Design study of 8 meter monolithic mirror UV/optical space telescope

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ABSTRACT

The planned Ares V launch vehicle with its 10 meter fairing shroud and 55,000 kg capacity to the Sun Earth L2 point enables entirely new classes of space telescopes. NASA MSFC has conducted a preliminary study that demonstrates the feasibility of launching a 6 to 8 meter class monolithic primary mirror telescope to Sun-Earth L2 using an Ares V. Specific technical areas studied included optical design; structural design/analysis including primary mirror support structure, sun shade and secondary mirror support structure; thermal analysis; launch vehicle performance and trajectory; spacecraft including structure, propulsion, GN&C, avionics, power systems and reaction wheels; operations & servicing; mass and power budgets; and system cost.

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

1. INTRODUCTION

An 8-meter class space telescope 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? (Postman, 2008) A recent design study conducted at Marshall Space Flight Center has shown 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 included optical design; structural design/analysis including primary mirror support structure, sun shade and secondary mirror support structure; thermal analysis; launch vehicle performance and trajectory; spacecraft including structure, propulsion, GN&C, avionics, power systems and reaction wheels; operations & servicing; mass and power budgets; and system cost.

Figure 1  Ares V can launch 6 to 8 meter class monolithic mirror telescope. (Image courtesy of Jack Frassanito & Associates and Harley Thronson)
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.

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 of extra mass, 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.

These two unprecedented enabling capabilities of the Ares V formed the basis for the foundational question of the MSFC design study. Is it possible to launch an 8-meter class space observatory using a conventional massive monolithic ground based telescope mirror? Instead of using lightweight (very expensive and high risk) mirror technology, it is possible to use a conventional massive (low cost and low risk) ground telescope mirror? And the answer is YES.

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**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.)
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.

3.2 Optical Design

The feasibility study considered two different telescope optical systems. An F/15 Ritchey-Chretién 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.

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, the easier it is to achieve a very good surface figure. Second, the cost for an 8 meter ground mirror is approximately $20M or $0.4M/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

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Spitzer</th>
<th>HST</th>
<th>AMSD</th>
<th>JWST</th>
<th>Ground</th>
</tr>
</thead>
<tbody>
<tr>
<td>Material</td>
<td>Beryllium</td>
<td>ULE</td>
<td>ULE &amp; Be</td>
<td>Be</td>
<td>Various Glass</td>
</tr>
<tr>
<td>Diameter</td>
<td>0.85</td>
<td>2.4</td>
<td>1.4</td>
<td>1.5 (6.5)</td>
<td>8.2 m</td>
</tr>
<tr>
<td>Area</td>
<td>0.5</td>
<td>4.5</td>
<td>~1</td>
<td>25</td>
<td>50 m²</td>
</tr>
<tr>
<td>Temperature</td>
<td>4</td>
<td>300</td>
<td>300/30</td>
<td>30</td>
<td>300 K</td>
</tr>
<tr>
<td>Surface Figure</td>
<td>75</td>
<td>6.4</td>
<td>20/77</td>
<td>25</td>
<td>8 mm rms</td>
</tr>
<tr>
<td>Areal Density</td>
<td>28</td>
<td>180</td>
<td>18</td>
<td>26</td>
<td>460 kg/m²</td>
</tr>
<tr>
<td>Areal Cost</td>
<td>10</td>
<td>10</td>
<td>4</td>
<td>6</td>
<td>0.5 $M/m²</td>
</tr>
</tbody>
</table>

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, Corning manufactured the Subaru Telescope ULE thin meniscus 8.3 meter primary mirror blank with 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

<table>
<thead>
<tr>
<th>Maximum Launch Load</th>
<th>Ares V</th>
</tr>
</thead>
<tbody>
<tr>
<td>Axial (Z)</td>
<td>4 g’s</td>
</tr>
<tr>
<td>Lateral (Y)</td>
<td>7 x 10-6 g’s</td>
</tr>
<tr>
<td>Lateral (Z) (Z is down range direction)</td>
<td>6 x 10-4 g’s</td>
</tr>
</tbody>
</table>

A structural analysis determine that 66 axial support points keeps 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.

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 Structural concept is that all the mass of the observatory is carried through the main support structure to an interface ring which attaches via another support system 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.

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 m$^2$ 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>
<thead>
<tr>
<th>Sun Angle</th>
<th>Temperature</th>
</tr>
</thead>
<tbody>
<tr>
<td>0 deg</td>
<td>200K</td>
</tr>
<tr>
<td>45 deg</td>
<td>190K</td>
</tr>
<tr>
<td>90 deg</td>
<td>160K</td>
</tr>
<tr>
<td>120 deg</td>
<td>300K</td>
</tr>
</tbody>
</table>

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 for long exposure observations which required a very stable observatory wavefront (such as extra-solar terrestrial planet finding and characterization). The primary mirror surface figure varies as a function of temperature based on the substrate material CTE value and uniformity.

At 300K, Corning specifies ULE to have a mean CTE value of 0 +/- 30 ppb/K from 5C to 35C and Schott specifies that Zerodur has a CTE value of 0 +/- 50 ppb/K with a uniformity of +/- 10 ppb/K.
3.6 Spacecraft

The observatory has two separate spacecraft: a telescope bus which houses 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.

The spacecraft propulsion system is sized to get the observatory from roughly a geostationary transfer orbit (energy, or C³, of -2.60 km²/s²) into a halo orbit about the Sun-Earth L2 point and performs 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 V expenditure for five years of 20 m/s. Propulsion during the trip from GTO 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 with 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). 6 wheel and 4 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).

3.7 Mass Budget

The entire mass budget for the 6-meter observatory including primary mirror, structure, light baffle tube, instruments, spacecraft, 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.
To stop health and status data is sent directly to the spacecraft avionics system. Also, power for the telescope thermal thrust system. The telescope also has a low gain antenna for communicating with the systems are 3 tube, used for station keeping basic guidance and navigation for station keeping.

The telescope has 18 m² of

A 6 meter Telescope OTE and Spacecraft/Instrument Bus

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 a 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.

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

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

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 a 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. COST DISCUSSION

The proposed 8 meter monolithic telescope concept seeks to disprove the old adage that the primary predictor of mission cost is mass. It is the author’s assertion that the Ares V payload mass capability is a disruptive technology that creates a new paradigm - by trading mass for simplicity it is possible to build a telescope with lower cost and lower risk. By eliminating complexity, it should be possible to design and build an 8-meter monolithic telescope with 2X the collecting area of the 6.5 meter JWST for less cost. Consider for example the complexity difference between packaging a 6.5 meter segmented primary mirror into a 4.5 meter dynamic launch envelope as compared to the simplicity of packaging an 8 meter monolithic mirror into an 8.8 meter dynamic launch envelope. The current cost for the JWST telescope and spacecraft (excluding science instruments and operation) is approximately $3B. The total cost for an 8-meter observatory (excluding science instruments and operations is estimated to be $1B to $1.5B. To illustrate this point further, consider the telescope primary mirror and its support structure.

Because of launch vehicle payload mass constraints, all previous space based telescopes have required low areal density primary mirrors - the bigger the telescope, the lower its required areal density. For example, the Hubble primary mirror has an areal density of 180 kg/m2 for a total mass of 810 kg or 7.4% of Hubble’s total mass of 11,000 kg. By comparison, the JWST primary mirror has an areal density of 25 kg/m2 for a total mass of 625 kg or 9.6% of the total JWST observatory mass of 6,500 kg. The explanation is simple, a primary mirror mass is limited to about 10% of a space observatory and the total mass of the observatory is limited by the launch vehicle. And, because space mirrors have low areal densities, they are difficult to manufacture and thus expensive. Space mirrors are inherently less stiff than ground mirrors. Thus, they have larger gravity sags; exhibit a fabrication effect called quilting; and are difficult to handle, mount and fixture. Because of this mass and stiffness difference, the cost of a space telescope mirror is typically 10X higher than for a ground telescope mirror.

The Ares V eliminates this constraint. The 8-meter monolithic concept proposes to use existing ground based mirror technology rather than the ultra light-weight mirror technology required for a large space telescope via an EELV. While this architectural choice adds approx 20,000 kg to the mass of the payload, it is estimated to save $200M to $500M in mirror hardware costs and $700M to $2B in total program costs. The precursor JWST mirror technology development program AMSD demonstrated that 1.4 meter light-weight beryllium and glass mirrors both cost approx $4M per square meter. Currently, the total cost for the 6.5 meter JWST primary mirror is in excess of $140M or close to $6M/m2. Thus, a 50 square meter mirror will cost $200M to $300M. Furthermore, It is likely that a UV/Visible quality version of the JWST primary mirror would be even more expensive, maybe $500M (using the HST $10M/m2 areal cost). By comparison, 8-meter class (50 square meter) UV/Visible quality ground based telescope mirrors typically cost less than $20M or $0.4M/m2. Given that 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, a $200M savings in the cost of a primary mirror translates into a $700M to $800M total program cost savings.

Risk is also significantly lower for a ground based mirror simply because they have been demonstrated. Currently there are nine 8-meter class monolithic telescopes in operation. Some of these mirrors have a surface figure better than 10 nm rms – close to the requirement for the Terrestrial Planet Finder primary mirror.

Similar cost savings and risk reductions are anticipated for the telescope structure (ATK private conversation). On JWST, the cost of the telescope structure is approximately 2/3rd engineering labor and 1/3rd fabrication & tooling. The
primary driver for the engineering labor is the need to design a very low mass structure at the limit of performance safety factors. It has been estimated that if the mass could be increased by several factors (3X to 5X) that the engineering time could be cut substantially. For an 8 meter class structure with a mass allocation of 10,000 kg, it is estimated that engineering labor will account for 1/3rd of the total cost with the balance for fabrication. And, that the total cost for an 8 meter telescope should not exceed the total cost of JWST (note also that JWST is cryogenic and the proposed UV/optical telescope operates at ambient temperature).

6. 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).

ACKNOWLEDGEMENTS

The author would like to thank Randall Hopkins and Alan Philips of the Marshall Space Flight Center Advanced Concepts Office who provided information, analysis, data and valuable guidance regarding the Ares V capabilities and preliminary shroud designs and John Hraba, Bill Arnold and Ken Pitalo of the MSFC Optics Office.

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Executive Summary

The unprecedented volume capability of an Ares V enables the launch of 8 meter class monolithic space telescopes to the Earth-Sun L2 point.

The unprecedented mass capability of an Ares V enables an entirely new design paradigm – Simplicity.

Simple high TRL technology offers lower cost and risk.

NASA MSFC has determined that a 6 to 8 meter class telescope using a massive high-TRL ground observatory class monolithic primary mirror is feasible.
Ares V Performance Capability

(at this point in time)
Ares V delivers 5X more Mass to Orbit

Current Capabilities can Deliver
- 23,000 kg to Low Earth Orbit
- 10,000 kg to GTO or L2TO Orbit
- 5 meter Shroud

Ares V can Deliver
- ~130,000 kg to Low Earth Orbit
- ~60,000 kg to L2TO Orbit (w/ phasing loops)
- 10 meter Shroud
Building on a Foundation of Proven Technologies

– Launch Vehicle Comparisons –

**Space Shuttle**
- Height: 56.1 m (184.2 ft)
- Gross Liftoff Mass: 2,041,166 kg (4.5M lbm)
- 25 MT (55k lbm) to Low Earth Orbit (LEO)

**Ares I**
- Height: 99.1 m (325 ft)
- Gross Liftoff Mass: 907,185 kg (2.0M lbm)
- 25.6 MT (56.5k lbm) to LEO

**Ares V**
- Height: 109.7 m (360 ft)
- Gross Liftoff Mass: 3,374,910 kg (7.4M lbm)
- 55.9 MT (123k lbm) to Direct TLI
- ~143.4 MT (316k lbm) to LEO

**Saturn V**
- Height: 110.9 m (364 ft)
- Gross Liftoff Mass: 2,948,350 kg (6.5M lbm)
- 45 MT (99k lbm) to TLI
- 119 MT (262k lbm) to LEO

**Altair**
- Height: 91 m (300 ft)
- 234,488 kg (517k lbm)

**Lunar Lander**
- Height: 61 m (200 ft)
- 108,862 kg (240k lbm)

**Orion**
- Height: 61 m (200 ft)
- 453,592 kg (1M lbm)
- 138,350 kg (302k lbm)

**Upper Stage**
- Height: 91 m (300 ft)
- 1,435,541 kg (3.2M lbm)

**Core Stage**
- Height: 91 m (300 ft)
- 1,769,010 kg (3.9M lbm)

**Solid Rocket Booster**
- Height: 30 m (100 ft)
- 907,185 kg (2.0M lbm)

**Earth Departure Stage**
- Height: 91 m (300 ft)
- 2,948,350 kg (6.5M lbm)
Current Ares V 10 meter Shroud

Shroud Dimensions

- 7.50 m [24.6 ft]
- 9.70 m [31.8 ft]
- 10.0 m [33.00 ft]

Usable Dynamic Envelope

- 4.44 m [14.6 ft]
- 8.80 m [28.9 ft]

Useable Volume

~860 m³

meters [feet]
Notional Ares V Shroud for Other Missions

Useable Volume
~1410 m³

* Note: The height of the shroud is limited by the height of the Vertical Assembly Building (VAB)
## Ares V Performance for Selected Missions

<table>
<thead>
<tr>
<th>Mission Profile</th>
<th>Target</th>
<th>Payload Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sun-Earth L2</td>
<td>C3 of -0.7 km$^2$/s$^2$ @ 29.0 degs</td>
<td>55,800</td>
</tr>
<tr>
<td>GTO Injection</td>
<td>Transfer DV 8,200 ft/s Final Orbit: 185 km X 35,786 km @ 27 deg</td>
<td>70,300*</td>
</tr>
<tr>
<td>GEO</td>
<td>Transfer DV 14,100 ft/s Final Orbit: 35,786 km Circular @ 0 degrees</td>
<td>36,200</td>
</tr>
<tr>
<td>Cargo Lunar Outpost (TLI Direct)</td>
<td>C3 of -1.8 km$^2$/s$^2$ @ 29.0 degs</td>
<td>56,800</td>
</tr>
</tbody>
</table>

* Performance impacts from structural increases due to larger payloads has not been assessed
Ares V LEO Performance

Ares V Payload vs. Altitude & Inclination (LV 51.00.39)

Circular Orbital Altitude (km)

Payload (tonnes)

- Inclination = 29 deg
- Inclination = 35 deg
- Inclination = 40 deg
- Inclination = 45 deg
- Inclination = 51.6 deg
Ares V Escape Performance

Payload vs. C3 Energy

- Ares V
- Ares V with Centaur V2

Payload (tonnes) vs. C3 Energy (km²/s²)
Design study of an 8-meter monolithic mirror UV/optical space telescope

H. Philip Stahl, Ph.D.
NASA MSFC
8 meter Monolithic Telescope & tube can fit inside Ares V 10 m envelop.

Minimize **Cost (& Risk)** by using **existing ground** telescope mirror technology – optics & structure.

8-meter diameter is **State of Art**
- 9 existing: VLT, Gemini, Subaru, LBT
- 23,000 kg (6 m would be ~13,000 kg)
- ~$30M (JWST PM cost ~$120M)
- 7.8 nm rms surface figure (~TPF spec)

Expect similar savings for structure
6 meter Optical Design

Ritchey-Chretién optical configuration
F/15
Diffraction Limited Performance at <500 nm
Diffraction Limited FOV of 1.22 arc minute
   (10 arc minute FOV with Corrector Group)
Coating: Aluminum with Mg F2 overcoat
   Average transmission > 63% for wave lengths of 200 to 1,000 nm
Primary to secondary mirror vertex: 9089.5 mm
Primary mirror vertex to focal plane: 3,000 mm

All Reflective Design
   Three Mirror Anastigmatic
   With Fine Steering Mirror
   Multi-Spectral 10 arc min FOV
   Reduced Throughput
Structural Design

Launch Configuration

Tube is split and slides forward on-orbit. Faster PM or taller shroud may allow for one piece tube.

Doors can open/close

Forward Structure is hybrid of Hubble style and four-legged spider

Truss Structure interfaces with 66 mirror support attachment locations

Launch Structure attaches Truss to Ares V
Structural Analysis

Launch loads: *maximum* values from POST3D (not concurrent)

- **Axial:** 4 g’s
- **Lateral-y:** $7 \times 10^{-6}$ g’s
- **Lateral-z:** $6 \times 10^{-4}$ g’s

8.2 meter 175 mm thick meniscus primary mirror can survive launch. 66 axial supports keep stress levels below 1000 psi.
Spacecraft Structural Modeling

Instrument Frame & Outer Skin Not Shown

3X Docking Latches

Instrument Interface

Upper Shelf

2X GHe Tank Skirt

NTO Tank Skirt

Middle Shelf

4X MMH Tank Skirt

MPS Nozzle Openings

Lower Shelf

Avionics & Power System Attachments
Spacecraft Structural Analysis Assumptions

Launch Load Case: 4.0g Axial + 2.0g Lateral

Materials: Metallic Structure Only
- AA 2219 for plate elements
- AL 7075 for Beam Elements

Factors of Safety: (per NASA-STD-5001)
- Yield Factor of Safety: 1.1
- Ultimate Factor of Safety: 1.4

Cross-Sectional View of Spacecraft
Structural Model Results

Upper Shelf:
- Shelf: Isogrid Panel 0.090” (minimum pocket thickness)

Middle Shelf:
- Shelf: Isogrid Panel 0.060” (minimum pocket thickness)
- MMH Skirts: 0.064” thk
- NTO Skirt: 0.088” thk
- GHe Skirt: 0.040” thk

Lower Shelf:
- Shelf: Isogrid Panel 0.060” (minimum (pocket thickness)

Instrument Support Frame:
- Upper Support: “T” Beam, 0.095” thk
- Uprights: 2” diameter, 0.030” thk
- Angled Supports: 1.75” diameter, 0.030” thk

Outer Skin:
- Upper Outer Skin: 0.26” thk
- Lower Outer Skin: 0.21” thk
Spacecraft Design Detail & Shroud Integration

Science Envelope = 2.5 m x 2.5 m x 2.0 m

NOTE: All dimensions are in meters.
# 6 meter Preliminary Mass Budget

<table>
<thead>
<tr>
<th>Mass (Kg)</th>
<th>Heritage</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Primary mirror assembly</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Primary mirror</td>
<td>13,000</td>
<td>calculated</td>
</tr>
<tr>
<td>Primary mirror support structure</td>
<td>6,750</td>
<td>estimate</td>
</tr>
<tr>
<td>Primary mirror center baffle</td>
<td>250</td>
<td>estimate</td>
</tr>
<tr>
<td><strong>Secondary mirror assembly</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Secondary mirror</td>
<td>100</td>
<td>calculated</td>
</tr>
<tr>
<td>Secondary mirror support &amp; drive</td>
<td>150</td>
<td>estimate</td>
</tr>
<tr>
<td>Secondary mirror baffle</td>
<td>30</td>
<td>estimate</td>
</tr>
<tr>
<td>Secondary mirror spider</td>
<td>400</td>
<td>estimate</td>
</tr>
<tr>
<td><strong>Telescope enclosure</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Metering structure with internal baffles</td>
<td>2,800</td>
<td>estimate</td>
</tr>
<tr>
<td>Rear cover</td>
<td>300</td>
<td>estimate</td>
</tr>
<tr>
<td>Head ring</td>
<td>200</td>
<td>estimate</td>
</tr>
<tr>
<td>Front cover &amp; actuator</td>
<td>300</td>
<td>estimate</td>
</tr>
<tr>
<td>Attitude Determination and Control System</td>
<td>150</td>
<td>JWST</td>
</tr>
<tr>
<td>Communications</td>
<td>76</td>
<td>JWST</td>
</tr>
<tr>
<td>Command And Data Handling System</td>
<td>54</td>
<td>JWST</td>
</tr>
<tr>
<td>Power</td>
<td>380</td>
<td>JWST</td>
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<tr>
<td>Thermal Management System</td>
<td>1090</td>
<td>JWST</td>
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<tr>
<td>Structures</td>
<td>920</td>
<td>estimate</td>
</tr>
<tr>
<td>Guidance and Navigation</td>
<td>50</td>
<td>estimate</td>
</tr>
<tr>
<td>Propulsion</td>
<td>20</td>
<td>JWST</td>
</tr>
<tr>
<td>Computer Systems</td>
<td>50</td>
<td>estimate</td>
</tr>
<tr>
<td>Propellant</td>
<td>50</td>
<td>JWST</td>
</tr>
<tr>
<td>Docking station</td>
<td>1,000</td>
<td>estimate</td>
</tr>
</tbody>
</table>

| **OTE W / Bus mass** | | |
| Science Instrument | 1500 | JWST | ISIM, contains Fine Guidance Sensor |
| Attitude Determination and Control System | 150 | JWST | estimate plus JWST scaled |
| Communications | 76 | JWST | |
| Command And Data Handling System | 54 | JWST | |
| Power | 380 | JWST | |
| Thermal Management System | 480 | JWST | |
| Structures | 755 | estimate | WAG |
| Guidance and Navigation | 50 | estimate | 50% WAG |
| Propulsion | 250 | JWST | |
| Computer Systems | 50 | estimate | WAG |
| Propellant | 1530 | JWST | |
| Docking station | 1,000 | estimate | WAG |

| **Science Instrument W / Bus mass** | | |
| Science Instrument | 1500 | JWST | ISIM, contains Fine Guidance Sensor |
| Attitude Determination and Control System | 150 | JWST | estimate plus JWST scaled |
| Communications | 76 | JWST | |
| Command And Data Handling System | 54 | JWST | |
| Power | 380 | JWST | |
| Thermal Management System | 480 | JWST | |
| Structures | 755 | estimate | WAG |
| Guidance and Navigation | 50 | estimate | 50% WAG |
| Propulsion | 250 | JWST | |
| Computer Systems | 50 | estimate | WAG |
| Propellant | 1530 | JWST | |
| Docking station | 1,000 | estimate | WAG |

38% Mass Reserve

Total mass = OTE W / Bus + Science Instrument W / Bus = 34,395 kg

8 meter Preliminary Budget is 45,000 kg (~20% Reserve)
Thermal Analysis

Spacecraft wrapped with 10 layer MLI blankets

16.0 m² thermal radiators

Load Cases

0° (base)
45°
90° (broadside)
120°
Spacecraft Thermal Analysis

Solar Flux at L2 = 1296 W/m² applied to base
Instrument Heat Output = 750 W
Avionics Heat Output = 850 W
Propellant tanks modeled as single nodes with heat leaks from the spacecraft walls
Steady-state operational temperatures determined
Spacecraft wrapped with 50 layer MLI blankets
16.0 m² thermal radiators
Propellant tanks maintained with MLI and heaters
Heaters required to keep propellant from freezing
Primary Mirror Thermal Analysis Results

Sun = 0°

45°

90°

120°

* Temperatures are in °C. Note varied temperature scale for each load case.
Primary Mirror Thermal Analysis

Active Thermal Management via 14 Heat Pipes yields a Primary Mirror with less than 1K Thermal Variation.

No Thermal Management yields a Cold PM

<table>
<thead>
<tr>
<th>Sun Angle</th>
<th>Temp</th>
</tr>
</thead>
<tbody>
<tr>
<td>0 deg</td>
<td>200K</td>
</tr>
<tr>
<td>90 deg</td>
<td>160K</td>
</tr>
<tr>
<td>120 deg</td>
<td>300K</td>
</tr>
</tbody>
</table>

with 1K Thermal Variation

Thus, possible End of Life use as a NIR/Mid-IR Observatory.

Figure Change will be driven by CTE Change from 300K to 150K

Zerodur CTE is approximately 0.2 ppm.
SiO2 CTE is approx 0.6 ppm.
Notional Spacecraft Propulsion System

Dual Mode: Hydrazine-NTP Bi-Prop / Hydrazine Mono-Prop

Propellant for 5 yr mission with redundant Thrusters

Key

- Filter
- Surge Orifice
- Check Valve
- Pressure Transducer
- Single Coil Latching Valve
- Closed Pyro Valve
- Open Pyro Valve
- Burst Disk
- Relief Valve
- Temperature Sensor
- Series Redundant Pressure Regulator
- Capped Service Valve
- Dual Coil, Solenoid Valve

Hydrazine Mono-Prop with RCS 20/5 lbf Thrusters (Aerojet) for Station Keeping

Hydrazine-NTP Bi-Prop with four 125 lbf Thrusters (Northrop) for trip to L2
Telescope has Independent Control System
Mono-Propellant Hydrazine

Trade Analysis:
Refueling (Orbital Express) = 40 kg
30 Year Propellant Supply = 30 kg

350 – 100 psi blowdown Aerojet Thrusters
Guidance Navigation and Pointing Control

Spacecraft Reaction Wheels provide all GNC

Worst condition for solar radiation pressure torque is at sun angle $\alpha = 90$. 

Momentum buildup occurs in one axis (y-axis)
GN&C Analysis

Two performance Parameters were analyzed and plotted against each other:

• Hours that Telescope can stare at a fixed point (remain at an inertial hold) before needing to perform a momentum dump due to solar radiation pressure torque

• How fast in minutes the Telescope can perform a 60 degree slew

6 wheel and 4 wheel configurations were analyzed along with the worst case single wheel failure for each configuration.

Each configuration was analysis for three different TELDIX reaction wheel versions with different (Torque : Momentum Storage)

Analysis

is only for the worst case sun angle = 0

As the sun angle increases so does the available science time.

did not account for any solar panel contribution to solar pressure cp location.

This is worst case since accounting for the solar panels would move the cp location closer to the cg. Also, Telescope geometry is preliminary and may change due refinement in design.
GNC: Reaction Wheels

Science Time vs Slew Time
6 and 4 Reaction Wheel Configurations
(Single Axis Solar Pressure Disturbance Torque and Single Axis Slew)

- TELDIX MWI 30-400
- TELDIX RSI 50-220
- TELDIX MWI 100-100

Best

Science Time (hrs)

60 deg Single Axis Slew Time (min)
Avionics and Power Systems Assumptions

Spacecraft

Avionics

• Spacecraft avionics systems are 1-fault tolerant for 5 year life
• Guidance and navigation system includes star trackers, sun sensors, and IMUs
• AR&D consists of a LIDAR long range system, and an optical short range system
• Computers handle all normal station keeping, maneuvers, data management, and ground communications
• Communication systems consist of Ka-band HGA for ground, and s-band for local comm and backup capability

Power

• Spacecraft power systems are 1-fault tolerant for 5 year life
• Power generation from two 9 m^2 deployable solar array wings with pointing ability
• Batteries are sized for 2 hours of power for midcourse and rendezvous operations (with arrays retracted)
• Spacecraft power system includes 800 w for mirror thermal control, and 750 w for telescope instrument package
Avionics and Power Systems Assumptions

Telescope

Avionics

• Telescope avionics systems are 3-fault tolerant for 30 year life
• Minimal guidance and navigation system, used only for station keeping during spacecraft exchange
• Minimal computer capability, used mainly for station keeping during spacecraft exchange
• All health and status data sent directly to spacecraft avionics system
• Low gain communications capability with the servicing spacecraft only

Power

• Telescope power systems are 3-fault tolerant for 30 year life
• 18 m^2 body mounted solar array around light tube, used for station keeping during spacecraft exchange
• Batteries sized for 0.5 hour attitude control contingency
• No active mirror thermal control during spacecraft exchange
Spacecraft Astrionics & Power Systems

6m Telescope - Spacecraft Astrionics and Power Systems

3-7-07
Mission Life

Initial Mission designed for a 5 yr mission life (10 yr goal) should produce compelling science results well worth the modest mission cost.

But, there is no reason why the mission should end after 5 or even 10 years.

Hubble has demonstrated the value of on-orbit servicing.

The telescope itself could last 30 or even 50 years.
30 to 50 year Mission Life

Copy Ground Observatory Model – L2 Virtual Mountain

Design the observatory to be serviceable
  Telescope has no inherent life limits
  Replace Science Instruments every 3-5 yrs (or even 10 yrs)

Replacement
Spacecraft in ELV

Observatory has split bus with on-board
  attitude control and propulsion during
  servicing. (already in mass budget)

Autonomously docks to observatory;
  replaces all science instruments and
  ALL expendable components.

Spacecraft in
  4.5 meter Payload Fairing
AR&D System Elements

**Spacecraft**
- Inertial Reference Unit
- Attitude Sensors
- On-Board Computer GN&C
- Attitude Control System
- Comm
- Long and Short Relative Nav Sensors
- Docking Mechanism

**Telescope**
- Inertial Reference Unit
- Attitude Sensors
- On-Board Computer GN&C
- Attitude Control System
- Comm
- Docking Mechanism

- Telemetry Uplink to spacecraft provides target (Telescope) vehicle state vector to allow Long Range Relative Navigation sensor acquisition of target to begin rendezvous guidance.

- Spacecraft Rendezvous guidance commands generated by using Relative Navigation sensor measurements.

- Short range Relative Navigation sensor provides measurements to guidance for docking.
Conclusions

The unprecedented mass/volume capability of an Ares V enables the launch of 8 meter class monolithic space telescopes to the Earth-Sun L2 point.

NASA MSFC has determined that a 6 to 8 meter class telescope using a massive high-TRL ground observatory class monolithic primary mirror is feasible.

Mature, High-TRL design enables early deployment.

Science Instruments, Expendables and Limited Life Components can be replace periodically via Spacecraft Autonomous Rendezvous and Docking.
Any Question?