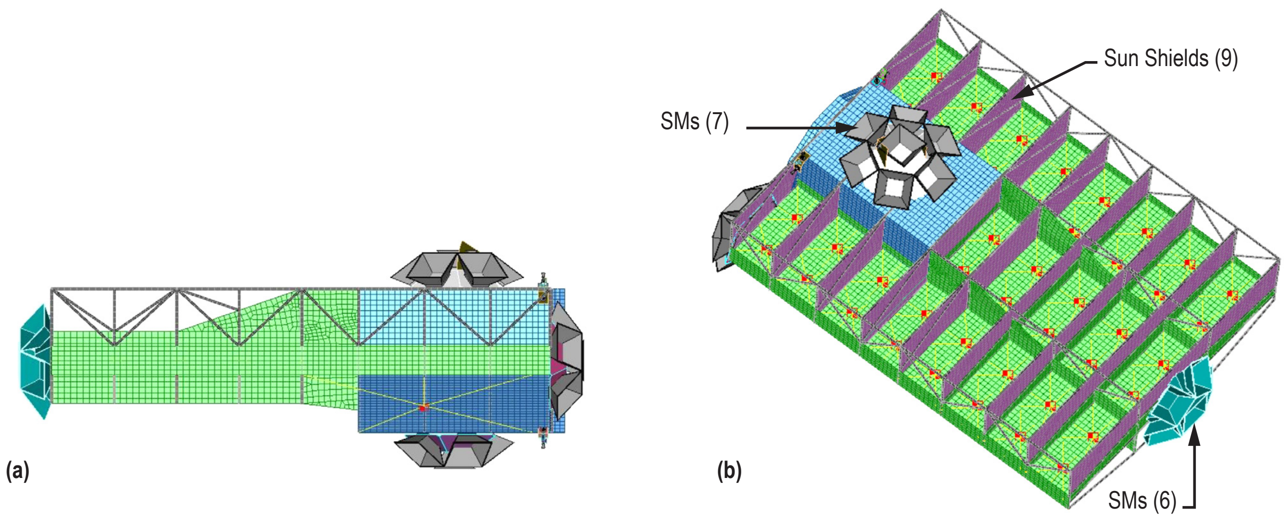

Figure 19. FEMAP model used for the Falcon 9: (a) Side view and (b) top view.

Table 20 summarizes the structural mass of the AXTAR-Falcon concept structure. Total mass after 30% contingency is 1,729 kg.

Table 20. Structural mass summary for the AXTAR spacecraft for the Falcon 9 configuration.

Structural Components	Mass (kg)	Comments
Spacecraft bus	323	Aft end housing spacecraft systems, launch adapter, and 12 LATAs
Science/LATA bus	246	Forward end of spacecraft supporting 30 LATAs
Galactic radiation shielding	612	Nonstructural mass shielding three sides of all LATA super modules
Solar shade supports	65	Nonstructural components supporting solar shielding
Solar radiation shielding	55	Nonstructural mass shielding all LATA super modules in the spacecraft X-Z plane
Secondary structure	29	SM mounting structures
Total dry mass	1,330	
Total mass after 30% contingency	1,729	

7.7 Thermal

Thermal control for the Falcon 9 configuration is consistent with the smaller Taurus II design. Mass estimates were determined by appropriate scaling to account for a larger bus and support structure and are presented in table 21. The total mass after a 30% contingency is estimated at 70.1 kg. Table 22 details the heat dissipation and operating temperatures for the subsystems and experiment components. Although no analyses were performed for this configuration, it is expected that all structural temperatures will be acceptable. Battery location adjustments may be required to accommodate the increased heat dissipation. A total of 2,362 W of spacecraft power/heat dissipation is estimated for this configuration.

Table 21. Thermal system mass summary for the AXTAR spacecraft for the Falcon 9 configuration.

Thermal System Components	Total Mass (kg)	Comments
Multilayer insulation/thermal tape	42.0	10–12 Layer blanket
Thermal filler	2.1	Chotherm
Paint/thermal coatings	9.1	White paint
Heaters/thermostats	0.7	–
Total dry mass	53.9	
Total mass after 30% contingency	70.1	

Table 22. AXTAR heat dissipation and operating temperatures for the Falcon 9.

	Total Heat Dissipation (W)	Operating Temp Range (°C)
ACS control system	43.4	–40 to 85
Reaction wheels (4)	108.0	–20 to 60
Magnetic torquers (6)	22.8	–43 to 66
Command and data system	178.8	–40 to 85
COMM system	120.3	–20 to 70
Power systems enclosure	189.0	–40 to 85
Batteries (6)	279.0	0 to 45
RCS tank heaters (3)	12.0	10 to 30
IDS interface to bus	40.0	–40 to 5
LATA (42) interface to structure	1,260.0	–40 to 10
SM (28) + telemetry hub interface	121.0	–40 to 10
Total	2,362.0*	–

* This total does not include the RCS tank heaters since this is an intermittent requirement.

8. RISK ANALYSIS

In identifying risks for both the spacecraft and the science instrumentation, a taxonomic approach was chosen. The risks were identified from an analysis of the uncertainties discovered in the following areas:

- Requirements: How certain and well defined are the science objectives and the requirements derived from them?
- Design: How certain are the design parameters and design decision consequences?
- Testing: How defined and complete are the plans for testing?
- Programmatic considerations: How certain are the funding program's guidelines, limitations, and mission requirements and the team's ability to satisfy them?

The individual risks were identified through interviews with each of the study participants. Each risk was then evaluated in terms of its likelihood (as a probability), its impact (in dollars), and possible options for management (mitigation, acceptance, transfer, etc.) at the next study level. There were a number of risks identified and they may be summarized as follows:

- Mission requirements are not well defined. No formal requirements analysis has been performed. Performing a formal requirements analysis will fully mitigate those risks.
- Electronics associated with the x-ray detectors have not been designed and power consumption is far lower than any previously implemented functional equivalent. There is no reason to believe that the power consumption requirement can be met; however, a risk impact analysis performed to determine the impact of higher power consumption on the spacecraft showed that the impact is minimal (a 50% higher consumption will result in about 30 kg of power system mass). Therefore, relaxing the requirement is suggested.
- Currently, there are no test plans for any of the science instrumentation, and no known similar test plans are available for adaptation. Consequently, the facilities available for testing along with the cost and schedule for testing is unknown. A formal test planning exercise will fully mitigate this risk.
- There are materials availability issues with high-purity Si and tin. These may be mitigated programmatically by advanced planning.

- Ultimate costs of both target launch vehicles are unknown, and there may not be an adequate vehicle available at launch date in the Medium-Class Explorer (MIDEX) cost range. This risk is common to most MIDEX missions and must be managed as testing of these vehicles progresses. A lighter version of AXTAR capable of fitting on a Minotaur IV would provide a mitigating contingency.

9. ANIMATION

The purpose of creating the AXTAR animation was to introduce the spacecraft design and its components to space science viewers. The animation focused primarily on the spacecraft instruments and LATA super module functionality. Textual labeling was used to highlight the instruments with technical information pertinent to their target. Additionally, the labeling served as basic narration for viewers.

The LATA super module sequence was depicted in greater detail than other instruments. An exploded view of this science instrument showed x rays represented by glowing particles being selected by the collimator and striking the Si pixel detector. The underlying interposer board was shown creating a charge. The sequence continued as the interposer board charges were converted into digital data and sent to the computer. This animated sequence is a comprehensive representation of LATA component assembly and functionality.

The basic processes used for this animation were modeling, texturing, lighting, animating, rendering, and compositing. The compositing process was somewhat robust for this animation. Each instrument and component was rendered on individual layers. While this process is more time consuming, it provides animators with more control of final appearances. Multiple layers of instruments, backgrounds, and text were composited together using Autodesk Combustion 2008. The entire AXTAR animation contains over 30 layers of elements.

The use of professional animation software and processes resulted in a complete AXTAR animation. The video can be shown at conferences and used in presentations.

10. CONCLUSIONS

The spacecraft design team completed two spacecraft concepts for the AXTAR mission—one with a minimal set of science instruments and one with a much larger set. While the design of the smaller configuration was driven to closure, the team had the time and resources to complete just one design iteration of the larger configuration. Since the main focus of the study was to complete the design of the smaller configuration and the larger configuration was a secondary objective, all study goals were met.

A summary showing the major elements of each spacecraft configuration is presented in table 23. The smaller configuration, which contains 20 LATA super modules and 27 SM cameras, can be launched on either the Taurus II (and Taurus II enhanced) or the Falcon 9 or similar vehicle. Launching on the Falcon 9 provides a lower final inclination, helping to minimize the SAA effect. The larger configuration, with 42 LATA super modules, is much more massive and requires a Falcon 9 or similar vehicle. For either configuration, reducing the mass will result in a lower final inclination. Also of note is that the spacecraft subsystems do not require the development of any new technologies; all subsystem components are either existing or are based on existing technology.

Table 23. Summary data for the two spacecraft designs.

	Smaller (Taurus II) Configuration	Larger (Falcon 9) Configuration*
LATA super modules (qty)	20	42
SMs (qty)	27	27–32
Total observatory mass (kg)	2,700	5,000 (est)
Final orbit inclination with Taurus II / Taurus II Enhanced, launched from CCAFS (deg)	25/18**	na/na
Approximate final orbit inclination with Falcon 9 launched from CCAFS / Kwajalein (deg)	12/5	19/5
Spacecraft technologies needing development	None	None

* These are preliminary results; design is not closed.
** Taurus II enhanced performance is very preliminary.

Additional design work could uncover potential mass savings in the design, as both spacecraft designs fell on the 'heavy' curve of the relationship between payload mass and dry bus mass (see fig. 3). The primary benefit of lower mass would be a lower final orbital inclination and result in reducing the SAA effect. If somehow the number of LATA super modules could be reduced and still satisfy the science mission goals, the spacecraft could possibly be launched on a smaller vehicle, such as the Minotaur IV.

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R.C. Hopkins, L. Johnson, H.D. Thomas, C.A. Wilson-Hodge,
M. Baysinger*, C.D. Maples*, P.D. Capizzo**, L.L. Fabisinski***,
L. Hornsby†, K.S. Thompson†, and J.H. Miernik‡

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14. ABSTRACT
The Advanced X-Ray Timing Array (AXTAR) is a mission concept for submillisecond timing of bright galactic x-ray sources. The two science instruments are the Large Area Timing Array (LATA) (a collimated instrument with 2–50-keV coverage and over 3 square meters of effective area) and a Sky Monitor (SM), which acts as a trigger for pointed observations of x-ray transients. The spacecraft conceptual design team developed two spacecraft concepts that will enable the AXTAR mission: A minimal configuration to be launched on a Taurus II and a larger configuration to be launched on a Falcon 9 or similar vehicle.

15. SUBJECT TERMS
black hole, neutron star, x-ray timing, spacecraft design

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National Aeronautics and
Space Administration
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George C. Marshall Space Flight Center
Huntsville, Alabama 35812