Design for an 8 meter monolithic UV/OIR space telescope

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ABSTRACT

ATLAST-8 is an 8-meter monolithic UV/optical/NIR space observatory to be placed in orbit at Sun-Earth L2 by NASA's planned Ares V cargo launch vehicle. The ATLAST-8 will yield fundamental astronomical breakthroughs. The mission concept utilizes two enabling technologies: planned Ares-V launch vehicle (scheduled for 2019) and autonomous rendezvous and docking (AR&D). The unprecedented Ares-V payload and mass capacity enables the use of a massive, monolithic, thin-meniscus primary mirror - similar to a VLT or Subaru. Furthermore, it enables simple robust design rules to mitigate cost, schedule and performance risk. AR&D enables on-orbit servicing, extending mission life and enhancing science return.

Keywords: Large Space Telescopes, UV/Optical Space Telescopes, Ares V Launch Vehicle, Astronomy

1. INTRODUCTION

NASA's planned Ares V launch vehicle with its 10 meter fairing shroud and 55,600 kg capacity to the Sun Earth L2 point is a disruptive capability which enables an entirely new generation of 21st Century Space Observatories building on the Great Observatories legacy [1]. These observatories can be 8-meter class monolithic mirror telescopes, 16-meter JWST class telescopes or even 25-meter class segmented telescopes. An 8-meter class monolithic ultra-violet/optical space telescope, with 5 times the angular resolution and 10 times the collecting aperture as Hubble, offers the opportunity to answer some of the most compelling science questions. How did the present Universe come into existence and of what is it made? What are the fundamental components that govern the formation of today's galaxies? How does the Solar System work? What are the conditions for planet formation and the emergence of life? And, maybe most importantly, are we alone? When the scientific history of the 21st century is written, detection of life on other planets will certainly be one of the most significant.

Postman et. al., have defined four research areas for an 8-meter class space astrophysics observatory: 1) detecting habitability and bio-signatures on terrestrial mass exo-solar planets; 2) reconstructing the detailed history of the assembly of stellar mass; 3) establishing the mass function and characterizing the accretion environment of super massive black holes; and 4) precisely determining the growth of structure in the universe by kinematic mapping galactic dark matter halos [2]. Regarding the search for life on exoplanets, an 8-meter class telescope will be able to observe the habitable zone of 280 total stars and 101 Solar class stars. Of these, it is estimated that 5 transiting exoplanets can be spectroscopically characterized over a 5 year period. Further, it is calculated than an 8 meter telescope could make a broadband image (R=4) in 1 hour and an R=100 (SNR=10) spectrum in 160 hours of an Earth twin at 20 pc [2]. Regarding star formation histories in the local universe, an 8-meter class telescope will be able to measure accurate colors and luminosities of stars in over 100 galaxies, of which over ten are giant spirals and one is a giant elliptical [2]. Regarding the relationship between galaxy formation and super massive black holes, an 8-meter class telescope will be able to obtain spatially resolved images and spectra of the disks of accreting gas around 1740 super massive black holes out to a redshift of 0.36 [2]. Regarding studying the gravitational instability paradigm as the driving physical mechanism behind the formation of galaxies and large structures, an 8-meter class telescope will be enable direct measurements of key properties (both for dark and visible matter) of structures and map their evolution in time [2]. The above science studies can be achieved with an 8-meter Hubble class telescope, i.e. diffraction limited at 500 nm with spectral coverage from 110 nm to 2.5 micrometers.

This paper reports on a recent design study conducted at Marshall Space Flight Center which shows that it is possible to package a 6 to 8 meter class monolithic observatory into a 10 meter Ares V fairing (Figure 1); have it survive launch; and place it in to a halo orbit about the Sun-Earth L2 point. Specific technical areas studied include optical design; structural design/analysis including primary mirror support structure, sun shade and secondary mirror support structure; thermal analysis; spacecraft including structure, propulsion, GN&C, avionics, power systems and reaction wheels; mass and power budgets; and system cost. Additionally, the study baseline architecture assumes servicing via autonomous rendezvous and docking to replace the spacecraft and science instruments as required - yielding an observatory operational lifetime of 20 to 30 years.

2. THE ARES-V LAUNCH CAPABILITY ENABLES NEW DESIGN CONCEPTS

The study started with the unique capabilities of the Ares V vehicle (Figure 2) and then considered how those capabilities might enable entirely new mission architectures.

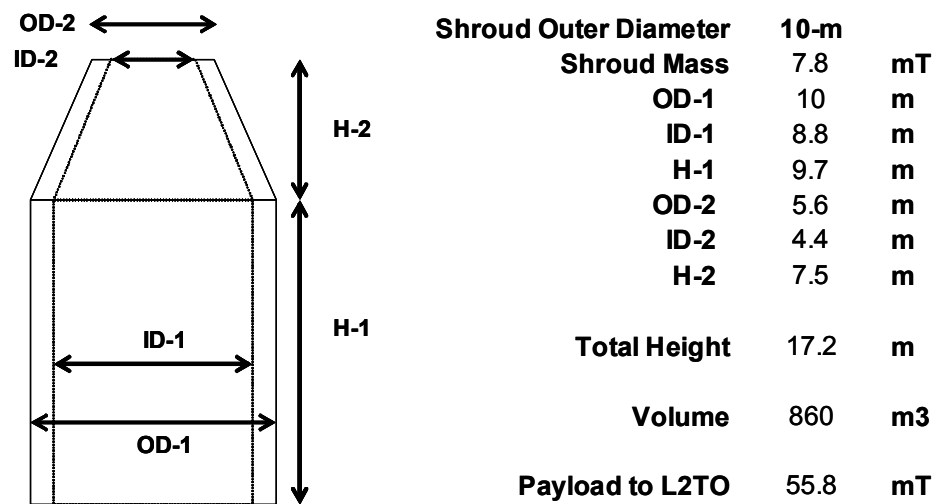|		
Shroud Outer Diameter	10-m	
Shroud Mass	7.8	mT
OD-1	10	m
ID-1	8.8	m
H-1	9.7	m
OD-2	5.6	m
ID-2	4.4	m
H-2	7.5	m
Total Height	17.2	m
Volume	860	m3
Payload to L2TO	55.8	mT

Figure 2 Ares V Baseline Shroud Dimensions and Payload Mass Capability. (Please note that the Ares V is an evolving vehicle and these are preliminary values and may not match the latest Ares V shroud dimensions and weights.)

First, the baseline 10 meter fairing has an 8.8 meter internal dynamic envelope diameter. This is sufficient to accommodate an 8-meter class monolithic circular primary mirror without the need for segmentation. A monolithic mirror provides superior science return because, as compared to a segmented mirror, it has a more uniform, symmetric and stable Point Spread Function. And, it avoids the risk of deployment and complex alignment and phasing control. The 10 meter shroud also allows an 8-meter monolithic mirror to be launched in a face up configuration which provides the most benign vibration and acoustic exposure. Looking further into the future, the 10 meter fairing also allows for even larger aperture segmented designs. Concepts are under consideration for 16 to 24 meter segmented telescopes.

Second, the payload mass of 55,800 kg to an L2 Transfer Orbit enables an entirely new paradigm – design simplicity. Given the available extra mass, designers can use more mature technologies and higher design rule safety factors to eliminate complexity, to lower cost and to lower risk. By using higher design margins it is possible to minimize the marching army size which also reduces the management burden – every $100M in component cost savings reduces total program cost from $300M to $500M. The cost savings of eliminating mass constraint is difficult to quantify, but anecdotal evidence suggests that early in a mass constrained mission, it may cost $100K of design effort to eliminate 1 kg of mass and that once the design is mature, it can cost as much as $1M to eliminate 1 kg of mass.

3. OBSERVATORY DESIGN

3.1 Design Concept

Figure 3 shows the MSFC design concept for an 8-meter monolithic primary mirror ultraviolet/optical space observatory packaged inside the Ares V 10-m fairing's dynamic envelope. The concept has three main subsystems: telescope, support structure and spacecraft. The telescope consists of an 8-meter primary mirror, secondary mirror and forward structure/baffle tube. The spacecraft provides all normal spacecraft functions (such as propulsion; guidance, navigation and control; communication; etc.) and houses the science instruments. The support structure supports the primary mirror. And, it carries the observatory mass (of the primary mirror, telescope forward structure and spacecraft) providing the interface of this mass to the Ares V for launch.
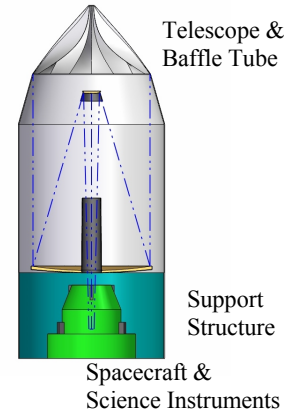

Telescope & Baffle Tube

Support Structure

Spacecraft & Science Instruments

Figure 3 MSFC 8-meter observatory concept in Ares V dynamic envelope

3.2 Optical Design

The feasibility study considered two different telescope optical systems. An F/15 Ritchey-Chretién (RC) design (Figure 4a) was examined for its excellent on- and off-axis image quality, compact size, and ultra-violet throughput. Also RC designs are the optical system mainly used by today's large telescopes. Therefore, it might be possible to reuse existing scientific instrument designs. Unfortunately, this optical design has only a relatively narrow 1-arc minute field of view (NFOV) that is diffraction limited at 500 nm. One method to achieve the desired wide field performance is to use a refractive corrector - although with a limited spectral range – in the scientific instrument suite. Another approach to achieving multi-spectral wide field performance is to use a three mirror anastigmatic telescope with fine steering mirror design (Figure 4b). This configuration has a wide 100 arc-minute (8.4 by 12 arc minutes) field of view (WFOV) that is diffraction limit at 500 nm. But, it also has lower ultra-violet throughput because of its two additional reflections. A potential solution is to implement a dual pupil configuration where UV and NFOV instruments operate at the Cassegrain focus and WFOV instruments operate off-axis providing their own tertiary mirror. The study assumed that the optical coatings will be the same aluminum with MgF overcoat used on Hubble to provide good spectral transmission from 120 nm to beyond 1 micrometers.

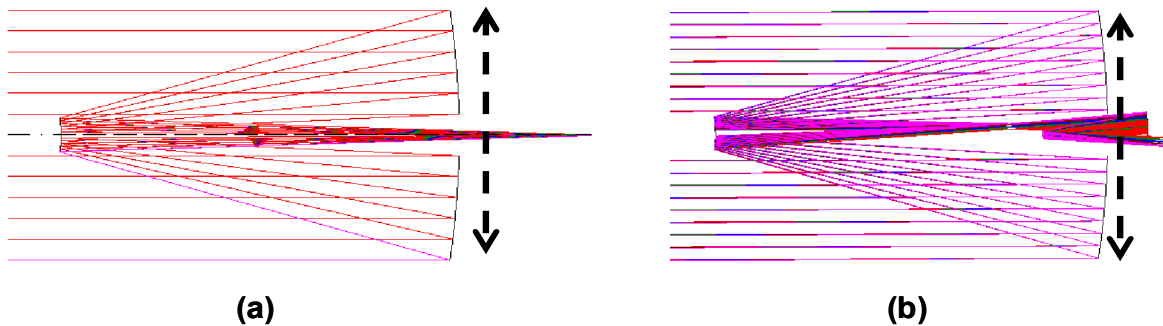**(a)**   **(b)**

Figure 4 Telescope Optical Path. The original optical configuration (a) was used to size the telescope subsystems and develop the mass budget. The revised optical configuration (b) results in an 80X larger field of view

3.3 Primary Mirror

For either telescope design, the monolithic mirror will be manufactured using existing ground based mirror technology. This approach has two specific advantages: technical maturity and cost risk. First, it has been demonstrated that one can actually polish an 8 meter class ground based telescope mirror to a surface figure of better than 8 nm rms (Geyl, 1999) (which is close to the desired 5 nm rms surface figure for the 8 meter Terrestrial Planet Finder program). This is important because as shown in Table 1, while Hubble's 2.4 meter 180 kg/m2 mirror was polished to 6.4 nm rms, the AMSD program only achieved 20 nm rms on its 1.4 meter segment 18 kg/m2 mirror. The higher the mirror's areal density (or in actuality its specific stiffness), the easier it is to achieve a very good surface figure. Second, the cost for an 8 meter ground mirror is $20M to $40M or $0.4M/m2 to $0.8M/m2 while the cost of a 50 square meter space technology mirror will be $200 to $500M ($4M to $10M/m2). While this architectural choice adds approx 20,000 kg to the mass of the payload, the estimated $200M to $500M savings in mirror hardware costs translates into total program cost savings

of from \$700M to \$2B (engineering design, system integration & test, management and fees/program reserves add to the total cost of any program by a factor of 2.5X to 3X of the hardware costs).

Table 1. Comparison of Space and Ground Mirrors						
Parameter	HST	Spitzer	AMSD	JWST	Ground	
Material	ULE	Beryllium	ULE & Be	Be	Various Glass	
Diameter	2.4	0.85	1.4	1.5 (6.5)	8.2	m
Area	4.5	0.5	~1	25	50	m2
Temperature	300	4	300/30	30	300	K
Surface Figure	6.4	75	20/77	25	7.5 to 15	nm rms
Areal Density	180	28	18	26	300 to 500	kg/m2
Areal Cost	10	10	4	6	0.5	\$M/m2
Year	1984	1999	2005	2008	Various	

The reason for both advantages is that ground based mirrors are very massive and hence very stiff. Thus, they are much easier to fabricate than space mirrors. Historically, space mirrors are very low mass (Table 1) and thus not very stiff. They have large gravity sags and are difficult to handle, mount and fixture. And, they are difficult to fabricate to very high precision. Thus, they are expensive.

Three ground based mirror technologies have been considered. The spare VLT (Very Large Telescope) mirror manufactured by Schott is an 8.2 meter diameter, 200 mm thick, Zerodur solid thin meniscus blank with a mass of 23,000 kg. If edged to 6.2 meters diameter and 175 mm thick, it would have a mass of 11,000 kg. The University of Arizona manufactures 6 to 8 meter class borosilicate mirrors using their honeycomb spin cast technique. Recently, Arizona manufacture two mirrors for the LBT (Large Binocular Telescope) that are 8.4 meter diameter, 900 mm thick and with a mass of 16,000 kg. Finally, the Subaru Telescope ULE thin meniscus 8.3 meter primary mirror (blank manufactured by Corning and finished by Brashear) had a mass of 21,000 kg.

3.4 Structural Design

A fundamental question of the design study was whether an 8-meter class ground based telescope mirror could even survive launch. The Ares V launch environment was analyzed by the MSFC Advanced Concepts Office using POST3D. The maximum launch loads (Table 2) are similar to those for existing launch vehicles. Please note, these loads are not concurrent. They are the maximum load experienced at some time during launch. And, the lateral loads do not include wind loading or vibration.

Table 2 Maximum Launch Loads of an Ares V via POST3D Analysis	
Maximum Launch Load	Ares V
Axial (Z)	4 g's
Lateral (Y)	7 x 10-6 g's
Lateral (Z) (Z is down range direction)	6 x 10-4 g's

A structural analysis determined that 66 axial support points keep the stress level on an 8.2 meter diameter 175 mm thick meniscus primary mirror below 1000 psi (Figure 5). Thus, the mirror will survive launch.
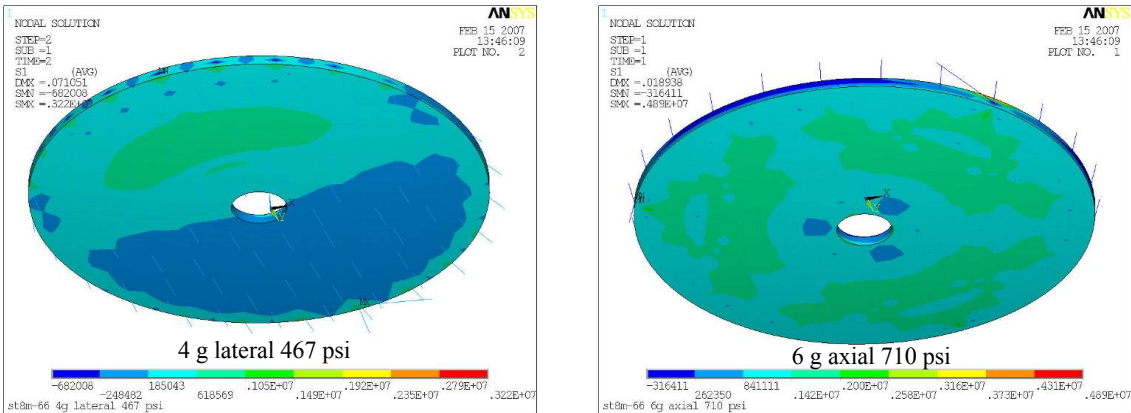

Figure 5 An 8.2 meter 175 mm thick mirror can survive launch loads. 66 axial supports keep bulk stress below 1000 psi.

The observatory structure is divided between the forward and back structure. The forward structure is similar to that of the Hubble Space Telescope. It provides the metering structure between the primary and secondary mirrors and holds the straylight baffle tube. Because of fairing length limitations, the forward structure is split into an upper and lower part. The lower structure is load carrying. It holds the secondary mirror assembly tripod structure. The upper part contains the upper baffle tube and the cover doors. The upper part slides forward on orbit to the full length of the straylight baffle. The cover doors open and close on-orbit as required. A secondary tripod structure extending from the primary mirror was considered but determined to be unable to achieve the desired system stiffness levels for an ultra stable telescope.

The back structure has multiple functions. First, it supports the primary mirror with 66 axial supports. Second, the forward structure is attached to the back structure as is the spacecraft. A key design element of the MSFC concept is that all observatory mass is carried through the back support structure to an interface ring which attaches via other supports to the Ares V launch vehicle. This design concept allows the use of a completely conventional spacecraft, i.e. it does not need extra mass because it does not provide the interface between the observatory and the launch vehicle.
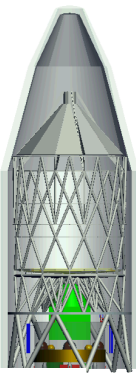

Figure 6 Observatory Support Structure

Structural design and analysis was performed for the spacecraft using standard NASA guidelines. No technical problems were identified. The primary product of this effort was a mass budget for the spacecraft

3.5 Thermal Design

Standard thermal design and analysis was performed for 4 different solar angles: 0, 45, 90 and 120 degrees where 0 degrees is the observatory back facing the sun and 90 degrees is the observatory broadside to the sun. It was modeled that the science instruments produce 750 W of heat and the avionic systems produce another 850 W of heat. The analysis assumed that the observatory is wrapped with five 10 layer MLI blankets and that the spacecraft has 16.0 m2 of thermal radiators. Thermal gradients were calculated for both the spacecraft and the 8 meter primary mirror. (Figure 7)

Without an active thermal management system, the primary mirror temperature varies as a function of sun angle from 160 K to 300K with approximately a 1K variation at each temperature.

Table 3 Primary Mirror Temperature	
Sun Angle	Temperature
0 deg	200K
45 deg	190K
90 deg	160K
120 deg	300K

Therefore, an active thermal management via 14 heat pipes is required to hold the primary mirror temperature at a constant 300K for all sun angles with less than 1K of thermal gradient. On-going thermal analysis will determine exactly how small of a thermal gradient can be achieved. This is important because long exposure observations (such as extra-solar terrestrial planet finding and characterization) require a very stable observatory wavefront. And, the primary mirror surface figure varies as a function of temperature based on the substrate material coefficient of thermal expansion

Sun = 0° 45°

90° 120°

Figure 7 Primary mirror temperature (°C) vs sun angle

(CTE) value and uniformity. At 300K, Corning specifies ULE glass to have a mean CTE value of 0 +/- 30 ppb/K from 5C to 35C and Schott specifies that Zerodur ceramic glass has a CTE value of 0 +/- 50 ppb/K. Additionally, Schott specifies that the CTE uniformity distribution within a Zerodur mirror blank is +/- 10 ppb/K.
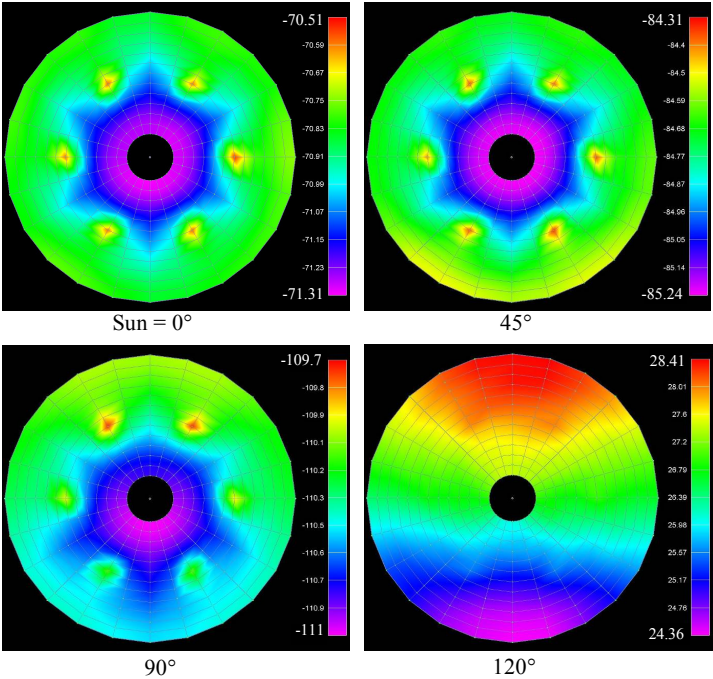
3.6 Spacecraft

The observatory actually has two separate spacecraft: a telescope bus which is part of the optical telescope element (OTE), and a replaceable spacecraft/instrument bus (SIB) (Figure 8). The SIB houses science instruments and subsystems to communicate with and control the telescope. Each spacecraft produces its own power. The telescope has 18 m2 of body mounted solar arrays around the light tube. The SIB has 9 m2 of deployable solar array wings with pointing ability. The SIB power system includes 800W for primary mirror thermal control and 750W for science instruments. The OTE performs its own on-board health diagnostics and communication to the SIB. The SIB provides the primary communication down-link.
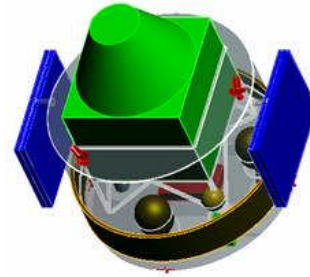

Figure 8 Spacecraft/Instrument Bus (SIB).
Top contains science instruments,
Solar panels are on side.

The spacecraft propulsion system is sized to get the observatory from roughly a 185 x 300,000 km parking orbit (energy, or C3, of -2.60 km^2/s^2) into a halo orbit about the Sun-Earth L2 point and perform all station keeping operations. The spacecraft has a dual mode hydrazine-NTP bi-prop / hydrazine mono-prop propulsion system with 5 years of propellant and redundant thrusters. The propellant load is based on an estimated station keeping ΔV expenditure for five years of 20 m/s, plus the ΔV required to place the telescope on to the L2 transfer trajectory. Propulsion during the trip from the parking orbit to L2 is provided by hydrazine-BTP bi-prop 125 lbf thrusters (Northrop). Station keeping at L2 is provided by hydrazine mono-prop RCS 20/5 lbf thrusters (Aerojet). The telescope has an independent control system with mono-propellant hydrazine using 350/100 psi blowdown Aerojet thrusters. The telescope propulsion system has 30 kg of propellant for 30 year mission.

Guidance Navigation and Point Control is provided by the spacecraft reaction wheels. A trade study was performed to obtain the optimum science performance as a function of wheel torque and momentum storage specifications. (Figure 9)

Two performance parameters were analyzed. The number of hours that the telescope can stare at a fixed point in space (remain at an inertial hold) before needing to perform a momentum dump due to solar radiation pressure torque. And, how fast in minutes the telescope can perform a 60 degree slew. The analysis was done for a sun angle of 90 degrees, which is the worst condition for solar radiation pressure torque. At any other sun angle, the available science time increases. And it was assumed that momentum buildup occurs in only one axis (y-axis). Six wheel and four wheel configurations were analyzed along with the worst case single wheel failure for each configuration. Each configuration was analyzed for three different TELDIX reaction wheel versions (Torque-Momentum Storage).
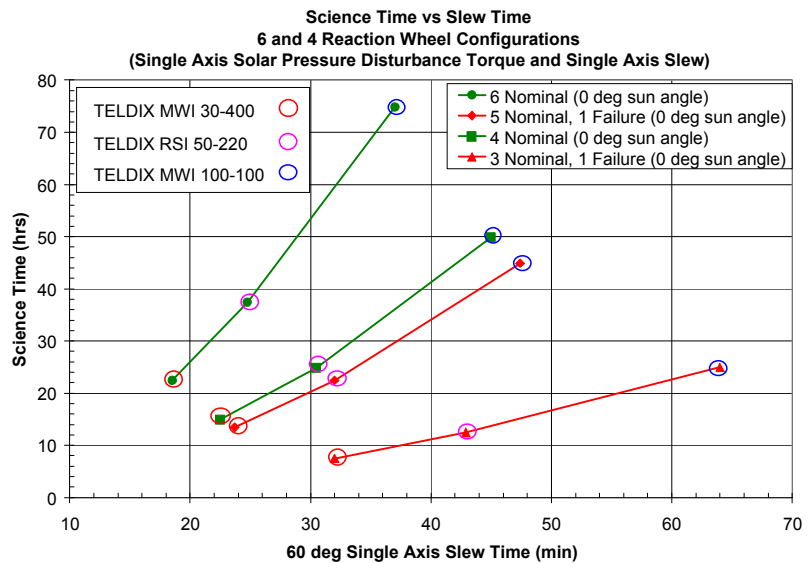

Figure 9 Science Time vs Slew Time Analysis for various Reaction Wheels

3.7 Mass Budget

The entire mass budget for the 6-meter observatory including primary mirror, structure, light baffle tube, instruments, space craft, avionics, etc. is less than 35,000 kg (Table 4) – a 38% mass margin on the Ares V's 55,600 kg Sun-Earth L2 launch capability. The mass budget for an 8 meter observatory is approximately 45,000 kg, with almost a 20% mass margin, of which the primary mirror is the largest contributor. These mass budgets clearly show that payload diameter/volume, and not the payload mass, is the limiting factor in the telescope size.

Table 4 Mass Budget for a 6 meter Telescope OTE and Spacecraft/Instrument Bus

	Mass (Kg)
Total mass = OTE W / Bus + Spacecraft and Science Inst	**33,849**
OTE W / Bus mass	**25,619**
Primary mirror assembly	17750
Secondary mirror assembly	671
Telescope enclosure	3,600
Avionics Subsystems	153
Power Subsystems	381
Thermal Management System	1,091
Structures	917
Propulsion	16
Propellant	40
Docking station	1,000
Spacecraft and Science Instrument	**6,230**
Science Instrument Package	1500
Avionics Subsystems	334
Power Subsystems	377
Thermal Management System	481
Structures	755
Propulsion	248
Propellant	1,536
Docking station	1,000
Launch Adapter	**2,000**

Please note that several elements of this mass budget are allocations, including the science instrument package, launch adapter and docking stations. All mass elements will be subject to refinement as the design matures.

4. EXTENDED MISSION LIFE BY IN-SPACE SERVICING

To extend the life of the observatory beyond its initial design life of 5 years to a target life of 30 years or more, the science instruments and as many subsystem components as possible are designed to be replaced at periodic intervals. These are all in the SIB (Figure 8) which can be replaced as a single unit every 3 to 5 years using autonomous rendezvous and docking (AR&D) technology (as demonstrated on Orbital Express). Beyond the obvious technical advantages of upgrading detectors, electronics and computers periodically, it is anticipated that designing subsystems for 5 years of operation instead of 10 years will produce sufficient cost savings to fund the periodic servicing missions. (This needs to be the subject of a design trade.) The SIB diameter is set at 4.5 meters such that these servicing missions can be launched via a conventional EELV.

Using this approach, the first 5 years of mission life could be dedicated to UV science with a narrow FOV UV spectrometer and a WFOV UV imager. Then, the next 5 years of mission life could be dedicated to visible science such as terrestrial planet finding with either an external occulter or an internal coronagraph. Eventually, it might be possible to have two different SIBs on station with the ability to switch between suites of science instruments. At the observatory end of life, as thermal control degrades, the telescope can be allowed to cool to <200K for an infrared science campaign.

During the period of SIB exchange, when the SIB is undocked from the observatory, the telescope spacecraft provides basic guidance and navigation for station keeping. The telescope has 18 m2 of body mounted solar array around light tube, used for station keeping, and batteries for up to 0.5 hour of attitude control contingency. The telescope avionics systems are 3-fault tolerant for a 30 year life. As previously discussed, the telescope has a mono-propellant blow-down thrust system. The telescope also has a low gain antenna for communicating with the servicing spacecraft. All telescope health and status data is sent directly to the spacecraft avionics system. Also, power for the telescope thermal management system is provided by the SIB. Thus, there is no active thermal control during spacecraft exchange.

The primary subsystems for pointing, communications, power, guidance, propulsion, as well as the science instrument package and fine guidance sensor, are located on the SIB. One notable exception is the thermal control for the primary mirror, which must be placed on the telescope bus. The SIB avionics and power systems are 1-fault tolerant for 5 year life. Power is generated from two 9 m2 deployable solar array wings with pointing ability. Batteries are sized for 2 hours of power during midcourse and rendezvous operations (when the power arrays are retracted). The SIB power system includes 800W for mirror thermal control and 750W for the telescope instrument package. The guidance and navigation system includes star trackers, sun sensors and inertial measurement units. AR&D will be facilitated with a LIDAR long range system and an optical short range system. Computers handle all normal station keeping, maneuvers, data management, and ground communications. And, the communication systems consist of Ka-band HGA for ground, and s-band for local communication and backup capability

5. CONCLUSION

The unprecedented mass and volume capabilities of NASA's planned Ares V cargo launch vehicle enable entire new mission concepts. Its 10 meter fairing and ability to place 55,600 kg of payload into Sun-Earth L2 completely changes the paradigm for future space telescopes – simplicity. Simple high TRL technology offers lower cost and risk. The Ares V capacities allow one to use mass to buy down performance, cost and schedule risk. Instead of expensive lightweight space mirrors, one can use low-cost low-risk proven ground based mirror technology. And, instead of expending excessive amounts of engineering design and analysis labor, the Ares V payload capacity allows for mission designs with larger than normal structural safety margins and fewer complex deployment mechanisms.

NASA Marshall Space Flight Center has conducted a preliminary design study which indicates that it is feasible to launch a 6 to 8 meter class monolithic primary mirror ultraviolet/visible observatory. An 8-meter class UV/optical space observatory with its very high angular resolution, very high sensitivity, broad spectral coverage, and high performance stability offers the opportunity to answer some of the most compelling science questions. How did the present Universe come into existence and of what is it made? What are the fundamental components that govern the formation of today's galaxies? How does the Solar System work? What are the conditions for planet formation and the emergence of life? And maybe most importantly, are we alone?

Finally, there is no inherent reason that an 8-meter space telescope using robust design concepts should have only a 5 to 10 year mission life. In fact, there is no reason that the telescope might not last 20 to 30 years. This extended mission life can be obtained via periodic robotic servicing of the spacecraft and science instruments using autonomous rendezvous and docking technology (as demonstrated on Orbital Express).

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