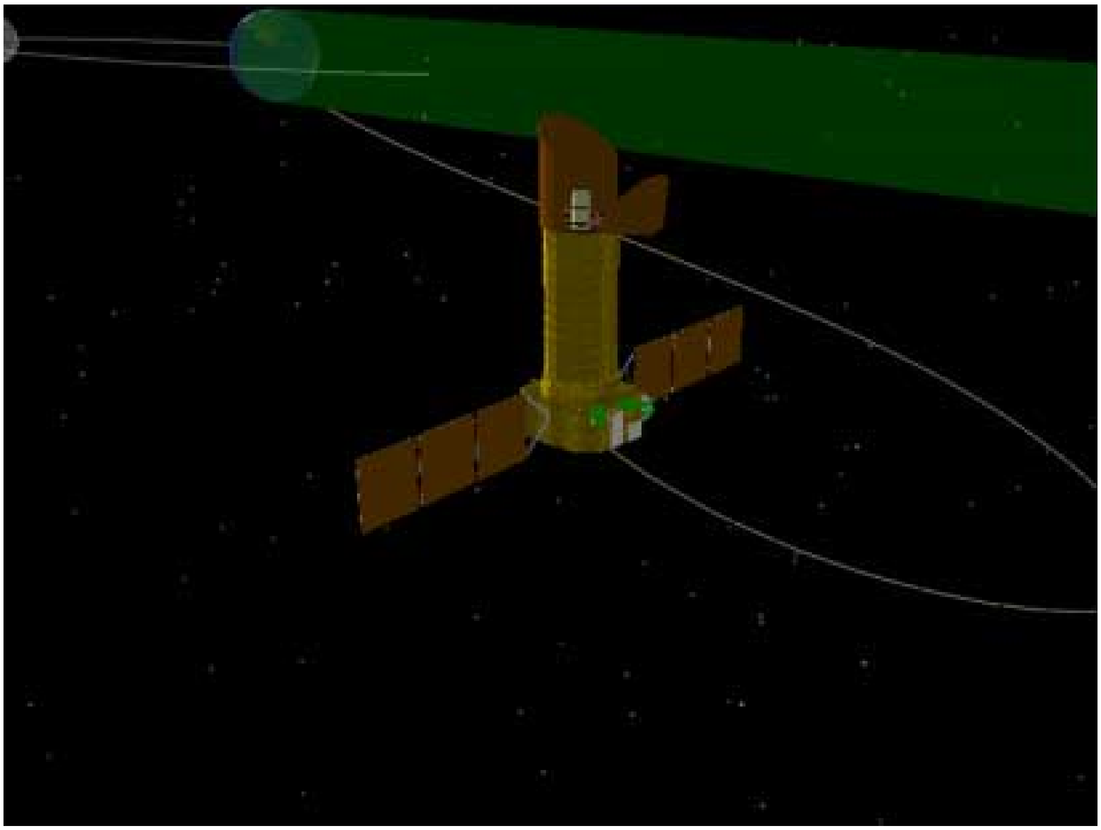
Advanced Projects Design Team's Assessment of

ESA's World Space Observatory Proposal



The JPL Advanced Design Team Assessment report of the World Space Observatory does not intend or imply any endorsement by the National Aeronautics and Space Administration of this project.

Assessment Date: 10-24-00 Release Date: 06-01-01

WSO APD TEAM ASSESSMENT EXECUTIVE SUMMARY

The World Space Observatory (WSO) offers the international community an ultraviolet space telescope that observes the universe from the solar system to distant galaxies. It also provides the opportunity for international cooperation. An ESA study provided a basic design for the WSO mission that has been reviewed by the Jet Propulsion Laboratory's Advanced Projects Design Team (APD Team). The assessment covered the scientific objectives and rationale as well as the spacecraft conceptual design. APD Team concluded that overall the WSO mission is both technically and scientifically feasible. The minor issues addressed in this review have been identified in the WSO Report and will be resolved in future development phases.

Science

The World Space Observatory will provide observations in a wavelength regime not adequately provided for in current or future space telescope missions. A wide range of ultraviolet astronomy will be possible, from distant galactic and black hole observations to comet and atmosphere observations in our solar system. The WSO will complement the Hubble Space Telescope (HST) capabilities in the UV regime including UV imaging and spectroscopy as well as high resolution visible imaging of planetary systems. The WSO observations will follow the work performed by the GALEX UV survey. Most importantly, the WSO mission successfully addresses the priorities of the astronomy and planetary science community through its ultraviolet observation capabilities.

Instruments

The WSO telescope has a 1.5–2.0 diameter mirror with one UV imaging and one UV spectrometry instrument. The UV imaging instrument will cover the 115–340 nm wavelength regime and provide images with a 0.1 to 0.3 arcseconds quality. There are two imaging detectors, one for maximum spatial resolution and one for maximum sensitivity. At the same time, the spectrograph will provide a 115–340 nm range with a spectral resolution of $5–8 \times 10^4$ with a point spread function <0.3 arcseconds. A visual instrument is included for visual imaging to correlate with the ultraviolet observations. The inclusion of a Solar Radiometer onto the spacecraft could be an important and relatively simple addition which would contribute very much to the understanding of the stability of the integrated solar radiation output.

Systems

The WSO delivers and supports a telescope and instruments to an orbit around the Earth-Sun LaGrange 2 point (L2). The nominal mission length will be 5 years, but consumables will be sized for a 10-year mission goal. The WSO spacecraft will be launched on a Russian Proton into a 200-km parking orbit around the Earth. From there the Proton upper stage boosts the spacecraft into an 800,000-km orbit around L2. The nominal launch data is July 1, 2006.

Overall, this is a very robust design with adequate redundancy for a mission of this length. Contingency placed against the dry mass is sufficient for this mission. This contingency, coupled with the large amount of launch mass margin, helps alleviate concerns of slightly low mass estimates at the subsystem level.

Attitude Control System

The proposed ACS design appears to have a fair chance of success. It requires the development of a custom, high-precision fine guidance sensor and an effective image reconstruction algorithm. Since these tasks are non-trivial, there will be some level of risk, but the conceptual design approach described in the proposal appears to be sound. During Phase A studies, the risk can be examined in greater detail and appropriate plans developed to minimize the risk.

Command and Data System

The basic architecture of the WSO command and data system, as proposed in the ESA study, is a good solid architecture. While other alternate architectures could be studied, there is no reason to believe that a better architecture can be developed. Other trade studies that need to be considered include the selection of the Low-Speed bus and High-Speed bus technologies.

Thermal Control System

The thermal design concept uses flight proven elements, which minimizes risk. The thermal control system accommodates the subsystems and part of the instrument electronics during all expected mission phases and operation modes. The thermal control system radiates the heat dissipated by the instruments and on-board equipment into deep space.

Telecommunications System

The telecommunications system is a robust design with a large telecommunication link margin. There are 3 LGAs with –3 dB gain, which allow easy operation during the low data rate operation. The emergency and the uplink mode are comfortably handled using the HGA as well as the LGA. The antenna pointing requirement is 0.254 degrees to limit the pointing loss to 0.1 dB. The gain of the HGA is assumed to be 30 dBi with a 0.8 meter antenna with an efficiency of 23%. Considering the excess ground station capacity this system design easily accommodates 32 Gbits of data to be downlinked each day.

Power System

The power subsystem must provide 1692 watts for instruments and housekeeping loads at end of life with the arrays at 45 degrees to the sun. This subsystem is designed for a 50 volt regulated bus with redundancy in the battery and battery electronics. The battery capacity is derived from the launch and deployment requirements.

The fixed solar array provides 1642 watts at EOL while tilted to 45 degrees off the sun. BOL power is listed as 2433 watts with 45-degree offset. The panels are populated to 94% with 17% efficient advanced silicon cells. The radiation damage of 16.1 % for 5 years seems too high, and more information is needed for advanced silicon designs. The solar array is the same as XMM and Chandra and was picked for its low cost. The array weighs more than a triple junction array, but the higher weight is acceptable on a Proton launch vehicle.

There are two 12 ampere-hour lithium-ion batteries. The 12 ampere-hour batteries are fully redundant for worst cases discharges.

Propulsion System

This propulsion system design is appropriate for this spacecraft. Monopropellant hydrazine systems have long lives and do not contaminate the spacecraft or science instruments. A spacecraft with a ten-year design life should have redundant thrusters even if the usage of the thrusters is low, as is the case here. It might be wise to deliberately carry an extra 20 to 50% of excess propellant if the spacecraft bus can accommodate the larger tanks. This would take advantage of the Proton's excess launch capability.

Structures

The spacecraft structure concept seems straightforward and robust. The mass estimate for the major bus structure seems conservative, even considering the complexities of mounting the telescope. Since it is heritage from the XMM mission, some economies of design and construction might be realized.

Ground Systems

The mission operations system has large amount of inheritance, which reduces cost and risk. The operations concept needs further discussion in the WSO Science Working Group. The method by which observation requests are made and priorities are established has not been defined in detail. A distributed operations concept is attractive since it allows the spacecraft to be operated like a ground observatory (i.e. scientists work autonomously). There are, however, many inherent issues with the distributed approach. Cost effectiveness and reliability of the distributed approach need to be evaluated.

The ground stations and communications network approach is sound. The baseline tracking plan employs ESA stations at Perth, Kourour, and Vilspa and the Maspalomas station for nominal operations, with 2 –3 hours per day from each station for a total of approximately 8 hours / day of tracking. The cost effectiveness of this scheme should be evaluated -- it might be less expensive to schedule more time from one station rather than small amounts of time at several stations. This, of course, needs to be coordinated with the data-taking activities. User ground stations might also be used (in addition to ESA stations and Maspalomas).

WSO Risk Evaluation:

The WSO Observatory Telescope evaluation showed that overall the design is very robust with an abundance of contingency in addition to a large margin against the proposed launch vehicle, thus the risk to this mission is minimized because of the following conservative design and margins.

- The design is fully redundant.
- The predicted WSO Orbiter launch mass margin is about 30% on the launch vehicle launch capability. The WSO Orbiter launch mass includes 30% dry mass growth contingency.
- While there are several subsystem mass estimates that seem low, the launch mass margins will allow for the increase in mass of these subsystems with minimum impact on the system mass contingency.
- The orbit is non-eclipsing, and this simplified the thermal control and the electrical power system.
- The simultaneous development of the full operations system in parallel with the other development phases suggest that the strong heritage component will allow most of the foreseen synchronization and duplication problems to be sorted out well in time.

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INTRODUCTION TO THE ASSESSMENT

The World Space Observatory (**WSO**) concept was discussed for the first time in the conclusions and recommendations of the 8th UN / ESA Workshop for Basic Space Science in the Developing Countries. (A / AC.105 / 723) The implementation of the WSO concept was determined to be most beneficial in the Ultraviolet spectra (110 nm < λ < 340 nm). This led to the advanced concept of a World Space Observatory in the Ultraviolet (***WSO/UV***) spectrum. The European Space Agency developed an Assessment Study for WSO / UV as described in ESA CDF-05(A).

The WSO / UV mission provides an excellent opportunity for planetary science return. Hence it is important to evaluate the capabilities of the mission described in CDF-05(A) for applicability to the studies of Planets and other Solar System bodies. Simultaneously it presents the opportunity to validate the original concept in the framework of a concurrent ESTEC- JPL design team. The process chosen was a APD Team assessment of the ESTEC design for WSO / UV as described in CDF-05(A).

The result of the APD Team assessment of the WSO / UV is the subject of this report and is consistent with the findings of the CDF-05(A).

WORLD SPACE OBSERVATORY SUMMARY

The World Space Observatory (WSO) mission was designed to provide an ultraviolet wavelength space telescope to the entire international community. Nations that traditionally have no access to space will have the opportunity to participate in the operation and use of this first-class space observatory. Educational and research opportunities in astronomy will be expanded beyond the traditional research nations.

A wide range of ultraviolet astronomy will be possible, from distant galactic and black hole observations to comet and atmosphere observations in our solar system. The WSO has an ultraviolet imager and spectrometer along with a visual camera. The mission will provide observations in a wavelength regime not adequately provided for in current or future space telescope missions.

The WSO delivers and supports a telescope and instruments to an orbit around the Earth-Sun LaGrange 2 point (L2). The nominal mission length will be 5 years, but consumables will be sized for a 10-year mission goal. The instruments will collect 32 Gbits of data per day and the spacecraft can store up to 64 Gbits of data. The telecommunications system must downlink this data within an 8-hour period each with a downlink data rate of 1.1 Mbps. The WSO spacecraft will be launched on a Russian Proton, whose launch vehicle capability for this study is 4796 kg. The current baseline design has a mass of 3465 kg. Launch date is July 1, 2006 into a 200-km parking orbit around Earth. From there the Proton upper stage boosts the spacecraft into an 800,000 km orbit around (L2).

APD TEAM ASSESSMENT OBJECTIVES

This assessment covers the "Assessment Study Report, WSO / UV Space Observatory" from May 2000. Everything from science objectives to detailed design was reviewed for feasibility, completeness, and risk. Requirements and trade-offs were reviewed and additions were suggested where appropriate. The results are presented within this document.

APD TEAM ASSESSMENT PROCESS

The members of the ESTEC Concurrent Design Facility supported the entire assessment process. During the Science Assessment, ESTEC provided detailed instrument information. ESTEC participated in the Advanced Projects Design Team concurrent design assessment acting as the customer and providing real-time elaboration of the ESTEC CDF concept design for WSO.

The APD Team assessment was funded by Michael Devirian, Manager, Origins & Fundamental Physics, and Stephen Wall, Leader, Center for Space Mission Architecture and Design (CSMAD), as a test case in collaborative engineering activities. CSMAD is a JPL Center of Excellence, established to ensure and maintain excellence in fields recognized by JPL as strategic to NASA goals.

Science Team Overview

Science Assessment Participants

The following representatives comprised the science assessment participants:

WSO/UV Science and Instrument Team Members	Affiliation
Willem Wamsteker, Study Manager	ESA / VILSPA / SCI-SA
Noah Brosch, Camera / Detectors	Tel Aviv Univ
Boris Shustov, Telescope	Inst of Astron, Moscow
AA Moisheev, Telescope Project	Lavochkin Corp, Moscow
N. Kappelmann, Spectrograph	Inst for Astronomy, Tubingen
K. Werner, Spectrograph	Inst for Astronomy, Tubingen
Hans Haubold, Project Scientific Affairs	UN-OOSA

JPL Science Members	Affiliation
Arthur Lane	JPL
Robert Nelson	JPL
Alejandro Soto	JPL
Robert West	JPL
ESTEC Concurrent Design	Facility Study Team

Science Assessment Process

JPL scientists reviewed and concurred with the WSO science objectives. Due to JPL's areas of expertise, an emphasis was placed on the benefits from WSO available to the planetary astronomy community. The WSO proposal members led by Willem Wamsteker provided instrument and spacecraft details. The accumulated science comments and suggestions were incorporated into the science portion of this assessment.

Team-X Overview

Study Participants

The following representatives comprised the study team:

JPL Team Members	Subsystem
Robert Oberto	Study Leader
Marie Deutsch	Study Coordinator
Alejandro Soto	Science
Terri Anderson	Ground Systems
Matt Johnson	Systems
Cesar Sepulveda	Instruments
Dick Cowley	Propulsion
Bob Kinsey (AC)	ACS
Joseph Smith	CDS
Paul Timmerman	Power
Bob Miyake	Thermal
Gerhard Klose	Structures
Anil Kantak	Telecom-System
Mary Boghosian	Telecom-Hardware
Alok Chatterjee	Mission

Advanced Projects Design Team Concurrent Design Assessment Approach

Advanced Projects Design Team implements a concurrent engineering approach to mission concept evaluation and pre-phase A design. Within a very short period of time, the APD Team can consider, implement, evaluate, and recommend acceptance/ rejection of numerous ideas, with a relatively high level of fidelity. The format of an APD Team vastly increases the efficiency of the design process, and makes the work performed for these small interplanetary mission studies possible. Figure 1 highlights the advantage of a concurrent engineering process.

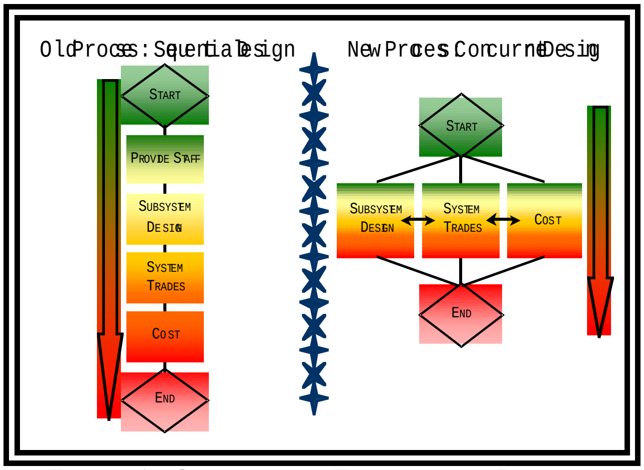


FIGURE 1 - CONCURRENT ENGINEERING PROCESS

The APD Team is comprised of experienced engineers from a range of subsystems working in parallel to develop and evaluate a spacecraft system-level design. Each subsystem engineer is an "expert" in his or her dedicated field (such as cost estimation, telecommunications hardware, mission design, programmatics, etc.) and brings considerable technical prowess to the team. This aids in rapid system-level trade evaluation since each team member can speak intelligently for his or her subsystem. Figure 2 shows how the APD Team is organized.

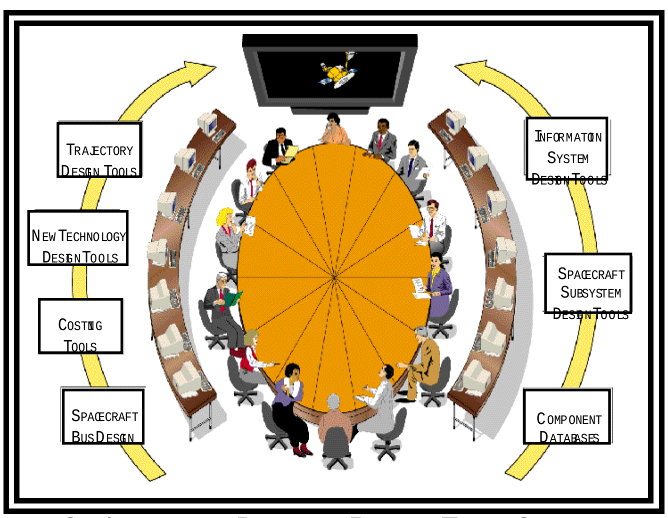


FIGURE 2 - ADVANCED PROJECTS DESIGN TEAM ORGANIZATION

Within the APD Team framework, the studies conducted during this investigation were performed following a standard Advanced Projects Design Team study flow—save a few exceptions. Typically, for interplanetary studies, single-string systems are found to be too risky (given the high cost). However, there are significant recent examples of single string interplanetary missions such as Mars Pathfinder and Deep Space 1. Due to the significantly lower cost associated with these micromissions, a higher level of risk was acceptable. Given this assumption, subsystem designers had more freedom to select "risky" solutions when considering trades. Details of such decisions are collated into a study report and delivered to the respective customer.

SCIENCE ASSESSMENT

SCIENTIFIC OBJECTIVES

Overview

The World Space Observatory will supply observation capabilities in an astrophysical wavelength domain that lacks sufficient coverage beyond 2005. The science objectives span a broad range of astronomical targets in the ultraviolet wavelength, including:

- Study the chemical evolution of galaxies and the cycling of interstellar matter.
- Study the re-ionization phase of the universe especially establishing the nature, location, and time of the Galaxy formation.
- Obtain spectroscopy and imaging in the ultraviolet.
- Discover and study Comets, Novae, Supernovae, Gamma-ray Bursters, OVVs and other unpredictable phenomena by providing a rapid response capability.
- Monitor atmospheric and ionospheric conditions on planets.
- Study long term characteristics of massive black holes.
- Study the history of star formation for 75% of the lifetime of the universe ($z \le 3$)
- Search for planet candidates.
- Study planetary objects, including the global atmospheric circulation and magnetospheric interaction for gas planets, atmospheric phenomena on Mars, composition and dynamics of comets and chemistry of processes on icy satellites .
- Study the complete stellar life cycles.

Through the science objectives, WSO will provide observations to help answer the fundamental astrophysical questions.

Assessment

The ultraviolet (UV) observation capability of the World Space Observatory (WSO) will provide a variety of opportunities for the planetary science community, ranging from independent studies of solar system planetary bodies to observation campaigns coordinated with active spacecraft missions. Planetary atmospheric scientists will benefit from UV spectra of planetary atmospheres and icy satellites surface chemistry, imaging and spectroscopy of the gas planets' aurora, and UV and visual imaging of the gas planets, Mars and Venus in methane absorption spectral bands. Finally, scientists will gain the ability to respond rapidly to new targets such as comets and gas planet cloud formations.

The WSO will complement the Hubble Space Telescope (HST) and GALEX capabilities in the UV regime including UV imaging and spectroscopy as well as high resolution visible imaging of planetary systems. As well, HST over-subscription will be mitigated by the existence of an observatory like WSO. WSO advantages over HST include longer continuous observation times and a less severe charged particle environment. The high resolution made available by the WSO will allow scientists to address important issues in molecular and atomic astrophysics which are beyond the capabilities of other instruments.

The ability to perform near real-time scientist-observatory interactions would augment the observation planning capabilities available to the scientist. The scientist would be included in the "control-loop" that drives the operation of each observation. With this attribute the observatory operates similar to a capable ground-based system.

The orbital position of the WSO spacecraft offers another opportunity for space physicists interested in studying the fields around the Earth. The inclusion of a Solar Radiometer onto the spacecraft could be an important and relatively simple addition which would contribute very much to the understanding of the stability of the integrated solar radiation output. Such an instrument could also act as a "health" monitor measuring the intensity of the particles that are degrading the spacecraft surfaces, especially the solar panels. Correctly designed, this possible instrument might not be a significant impact on the spacecraft mass and power margins. The WSO team should consider adding a space physics experiment to the science objectives.

Science Objectives' Correlation to Astronomy Priorities

Overview

In May 2000 the Space Studies Board of the National Research Council released a report on the "Astronomy and Astrophysics in the New Millennium" which explains a prioritized plan for astronomy and astrophysics research for the next decade. The key problems to address through research and observation are:

- Determining the large-scale properties of the universe: the amount and distribution of its matter and energy, its age, and the history of its expansion.
- Studying the dawn of the modern universe, when the first stars and galaxies formed.
- Understanding the formation and evolution of black holes of all sizes.
- Studying the formation of stars and their planetary systems, and the birth and evolution of giant terrestrial planets.
- Understanding how the astronomical environment affects Earth.

The list of supporting, future space telescopes designed to support these priorities lacked an ultraviolet telescope needed to answer some critical aspects of the problems stated above.

In 1994 the Committee on Planetary and Lunar Exploration, part of the Space Studies Board, released an "Integrated Strategy for the Planetary Sciences 1995-2010". The document identifies extrasolar planet searches, comet investigations, and gas planet studies as some of the primary priorities for the planetary science communities. Atmospheric and magnetospheric research are identified as physical processes requiring further study, especially with respect to Jupiter and the Jovian system.

Assessment

The WSO mission successfully addresses the priorities of the astronomy and planetary science community through its ultraviolet observation capabilities. Galaxy formation and evolution, black hole evolution, extrasolar planetary systems, and stellar evolution for 75% of the universe's lifetime are among the astronomy priorities that are captured within the WSO's science objectives. For planetary science, the WSO provides observations of planetary atmospheres and magnetospheres for a range of planets from Venus to Neptune, as well as observations of comets, icy satellite surfaces and extrasolar planetary systems.

Measurement Objectives

Overview

The WSO telescope has a 1.5 – 2.0 diameter mirror with one UV imaging and one UV spectrometry instrument. The UV imaging instrument will cover the 115 – 340 nm wavelength regime and provide images with a 0.1 to 0.3 arcseconds quality. There are two imaging detectors, one for maximum spatial resolution and one for maximum sensitivity. A visual detector is included for visual imaging to correlate with the ultraviolet observations. At the same time, the spectrograph will provide a 115 – 340 nm range with a spectral resolution of 5 – 8 x 10^4 with a point spread function < 0.3 arcseconds.

The UV imaging and spectrometry detectors are required to meet the science objective of ultraviolet imaging and spectral observations. The subarcsecond quality is designed to normal astronomy standards for "seeing" quality allowing for good resolution, high value observations.

To maintain the good optical qualities of the telescope, the pointing of the spacecraft is limited. The telescope cannot be pointed within 45 degrees of the sun and, due to non-articulating solar arrays, the telescope cannot be pointed in the anti-sun direction. Within these constraints a large amount of astronomy targets remain accessible to the WSO telescope.

Assessment

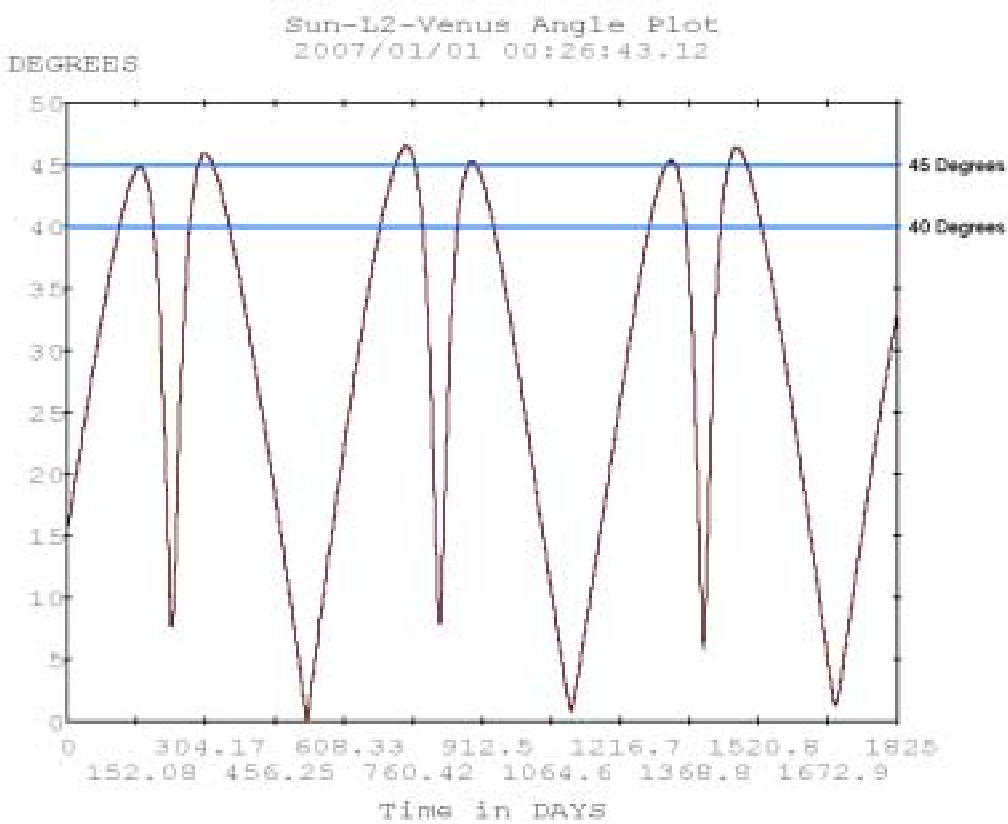
The telescope and sensor suite will provide good observation capabilities to the astronomy community and the planetary science community. The visual imager will be especially good for planetary atmosphere research due to the ability to match spectral data with visual images.

Planetary science observing opportunities on the WSO will require properly selected filters for the visual and ultraviolet cameras. For example, twelve filters for the visual camera and six filters for the ultraviolet camera provide a wide range of data for planetary science, especially if the filters are chosen to capture methane absorption. Possible UV filters include a short wave filter (below 140 nm) that transmits Lyman-α coupled with medium or wide pass filters (with a ~30 - 40 nm width) spaced throughout the 140 nm to 320 nm spectral regime. WFPC-2 and STIS filters are good examples.

Extending the wavelength regime to 1000 nm would allow for studies of the methane band and continuum spectral regions for giant planet and Titan studies at wavelengths between 800 nm and 1000 nm. An extension to 1000 nm also allows for the possible use of filters in this regime, for example 727 nm and 889 nm methane absorption filters paired with 750 nm and 935 nm continuum filters. (The preceding filter and wavelength suggestions are for illustrative purposes; actual selections would need to be determined by a WSO science group.)

Two pointing restrictions for the WSO limit the valuable observations available to planetary science. First, due to the fixed solar arrays in the current design, the WSO telescope cannot be pointed in the anti-sun direction due to the subsequent loss of power from the solar arrays. However, observations of planetary bodies in the anti-sun hemisphere are required for obtaining images and spectra at zero phase angles. Redesigning the solar arrays to allow for articulation would provide these valuable planetary science observations. This issue to be addressed in the phase A study.

The second pointing restriction requires a boresight/sun angle to be greater than 45 degrees. This requirement prevents all but very infrequent observations of Venus and the Venus' atmosphere. Modifying this requirement to 40 degrees allows observations of Venus for a few days before and after each of its quadrature points. Certainly, such a modification will have impacts on lighting and thermal design for the telescope, but accommodation of this modification allows an important planetary target to be included in the WSO mission. Figure 1 shows the Sun-L2-Venus angle for a five-year mission from the beginning of 2007. Note that the Venus observing time is considerably increased when the pointing restriction is lowered to 40 degrees.



INSTRUMENTS

Overview

The WSO mission instruments consist of the Russian supplied telescope (T-170) and the two sensors: an imager and a spectrometer.

The Russian built telescope is a Ritchey-Chretien type with a 1.7 m primary mirror and an angular resolution in the focal plane of 12.05 arcsec/mm.

Ritchey-Chretien telescope.

Focal Length	17 m
F/ number	10
Back focal length	750 mm
FOV	40 arcmin
RMS WFE	Lambda/30 @ 632 nm
Diameter Primary mirror	1715 mm
Diameter central hole Primary mirror	660 mm
Distance Primary mirror-Secondary mirror	3500 mm
Diameter secondary mirror	475 mm
Diameter central hole Secondary mirror	85 mm

The spectrometer, or High Resolution Double Echelle Spectrograph (HIRDES), comprises three wavelength channels:

- Vacuum UV Echelle Spectrograph 110-180 nm, R=60000
- UV Echelle Spectrograph 178-350 nm, R=48000
- Long Slit Spectrograph (0.08 mm x 5 mm), R=500

The spectrometer will consist of two electronic boxes and one high voltage power with an allocation of 141 kg and 150 W. The detector is a micro-channel plate detector with 1024 x 512 pixel dimensions. Each pixel is 35 square microns with 16 bits of data. Two modes of operation for the spectrometer: a) store/download the full three spectrum channels observing for 24 hours, or, b) store/download the full three spectrum channels observing for 10 seconds.

Four units with one channel each and wavelength ranges of 122 nm to 320 nm constitute the imager instrument. Two of the imagers have a 1 arcmin field of view (FOV) and an instantaneous field of view (IFOV) of 0.03 arcsec. The other two imagers have a 5 arcmin FOV and an IFOV of 0.15 arcsec. The detector is a micro-channel plate CsTe with a pixel size of 25 microns squared and 12 bits per pixel whose overall active size is 50 mm x 50 mm. Each imager is allocated 16.5 kg and 25 W. Two modes of operation for the imager: a) store/download only 100 pixels of 2 out of 4 imagers with a time resolution of 100 msec, or, b) store/download for each of the 4 imagers 1 imager per minute. (Total duration limited depending on maximum 1 – 1.5 Mbps for download.)

The fine guidance sensor has a wavelength range of 400 nm to 800 nm (visible), a CCD detector size of 1024 x 1024 x 16 bits with 24 microns squared per pixel. The overall active size is 25 mm x 25 mm. The FOV is 55.2 arcsec squared field extension while the IFOV is 0.054 arcsec. Three fine guidance sensors are arranged symmetrically around the optical axis with a distance of 20 mm from each detector to the optical axis.

The visual imager has a wavelength range of 400 nm to 800 nm (visible), a CCD detector size of 2048 x 2048 x 12 bits with 24 microns squared per pixel. The overall active size is 50 mm x 50 mm. The FOV is 4 arcmin squared field extension while the IFOV is 0.12 arcsec.

The data rates for the sensors are as follows:

Minimum data rates for:	imagers =32 Kbps	HIRDES = 583 bps
Maximum data rates for:	imagers = 4.48 Mbps	HIRDES = 5.03 Mbps

Assessment

A few concerns exist for the instrument design. Since total power in the instrument spreadsheet is carried as a straight sum across all instruments, the power requirements may be overstated by assuming that *all* of the instruments are on at the same time. Although significant margin exists for power, the assumption that all instruments are always on should be reconsidered. In later studies, the various power modes need to be examined; no guidance was provided for Safe, Cruise, and Launch modes. Finally, even though the proposal mentions the brief radiation exposure to the Van Allen Belts during launch, nothing was stated regarding shielding. Shielding requirements for the instruments and detectors need to be investigated.

The one area most remiss in its potential corrective capability is the issue of truly simultaneous measurements. In some cases, possibly almost all the cases, the issue is not putting two spectrometers on the same point simultaneously (which could be desired), but rather getting adequate proof of the domain from which spectroscopy data are acquired -- especially in spatially heterogeneous or temporally evolving domains (AND a few situations have BOTH). What is not present is a way to direct via either a fixed or variable ratio beamsplit between the spectrograph entrance slit(s) and the imaging system observable field.

In order to accommodate studies of atmospheric chemistry of planets and satellites a removable filter on the camera is needed to allow the Lyman-α observations of solar system bodies.

MISSION ANALYSIS

MISSION DESIGN

Overview

The WSO mission destination is an orbit around the L2 point in the Sun-Earth system, which is about 1.8 million kilometers from Earth away from Sun. The Sun, Earth, and L2 point make a line with the Earth at the center, and the Earth and L2 point move in fixed orientation around the sun. The orbit is 800,000 km about the L2 point, and is a stable orbit requiring small ΔV for maintenance. This orbit allows for operation of the spacecraft beyond Earth orbit and outside of any Earth eclipse of the sun. The orbit about the L2 point is semi-stable requiring a modest amount of station keeping throughout the mission lifetime.

The launch system selected for the mission is the Proton/Block-DM system. The launch capability for a C_3 of 0.5 is about 7000 kg, but there may be structural load limitations on the payload mass unless modifications are made. The launch sequence is a Proton Block-DM launch from Baikonour in Kazakhstan. The first phase launches the WSO and the Block DM into a 200-km parking orbit. At the appropriate time and location the Block DM upper stage is fired to inject the WSO in a direct transfer orbit to the L2 region with a C_3 of –0.5 km2/s2.

The total ΔV for the WSO is about 95 m/s out of which about 75 m/s is consumed for trajectory corrections and orbit insertion. The remaining 20 m/s is for orbit maintenance for a 10 year mission lifetime, as opposed to the 10 m/s assumed for a 5 year missions.

Assessment

The WSO mission has an excellent mission design approach and rationale. The L2 orbit provides a simplified operation since there are no eclipses limiting the times of observation. As well, the thermal requirements are minimized since the thermal environment will remain essentially constant throughout the mission lifetime. The selection of this launch system is a suitable one due to the enormous flight history and success rate. The Proton has flown since 1965 and has a 98 percent success rate, the highest for any heavy lift vehicle.

SPACECRAFT SYSTEM ASSESSMENT

SYSTEMS

Overview

The principal mission element is the World Space Observatory spacecraft. The primary design drivers are the tight requirements placed on the attitude control system. The control requirement is 0.066 arcsec 3σ over a 24-hour period, and knowledge is needed to an accuracy of 0.0066 arcsec 3σ. The instruments will collect 32Gbits of data per day. The telecommunications system must downlink this data within an 8-hour period each day. This results in a necessary downlink data rate of 1.1 Mbps. This system design easily accommodates the data to be downlinked each day with the excess ground station capacity. The storage requirement was driven by the desire to carry two days worth of data in case a telecommunications pass is missed.

The launch vehicle for this mission is a Russian Proton. The launch vehicle capability assumed for this study is 4796 kg. The current baseline design has a mass of 3465 kg, including 30% mass contingency on the instrument and 38% mass contingency on the spacecraft bus. This results in a launch margin of 1331 kg (27.8%). Launch date is July 1, 2006 into a 200-km parking orbit around Earth. From there the Proton upper stage boosts the spacecraft into a 800,000 km orbit around L2. This orbit was chosen because it is thermally stable and has no eclipses. This orbit also requires little maintenance, simplifying operations. The mission requirement is a five-year mission life, with a goal for 10 years. Full redundancy is assumed for this mission, which is consistent for a mission of this length and class.

Nominal operations are divided into two power modes, science, and re-pointing. There is no separate telecommunications mode since the design allowed the power amplifier to remain fully operational at all times, but the telecommunications system will use passes of 8 hours per day to download the required 32 Gbits of data per day. During a normal science observation mode the instruments use a total of 456 W of power with 300 W used by the telescope. The repointing mode can occur up to ten times a day with fine adjustments and moving target tracking. The instruments are also assumed to be at full power for this power mode.

There are limitations placed on the WSO spacecraft with regards to the spacecraft-sun angle. In order to keep the telescope structure and instruments warm enough to avoid noticeable optical distortions the angle needs to be kept between 45° and 135°. The telescope's baffle and thermal constraints prevent the spacecraft from pointing closer to the sun than 45°, which means that the entire celestial sphere is not accessible at any given moment. However, during the course of the year the telescope has access to the entire sphere twice. The orbit is designed so that the telescope will not go through an eclipse during the ten year desired mission life, thereby helping simplify the thermal design a great deal.

SYSTEMS TABLE 1: STUDY GUIDELINES AND ASSUMPTIONS

Team X Study Guidelines

World Space Observatory Red Team 10-00
WSO Orbiter

	Programmatics/Mission
Customer	Steve Wall, Mike Devirian
Study Lead	Robert Oberto
Mission	World Space Observatory Red Team 10-00
Target Body	L2
Orbit	800000 km L2 orbit
Science/Instruments	Telescope, Optical Bench, Spectrometer, UV-Imager, VIS-Imager, Fine Guidance Sensor
Potential Inst-S/C Commonality	Fine Guidance Sensor
Desired Launch Vehicle	Proton
Launch Date	Jul-06
Mission Duration	5 years 2 months, 2 month transfer, 5 years observational science, 10 year goal
Mission Class	A/B
Technology Cutoff	2002
Minimum TRL	6
	Spacecraft
Redundancy	Full
Stabilization	3-axis
Heritage	XMM (Newton)
L/V Capability, kg	4796 to a C3 of -0.5
Radiation Total Dose	15 krad behind 100 mils alum., no RDM added
GPS	No
Drag Makeup, m/s	N/A
Post-Launch Delta-V, m/s	85
P/L Mass, kg	1673.5 total, 1444 telescope, 20 optical bench, 141 spectrometer, 51.5 UV-Imager, 17 VIS-Imager
P/L Power, W	456 W, 300 telescope, 125 spectrometer, 25 UV Imager, 6 VIS Imager
P/L Data Rate, kb/s	9500 peak
P/L Pointing, arcsec	0.066 arcseconds 3-sigma over 24 hours (relative pointing) control, 0.0066 arcsec knowledge 3 sigma
Tracking Network	ESA Stations (Perth, Kourou, Vilspa) & Maspalomas

Assessment

Advanced Projects Design Team used the equipment list provided to us by ESA, and no separate design was done for this study. However, changes were made from the existing proposal. In the proposal there were allocations made for the Fine Guidance Sensors required for telescope pointing in both the ACS system and within the payload itself. This design carries them only in the ACS system. Also, the structures subsystem carries the allocation for the solar array substrate not the power subsystem as is done in the ESA proposal. Structures also carries all system cabling as a line item shown on the system sheet on the following page.

Overall, this is a very robust design with adequate redundancy for a mission of this length. There is also plenty of contingency placed against the dry mass. This contingency, coupled with the large amount of launch mass margin, helps alleviate concerns of slightly low mass at the subsystem level. However, the subsystem mass issues are still documented where appropriate in the subsystem reports and listed with the system issues and concerns.

SYSTEMS TABLE 2: MASS AND POWER SUMMARY

World Space Observatory Red Team 10-00

WSO Orbiter

SYSTEMS WORKSHEET

Analyst: Matt Johnson

Start Date: 10/24/00 Directory: Team X-Cost:ACTIVE STUDIES:OTHER STUDIES:WSO RTR 10-00:World Space Obs RTR_EXCEL:WSO Orbiter1

Stabilization - cruise	**3-Axis**		Pointing Direction - cruise	**Sun**			
Stabilization - science	**3-Axis**		Pointing Direction - science	**TBD**	Mission Duration	**5.0**	years
					Max probe sun distance	**1**	AU
Pointing Control	**0.07**	arcsec	Radiation Total Dose, krad	**15**	Inst. Data Rate	**9500**	kb/s
Pointing Knowledge	**0.01**	arcsec	Science BER	**1.00E-06**	Data Storage	**64 Gb**	Mb
Pointing Stability	**0.54**	arcsec/sec	Redundancy	**Full**			
Determined by:	**Inst.**				Maximum Link Distance	**1.8E+06**	km
			Technology Cutoff	**2002**	Return Data Rate	**1.1 Mbps**	kb/s

	Mass Fraction	Mass ROM	Mass (kg)	Mode 1 Power (W) *Science*	Mode 2 Power (W) *Re-pointing*	Mode 3 Power (W) *Safe*	Mode 4 Power (W) *Cruise*	Mode 5 Power (W) *Launch*	NASA TRL *Today*	Last Updated
Payload										
Instruments	68.8%	1673.5	1673.5	456.0	206.0	206.0	206.0	206.0	5	10/25/00 8:27
Payload Total	68.8%	1673.5	1673.5	456.0	206.0	206.0	206.0	206.0		
Bus										
Attitude Control	4.9%	118.9	118.9	246.8	250.8	26.3	232.8	59.5	6	10/24/00 15:27
Command & Data	2.1%	50.9	50.9	133.6	133.6	113.6	113.6	113.6	7	10/24/00 11:36
Power	3.0%	73.1	73.1	145.8	113.9	104.5	108.7	80.2	6	10/24/00 14:46
Propulsion	0.8%	19.7	19.7	1.5	1.5	25.5	1.5	25.5	6	10/24/00 15:35
Structure	14.5%	352.1	352.1	0.0	0.0	0.0	0.0	0.0	6	10/24/00 11:33
S/C Adapter	2.0%	48.3	48.3							
Cabling	1.4%	32.9	32.9							10/24/00 11:33
Telecomm	1.2%	28.9	28.9	162.3	162.3	162.3	162.3	162.3	9	10/24/00 10:50
Thermal	1.5%	35.9	35.9	101.7	101.7	270.0	120.0	50.0	6	10/24/00 11:18
Bus Total		760.7	760.7	791.7	763.7	702.2	738.8	491.1		
Spacecraft Total (Dry)		2434.2	2434.2	1247.7	969.7	908.2	944.8	697.1		
Mass/Power Contingency		794.3	794.3	374.3	290.9	272.5	283.4	209.1		
Spacecraft with Contingency		**3228.5**	**3228.5**	**1622.0**	**1260.6**	**1180.6**	**1228.3**	**906.2**		
Propellant & Pressurant	4.0%	136.2	136.2	For S/C mass =	3356		Delta-V1	85	m/s	10/24/00 15:35
Spacecraft Total (Wet)		**3364.7**	**3364.7**				Contingencies			
L/V Adapter		100.0	100.0				Mass	Power		10/24/00 11:33
						Instruments	30%	30%		
Launch Mass		**3464.7**	**3464.7**			Other	N/A	N/A		
						S/C, dry	38%	30%		
Launch Vehicle Capability		4796.0	**4796.0**	**Proton**		Launch C3	-0.5		9	10/24/00 11:24
						Fairing type	standard			
						Fairing dia., m	?			
Launch Vehicle Margin		**1331.3**	**1331.3**	**27.8%**						

ATTITUDE AND CONTROL SYSTEM (ACS)

General Overview

A summary of the WSO attitude and control system:

- Initial stellar inertial attitude determination is based on precision star trackers that provide both attitude and rate information.
- The Solid State rate Measurement Unit (IMU) is used for fault detection and correction only.
- Sun sensors are used with the star tracker in a sun-pointing mode and with the IMU in an emergency safe mode.
- Fine guidance sensors provide a highly stable attitude reference once they have acquired guide stars. The precision star trackers provide attitude information for star acquisition.
- Precisely balanced reaction wheels are used to provide torque for fine pointing control. Hydrazine thrusters are used to de-saturate the reaction wheels.
- A high gain antenna is attached to the spacecraft bus through a two-degree of freedom gimbal.
- Two large (10.5 square meter) solar arrays are fixed to the spacecraft bus in a symmetric configuration.

General Assessment

Overall, the proposed ACS design appears to have a fair chance of success. While it does not require new technology, it does require the development of a custom, high-precision fine guidance sensor and an image reconstruction algorithm that has not been described in the proposal. Given these tasks, there would be some level of risk but the conceptual design approach described in the proposal appears to be sound. During Phase A studies, the risk can be examined in greater detail and plans developed to minimize the risk.

Approach to Achieving High Resolution

Overview/Assessment

According to the proposal, the telescope optics are such that a point source will produce an Airy spot on the focal plane with a diameter of about 0.2 arcsec. The high-resolution imager has a pixel size of 0.03 by 0.03 arcsec, so the spot would be about 7 pixels across. The preliminary error budget included in the proposal suggests that 2-sigma uncompensated jitter will have a magnitude of 0.034 arcsec, or 1 pixel. In that case, the spot on the focal plane due to a point source will be smeared so that its effective diameter is about 9 pixels. This is acceptable image quality, but less smear would be desirable.

On-board compensation and ground-based image reconstruction are proposed in order to reduce smear. This strategy makes use of compensation and reconstruction techniques rather than requiring structural modifications or improved sensor performance. Its viability depends on the reconstruction approach used and should be reviewed in detail when provided.

Fine Guidance Sensor Development

Overview/Assessment

In the proposal, the fine guidance sensor (FGS) has been identified as a development risk area. At present, the FGS's are an engineering design only. The proposed plan is to custom build a highly precise sensor using commercially available hardware components and software. An off-the-shelf (OTS) star tracker will be stripped down to remove its optics, but retain the focal plane with CCD array. The stripped down star tracker will be mounted on the telescope optical bench so that it can make use of the telescope optics. Modified OTS star tracker software will be used to do centroiding for a number of stars at a time.

This design approach is straightforward and makes good sense. However, in practice the process will be non-trivial because the device must be integrated onto and tested on the optical bench and the final product must be highly accurate. While the individual pieces for the FGS will be previously flight-proven, the device as a whole will be a new sensor system.

The proposed schedule allows 2 or 3 years for device integration and testing, system-level integration and testing, and verification of FGS performance capabilities. It is believable that the schedule can be met because this new sensor system requires only component modifications as opposed to new sensor development. However, the integration, testing, and verification process is complex, so there is some risk of delays. Sensor system-level testing and verification will need to occur after the telescope, optical bench, and at least one FGS are fully integrated. In the unlikely event of delays during this phase, there could be significant cost implications for the project.

Autonomous Star Tracker Development

Overview/Assessment

The baseline for ACS includes an autonomous star tracker (AST) with SAC-C heritage and built by the Italian company, Alenia. The AST will have three heads and provide 3-axis attitude knowledge and rate information without the possibility of blinding. The proposal states that the AST is not yet flight proven but is expected to be within two years. SAC-C is due to be launched next month. It carries a 1-head version of the star tracker. The proposed 3- head version will have somewhat greater complexity: It is expected to carry its own star catalog, auto-acquire and track a large number of stars, do centroiding for each star, and provide a single attitude quaternion and a rate vector in an inertial reference frame. It was not clear during the study that a 3-head version of the star tracker will be flight-proven within two years.

The ACS equipment list allocates 19.1 kg and a peak power of 36.5 Watts per AST unit. There is a question as to why the baseline includes such a massive star tracker that has not been flight-proven, when there are other choices for flight-proven star trackers. The actual mass and power may be significantly lower when the AST is built. On the other hand, there are a number of choices for OTS star trackers with comparable accuracy, and a set of 3 would likely have mass and power lower than that allocated for the AST. The proposal is to use alternative star trackers if the baseline AST is not sufficiently flight-proven in time to meet the schedule.

Relative Pointing Control (Pointing Stability)

Overview/Assessment

The required “pointing stability” for the telescope is specified in the proposal as ±0.05 arcsec, 1 sigma, for pitch and yaw (the telescope boresight) over 24 hours. The specified requirement for roll (twist about the boresight) is ±0.5 arcsec, 1 sigma, over 24 hours. Advanced Projects Design Team uses the term “pointing stability” to refer to stability over a fraction of a second. In this report, the requirement for pointing control over 24 hours will be referred to as a relative pointing control requirement.

Approach to Meet the Yaw and Pitch Pointing Requirement

Overview

The high-resolution imager has a pixel size of 0.03 by 0.03 arcsec. The specified 1-sigma relative pointing control requirement would allow smear of ±1.7 pixels for the imager. The customer noted during this study that the relative pointing control requirement is more of an estimate of what can be achieved than a requirement. Ideally smear would be limited to a fraction of a pixel. The preliminary error budget in the proposal shows a pointing error of 0.022 arcsec, 1 sigma, for pitch or yaw. The customer noted during this study that the preliminary error budget includes some values that represent 1-sigma and some that represent 3-sigma errors. In particular, the reaction wheel vibration and noise contributions (from SOHO and XMM flight experience) are 3-sigma numbers. After correcting the error budget to use 1-sigma numbers for wheel vibration and noise, the pointing error is 0.017 arcsec, 1 sigma. The 3-sigma pointing error corresponds to smear of ±1.7 pixels, which is higher than the ideal of a fraction of a pixel. In contrast, large margins are expected in a preliminary error budget so that unforeseen errors do not become showstoppers later in development.

The proposed approach is to compensate for smear on-board using FGS-sensed star motion or to downlink time-tagged photons and do ground-based reconstruction of the image. The data rates for the HIRDES have been set to allow time-tagging. The FGS will be sampled at 10 Hz, so it could potentially be used to compensate for low frequency jitter up to a few Hz. The baseline approach assumes that the effects of higher frequency jitter can be mitigated using ground-based image reconstruction. It is assumed that the RWAs will be successfully balanced to reduce vibrations to very low levels even at higher frequencies. The RWAs are expected to require unloading only every 7 to 8 weeks. Unloading more often can also be done to keep wheel vibration frequencies low and further reduce vibration levels.

Assessment

The assumption that the wheels can be precisely balanced seems reasonable since it is based on flight experience of similar wheels used for SOHO and XMM. The proposed bus structure does not include vibration isolation for the wheels, and there will be limits to how well the effects of high frequency jitter can be mitigated. So, mission success may depend on precise high-speed balancing of the wheels. Consequently, there will need to be careful monitoring of the balancing process and thorough testing to verify acceptable wheel characteristics. Alternatively, the possibility of adding vibration isolation is a subject for Phase A studies. If vibration isolation were implemented, then on-board compensation using star motion sensed by the FGS and ground-based reconstruction using time-tagged photons would provide added assurance of being able to achieve very high quality imaging.

Using sensed star motion in the FGS field of view to compensate for low frequency jitter is a sound idea. Actual on-board implementation will be non-trivial. Thorough ground-based testing of the fully integrated telescope/optical bench/FGS and the flight code for the compensation algorithm will be needed to verify correct implementation resulting in adequate performance.

Ground-based reconstruction of the image to mitigate higher frequency jitter is a reasonable concept. However, no detail is provided in the proposal regarding the reconstruction approach. Since the preliminary error budget shows uncompensated jitter well in excess of ideal for high-resolution imagery, either the reconstruction approach must be highly effective or the reaction wheels precisely balanced or both. The effectiveness of the reconstruction approach needs to be quantified in order to determine what level of higher frequency jitter can be tolerated, which determines how well the wheels must be balanced. Phase A studies will be needed to quantify the effectiveness of the reconstruction approach, in order to determine the quality of high-resolution imagery that is achievable for this mission.

Driver for the Roll Pointing Requirement

Overview/Assessment

The specified relative pointing requirement (pointing stability) for roll is ±0.5 arcsec, 1 sigma, over 24 hours. The driver is that a star image must past through a 1 arcsec slit in the spectrometer instrument. Even with several arcseconds of star “misalignment” from the boresight, 1-sigma errors in excess of 0.5 arcsec are acceptable. The preliminary error budget in the proposal shows an RSS value for roll of 0.542 arcsec, 1 sigma.

Relative Pointing Knowledge

Overview/Assessment

Pointing knowledge requirements are not system drivers. A suitable requirement for relative pointing knowledge is ±0.005 arcsec, 1 sigma, per axis for pitch and yaw, which is one-tenth of the relative pointing control requirement. A suitable requirement for roll axis relative pointing knowledge is ±0.05 arcsec, 1 sigma. The fine guidance sensors (FGS’s) are expected to provide the means to achieve this level of knowledge. The pixel size for the CCD array used in the FGS is 0.054 by 0.054 arcsec. The PSF of a star on the focal plane of the FGS is 3 pixels wide. Centroiding algorithms are conservatively assumed to be capable of determining star position to within 1/25th of a pixel, about 0.0022 arcsec, 3 sigma.

Pointing Stability

Overview/Assessment

In this report, pointing stability refers to stability over a fraction of a second. It is the stability needed to limit smear on the focal plane of the FGS to less than 1 pixel over 0.1 seconds. This assumes that the FGS will be sampled at 10 Hz. It is further assumed that the PSF for a star is 3 pixels wide and that smear less than 1/3rd of the PSF is acceptable. A suitable pitch and yaw requirement is pointing stability to within ±0.18 arcsec/sec, 1 sigma. A suitable roll stability requirement is to within ±1.8 arcsec/sec, 1 sigma.

Absolute Pointing Control

Overview/Assessment

Absolute pointing control requirements are not system drivers, and consequently, there is no preliminary error budget for absolute pointing control. The proposed spacecraft does in fact have absolute pointing control requirements for the fine guidance sensors, for the spectrometer, and for the high gain antenna (HGA). The HGA requires absolute pointing control to within 0.2 to 0.3 degrees. Given that the FGS’s and spectrometer require much tighter pointing control, HGA pointing is not a driver.

Absolute Pointing for the Fine Guidance Sensors

Overview/Assessment

During the study, the customer specified an absolute pointing requirement for the FGS's in order to acquire and track guide stars. The field of view (FOV) of each FGS is 55 arcsec. The absolute pointing control requirement in the pitch and yaw axes is one third of the FGS FOV, which is ±18 arcsec, 3 sigma. A suitable but somewhat arbitrary absolute pointing requirement for roll is ±360 arcsec, 3 sigma. With the FGS's located 250 arcsec off of the boresight of the telescope and the roll axis aligned with the boresight, a roll error of 360 arcsec would lead to only 0.4 arcsec of pointing error at each FGS, which is acceptably small.

Absolute Pointing for the Spectrometer

Overview/Assessment

The spectrometer has three entrance slits. The customer stated during the study that each slit has a total width of 1 arcsec. The proposal says that during operation, the spacecraft will be pointed with micro-slews such that the telescope PSF of a point source fills one slit at a time. The telescope airy spot is just about 0.5 arcsec diameter at the spectrometer slits. So, to position a spot within a slit will require a high degree of absolute pointing control accuracy in pitch and yaw. A suitable requirement is ±0.1 arcsec, 3 sigma. The 24-hour relative pointing control requirement for pitch and yaw is ±0.15 arcsec, 3 sigma. The sum of these absolute and relative pointing control errors is ±0.25 arcsec, 3 sigma. So, a target star would remain within a slit for up to 24 hours.

In regards to absolute pointing control to within ±0.1 arcsec, 3 sigma, it is assumed that there will be ample settling time after micro-slews to allow slew-induced vibrations to damp out. The expected FGS knowledge capability (relative to guide stars) is 0.0022 arcsec, 3 sigma. This ought to provide adequate attitude knowledge to achieve the required absolute pointing control after a micro-slew.

The operational approach to perform micro-slews assumes that the FGS's have acquired and are tracking guide stars close to the target star for the spectrometer. Each FGS has a 55 arcsec FOV (minus some margin) within which to do the micro-slew. A micro-slew less than 25 arcsec would be performed so that the target star PSF fills the desired slit.

Absolute Pointing Knowledge

Overview/Assessment

Absolute pointing knowledge requirements are not system drivers. These pertain to pointing control needed for the FGS's to acquire and track guide stars. In this case, the pointing knowledge will come from star trackers located on the bus. A suitable choice would be half of the absolute pointing control requirement for the FGS's. This would be ±9 arcsec, 3 sigma, in pitch and yaw, and ±180 arcsec, 3 sigma, in roll. The star tracker in the ACS baseline for the mission is expected to absolute pointing knowledge of 2 to 4 arcsec, 1 sigma. This represents raw star tracker performance, i.e., performance before calibration. The particular device chosen is a 3-axis star tracker that will provide 2 to 4 arcsec accuracy in all three axes. There are alignment errors and jitter to account for in addition to star tracker accuracy. However, it can be expected that star tracker errors will be reduced somewhat via calibration, so the required pointing knowledge does appear to be achievable.

Tracking Control

Overview/Assessment

The telescope will be required to track near objects moving at up to 0.2 arcsec/sec. This value is comparable to the situation with Comet Hyuatake which moved at .16 arcsec/sec at a distance of 33×10^6 km from Earth. The relative pointing control, relative pointing knowledge, and pointing stability requirements above will apply during tracking. There is an open question as to what level of jitter will be present during tracking. At this preliminary stage, it is not expected to have a finite element model for the system and there are no quantitative predictions for the jitter level.

The baseline approach will be to accelerate smoothly to the required tracking rate, allowing 1 to 2 minutes for settling before the start of observations. On the ground, an open-loop torque profile will be determined that will bring the spacecraft to the desired rate with a small error angle offset. The torque profile generation will require high fidelity modeling of the target object and spacecraft trajectories. After allowing for settling, the constant rate and a constant error offset will be maintained during the observation interval. The controller bandwidth is expected to be on the order of 2 Hz. A first flexible mode is expected to be around 0.3 Hz and a second around 1.8 Hz. To insure controller stability, a notch type filter will be used to reduce sensitivity to the first mode.

Slew Maneuvers

Overview/Assessment

The telescope may be required to slew up to 10 times per day over the life of the mission. These slews will range from very small angles to more than 40 degrees at a time. The proposal includes a table of estimated minimum slew times. However, this table is based on using the maximum reaction wheel capability and does not allow for external momentum buildup that would limit the maximum achievable slew rate. Neither does the table allow for settling time at the end of slews. The expectation is that slew maneuvers will induce solar array vibrations that take at least 1 to 2 minutes to damp out. The allowable slew times still need to be specified based on a realistic angular momentum budget and accounting for required settling times. As noted under Trade Considerations, this is a subject for Phase A studies.

Trade Considerations

Reaction Wheel Unloading Frequency versus Slew Times

The proposal includes an equipment list for ACS that shows 9.1 kg per reaction wheel. Each wheel has a momentum storage capability of around 40 Nms. There are four reaction wheels in a tetrahedron. One unit is to be kept in cold redundancy until needed. The wheels are to be supplied by MMS and are a heritage design from XMM. The plan is to dynamically and statically balance the to minimize the vibration they induce.

The expectation is that wheel unloading will only be needed every 2 months or so. A Advanced Projects Design Team preliminary sizing analysis shows that 42 Nms of angular momentum would be accumulated in 50 days due to solar torque, assuming a 1-inch offset between the center of pressure (CP) and the center of mass (CM). This is based on solar arrays with a total of 21 square meters of surface area facing directly towards the sun. The total fuel that would be required for momentum unloading over 10 years is less than 0.5 kg. The assumed one inch offset between the CP and CM seems a bit optimistic for a structure this large. However, if the offset is larger, the fallback is to unload the excess momentum more often.

There will be up to 10 slews per day ranging from very small angles to more than 40 degrees. The table of minimum slew times in the proposal is based only on maximum reaction wheel capability. It does not allow for storage of momentum due to external torques, nor does it include any momentum margin, and it does not include settling time to allow for slew-induced vibrations to damp out. The customer stated during the study that the plan is to allow 1 to 2 minutes for settling time following each slew.

Taking all factors into account, the wheels would appear to be undersized. Increasing the wheel size would lead to some reduction in the time available for observations, given that there will be up to 10 slews per day. Or, the wheels could be unloaded more often to insure that very little capability is used for storing accumulated momentum due to external torques. Trading off increased slew times versus more frequent wheel unloading is a subject for Phase A studies.

COMMAND AND DATA SYSTEM

Overview

The WSO spacecraft data system will perform the command and data handling functions for the spacecraft. The Integrated Control and Data System (ICDS) will also connect to the attitude determination and control sensors and actuators and will run the attitude determination and control software. Science data storage and formatting will also be part of the data system. Science data processing will not be required onboard this spacecraft. The spacecraft will supply power to the data system.

The Attitude Control System controls and monitors various attitude reference devices, determines the spacecraft attitude, and issues commands to control the spacecraft attitude. The software for the ACS will be located in the ICDS processor, and will be integrated into the ICDS processor software load.

The ICDS is required to perform many critical spacecraft functions. Several examples are listed here:

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

The ICDS controller software will use a commercially available real-time operating system, and will be programmable in a high level language, such as C or C++. The ICDS processor will host the software for the spacecraft, including the ACS. If a subsystem needs to have the controller perform specific functions, then the subsystem team will provide the software to the ICDS team.

RELEVANT MISSION PARAMETERS

Launch	July 2006
Primary Mission Duration	5 Years and 2 Months
Additional Extended Mission	4 Years and 10 Months
Redundancy Required	Full
Technology Cutoff Date	2002
Phase C/D Duration	36 months
Telecom Uplink Rate	4 Kbps
Telecom Downlink Rate	1.5 Mbps
Number of Instruments	4
Science Data Input Rate	9.5 Mbps possible, 5 Mbps max planned
Science Data Processing Rqmts: Compression Required	None
Science Data Volume (Memory)	64 Gbits
Mass Limitations	62.6 Kg allocation
Power Limitations	139.9 W allocation
Radiation (Total Ionizing Dose)	15 Krads
Power Source	Solar
Mission Class	A/B
Parts Class	Commercial & Mil-Std-883B screened (See References 1, 2, and 3)

MISSION SPECIFIC CONTROLLER INTERFACES

Attitude Determination	Sun Sensors (TBD)
	Star Trackers (TBD)
	IRUs (TBD)
	Magnetometers (TBD)
Attitude Control	Reaction Wheels (TBD)
	Torque Rods (TBD)
Science	Instruments: 4
Mass Memory, per String	64 Gbits (total, internally redundant)
Telecommunications	Uplink: 4 Kbps
	Downlink: 1.5 Mbps

The ICDS team will coordinate the software interfaces so that the capabilities of the controller (MIPS, memory, scheduling, etc.) are not exceeded. The ICDS team will also integrate the delivered software with the other software elements before spacecraft integration.

Instrument and Spacecraft subsystem elements will perform the functions defined in the following table. All phases of the science data flow within the spacecraft are covered.

FUNCTIONAL ASSIGNMENTS TO SUBSYSTEMS

Functions	**Instruments**	**ICDS**	**Telecom**
Raw Science Data Gathering	X		
Hamming Code/Interleaving	N/A	N/A	N/A
Data Correlation	X	X	
Raw Science Data Transfer		X	
Data Compression	N/A	N/A	N/A
Data Storage		X	
CCSDS Packet Headers		X	
CCSDS Frame Headers		X	
Reed Solomon Encoding		TBD	TBD
Convolution Encoding			X
Downlink Transmit			X

Note 1: Reed Solomon encoding hardware may be implemented in the CDS or the Telecommunications subsystems.

The data system will collect science and SOH data during observations by the WSO instruments. There will be no compression of the science data by the data system. The spectrograph does a considerable amount of data processing internally and is accounted for in the data rates given. Calculations on the data volume generated by each of these instruments can be found in the instruments/payload section of this report.

Team-X selected example ICDS element hardware based on the proposal selections and Team-X experience. For the Team-X review, the available selection of ICDS elements and boards is limited to those devices currently in the Team-X database. In most cases, the selection of devices is very limited, especially for the types of elements commonly used in Europe. Most of the devices selected for this study where from the Mars Sample Return 2003 study that was recently conducted at JPL. While these elements are not presently under development, they were used as placeholders that should be similar to the devices that could be finally selected, and they are all VME based. This was considered to be a reasonable approach because the WSO mission needs to use devices from a number of countries across the world.

Converter Module: The MSR-03 power converter module was selected for this device. While the exact requirements for the Converter Module is not know, this device should be consistent with the requirements of the other elements that were selected.

P/L or MM Interface Module: The MSR-03 I/F module was selected to represent this interface module. This module was intended to provide one bus interface and 32 analog input channels. A third MSR-03 I/F module is used to interface to the ACS sensors and actuators. This will provide the following I/O for the ICDS (half): 1 Low-Speed Bus, 1 High-Speed Bus, 1 ACS device bus, and 96 analog inputs.

Reconfiguration Module: Nothing is known about this module, and what its functions include. The MSR-03 Module I/F card was chosen to represent this element, since it probably performs similar functions.

Telecommand Module: The MSR-03 Uplink/Downlink card was chosen to represent the WSO Telecommand and Telemetry Modules, since this device performs the same functions.

Telemetry Module: The MSR-03 Uplink/Downlink card was chosen to represent the WSO Telecommand and Telemetry Modules, since this device performs the same functions. The quantity of zero (0) is shown in the block diagram for the Telemetry Module, because the function can be performed by the Telecommand Module.

Mass Memory Module: The requirement is for 2 Gbits (256 Mbyte) of memory for each half of the system. The Lockheed Martin Mass Memory card is a VME card with 8 Gbit (1 Gbyte) of memory. This card is used to meet the ICDS memory requirements.

Mass Memory: There is a requirement for 64 Gbit of data storage for science data for the WSO mission. A 64 Gbit mass memory device from SEAKER was selected to meet this requirement. While this device is still in development, there is a back-up device from Odetics. The Odetics device is also under development, but it was not selected because it consumed more power.

Low-Speed Bus: A low-speed bus is needed for this mission for the movement of spacecraft command and control data around the vehicle. A low-speed bus was assumed for this purpose in the WSO study, and this is a very reasonable approach. The OBDH bus was assumed by the ESA study, which is also a very reasonable assumption to make for a European study. However, the MIL-STD-1553B bus is much more popular among suppliers in the United States. The selection of the low-speed bus should be an important trade study for Phase A/B of the WSO mission. If an objective of the mission development phase is to include contractors from other parts of the world, then there might be a significant advantage to one of these busses over the other. However, both of these busses are 20 to 25 years old, and are showing their age. Another approach might be move on to a new bus, such as the I^2C bus. The I^2C bus was developed by a European company (Siemens), but is under consideration for the spacecraft command and control bus in the United States.

High-Speed Bus: A similar situation exists with the high-speed bus as with the low-speed bus described above. While there is no adequate bus ready for space applications at this time, the SpaceWire bus is planned to be used in European spacecraft and the IEEE-1394 bus is planned spacecraft in the United States. While it is possible that one of these busses will become a clear international winner by 2002 (the technology cut-off date for the WSO mission), it is likely that a trade study will be need to be conducted. As in the case of the low-speed bus, the status of development of these (and possibly other busses) in other countries will need to be a factor in this trade study.

The mass, power, and flight hardware recurring engineering estimates are shown. This table is copied from the CEM distribution spread sheet. The mass of the dual string data system architecture is estimated to be 35.6 Kg. The estimated power dissipation will be 133.6 W in the science collection mode.

Assessment

The ICDS developed for the WSO study by ESA was based on a number of devices that are available to the ESA team, and are based on inheritance of some European missions, and technology developments in Europe.

The block diagram for the Team-X version of the ICDS is based on the way that these types of devices are constructed in the United States. As we can see in the diagram, the ICDS is constructed with identical two half systems, a primary and a backup. Each of these half systems uses a VME bus backplane to provide the interconnect between most of the elements of the half system. The Mass Memory Module will use a unique interface that allows for a cross-strapping of the Mass Memory Modules between the two halves of the ICDS.

DATA SYSTEM MASS AND POWER ESTIMATES

Mission: World Space Observatory Red Team 10-00
Element: WSO Orbiter

Referenced from other cell in worksheet
Inputs from Other Subsystems
Inputs Required from You
Database/Calculated
Bold Text Outputs
Red Text Don't Touch

COMMAND AND DATA HANDLING WORKSHEET

Estimated by: ***Joseph Smith***

Last Update ***10/24/00***

Update on Save

Number of CDS Strings **2**

Worksheet Version: 4.04
Date: 6/28/99

Data Base File Name: CDS\Master_db
Version: 3.00
Date: 9/26/00

Technology Cutoff **2002**

			Unit	Mass [kg]	Power per Unit (W)	Cost ($K) Reoccurring	Cost($K) Development	Power [W] Science	Repointing	Safe	Cruise	Launch	NASA TRL	Comments .
		TOTAL		**35.6**	**66.8**	**4,360.0**	**no entry**	**133.6**	**133.6**	**113.6**	**113.6**	**113.6**	**7**	
Processor		**NM PowerPC 604 Module, 30-40MIPS**	2	1.60	30.00	500.00		30.00	30.00	30.00	30.00	30.00	2	Separate Mass Memory module
VME Board	Pwr Cnvtr	**#REF!**	2	2.00	12.00	300.0		12.00	12.00	12.00	12.00	12.00	7	see 10-1-99_CnDH.ppt
VME Board	P/L I/F	**MSR 03 I/O (1553, Discrete,etc)**	4	2.00	33.20	480.0		33.20	33.20	33.20	33.20	33.20	9	see 10-1-99_CnDH.ppt
VME Board	AOCS I/F	**MSR 03 I/O (1553, Discrete,etc)**	2	1.00	16.60	240.0		16.60	16.60	16.60	16.60	16.60	9	see 10-1-99_CnDH.ppt
VME Board	Up/Down	**MSR 03 Uplink/Downlink**	2	0.94	10.00	240.0		10.00	10.00	10.00	10.00	10.00	9	see 10-1-99_CnDH.ppt
VME Board	Mass Memory	**#REF!**	2	1.00	10.00	300.0		10.00	10.00	10.00	10.00	10.00		see 10-1-99_CnDH.ppt
VME Board	Reconfig	**MSR 03 Module I/F**	2	1.00	1.80	200.0		1.80	1.80	1.80	1.80	1.80	9	see 10-1-99_CnDH.ppt
VME Board	Motherboard	**#REF!**	2	1.82		100.0							9	see 10-1-99_CnDH.ppt
VME Board	Housing	**#REF!**	2	4.24										
Solid State Memory		**SEAKR, 64 Gbits**	1	20.00	20.00	2,000.0		20.00	20.00				5	Based on phone call quote, very rough estima
Remote Engineering Unit														

Shielding (kg)
TBD krads TID behind 100 mils of Aluminum

Estimated Subsystem Cost ($M FY97) **19.9**

MCM's per CDS String

The VME based technique for interconnect between the different elements of the ICDS is a common method in the United States. This technique is probably more mass efficient than the technique that is implied in the ESA study for the WSO ICDS. The ESA study implies that each of the elements of the ICDS is separately packaged, possibly each in a different box. These elements are then interconnected with the use of a low speed spacecraft command and control serial bus and also a high-speed instrument serial bus. This technique may have an advantage of reducing the interactions between the ICDS elements, thus allowing them to be built by a number of different companies in a number of different countries.

The basic architecture of the WSO data system, as proposed in the ESA study, is a good solid architecture. While other alternate architectures could be studied, there is no reason to believe that a better architecture can be developed.

One significant difference between the ESA and Team-X studies is the type of interconnect between the different elements inside of the ICDS. The ESA study assumed both a low-speed serial bus (such as OBDH) and a high-speed serial bus (such as SpaceWire). The Team-X study assumed a VME backplane bus, which would be more consistent with commercial practice for this type of system. In either case, this is an important trade study that will need to be resolved during the phase A/B studies.

Other trade studies that need to be considered include the selection of the Low-Speed bus and High-Speed bus technologies. These trades were discussed above.

Block Diagram

This is a block diagram of the World Space Observatory data system from the Team-X study. This system is very similar to the ESA study results.

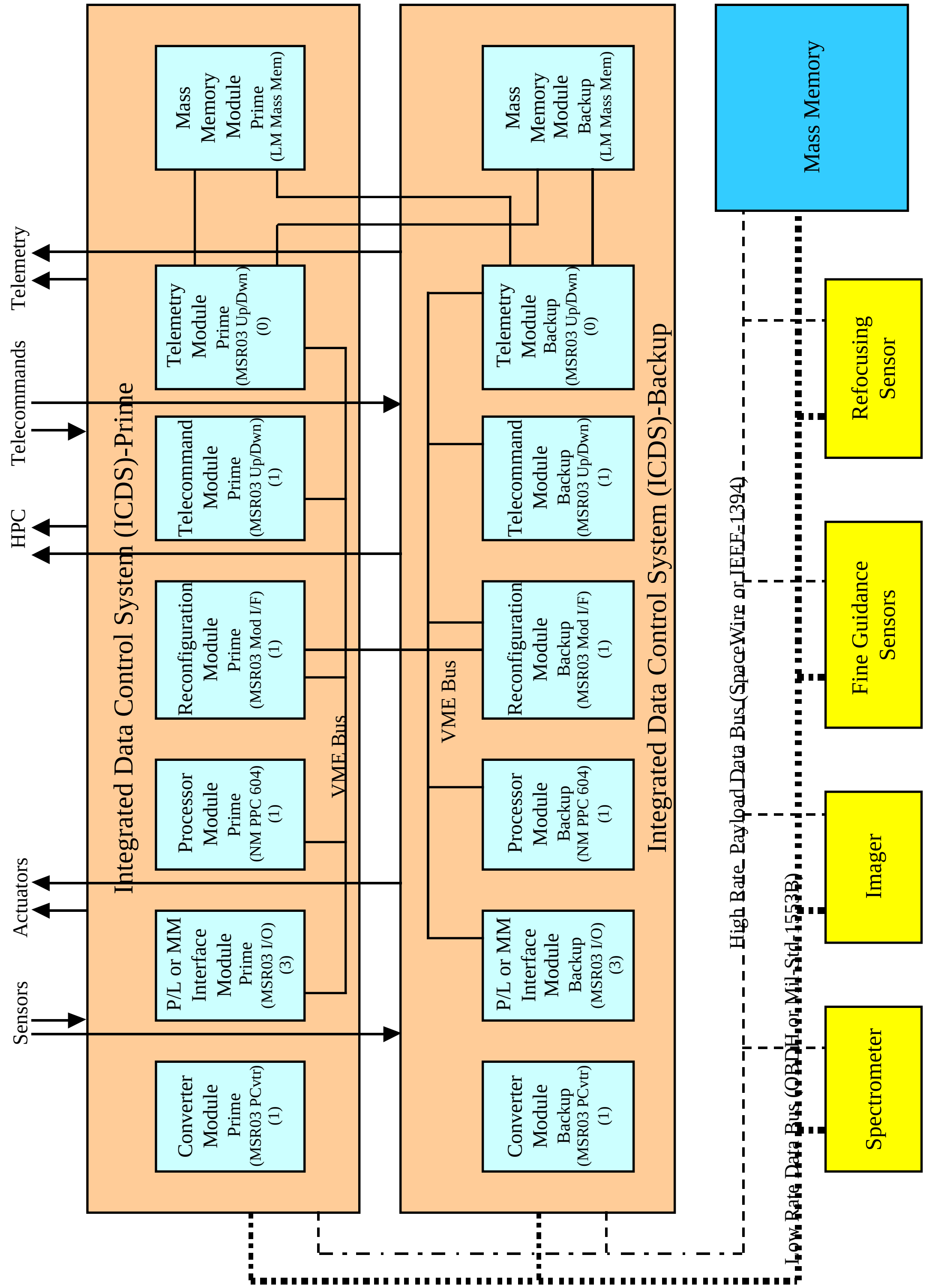


FIGURE 1 – WSO SPACECRAFT DATA SYSTEM ARCHITECTURE FROM THE TEAM-X STUDY

THERMAL CONTROL SYSTEM

Overview

The thermal control system accommodates the subsystems and part of the instrument electronics equipment (-10^{O} C / +50^{O} C assumed as operating temperature range), in addition to the specific requirement s of the WSO, during all expected mission phases and operation modes. As well, the thermal control system radiates the heat dissipated by the instruments and on-board equipment into deep space. The thermal control of the instruments is required to stabilize the temperature during operation and maintain the temperature within the survival limits during non-operational modes.

For the telescope, it is assumed that the thermal control is essentially integrated (the thermal control element mass is included in the construction / structure mass); the telescope does not present a thermal heat flow larger than 10 watt into the platform. The orientation of the Sun to the optical axis is constrained to between 45^{O} – 135^{O}. The advantages of such restrictions are:

- The temperature variation of the Telescope baffle can be reduced by a factor of 5
- Less heater power for the Telescope required (about 20%)
- Less variations of the solar heat input into the telescope and part of the spacecraft

Solar flux can be used for passive heating of the telescope and spacecraft, thereby reducing the required heater power.

The Thermal Design philosophy used for the WSO service module uses passive techniques, with the addition of heater power for special tasks. The design takes into account XMM program past experience, adapted and extended as necessary to meet the new requirements of the WSO.

The WSO thermal design includes particular features for maintaining thermal control. External radiator areas on the solar array sidewalls and the anti-sun side, covered with Optical Surface Reflectors (OSR's) and Multi-Layer Insulation (MLI) Blankets trimmed as necessary to have a better heat rejection into deep space of internal thermal dissipations. Subsystem equipment should be predominantly mounted directly on the inboard faces of radiator areas. The appropriate OSR's radiating area is designed for the maximum dissipation of the spacecraft. The OSR area required is about 2.5 m^2 plus 1.7 m^2 for the instruments without the telescope. The degradation of OSR was not taken into account, because during nominal operation, the radiators will never face the Sun. Interface filters are used as necessary to aid heat rejection from dissipation equipment.

Exposed external surfaces of the spacecraft, i.e. areas other than radiators, optical sensor, propulsion nozzles and antennae feeds, are covered with MLI blankets. The blankets are made of Aluminized Mylar and / or Betacloth sheets, with an electrically conductive outer sheet or laminate to prevent electrostatic discharge and grounded to the spacecraft structure.

Spacecraft internal surfaces generally have a high emittance finish to aid radiative heat transfer and to minimize the temperature gradients within the spacecraft. Therefore, all Aluminum internal surfaces and internal equipment need to be painted black. Antennas and Solar Arrays are thermally isolated from the spacecraft structure and can be treated independently. To maintain low temperatures on the batteries, they are thermally isolated from the spacecraft internal environment. For the WSO / UV, LI-ion batteries (0^{O}C / +20^{O}C is operating temperature range), are mounted to be radiatively coupled directly to space. Heaters controlled by thermostats provide control of minimum temperature. The propellant tanks are mounted via low-conductance stand-offs and wrapped in MLI and heaters to prevent propellant freezing.

WSO / UV uses heaters for two main purposes: a) to provide active temperature control during nominal operation of several platform components, such as instruments, tanks and batteries, and b) to actively control the payload and platform equipment by regulating its temperature as well as providing heat to substitution for units (according to the operating modes). The flight standard heaters are thermofoil flat (redundant, single layer) and linear thermofoil heaters. Heaters will be connected to the power bus at the available voltage. All heater circuits are assumed to be thermostatically controlled.

The thermal control of the instruments (except the telescope) is based on the use of heaters to stabilize the temperature of the optical bench to 20^{O}C. Only the Fine Guidance Sensors mounted on the optical bench operate at 5^{O}C. Peltier elements will be used to cool down the Fine Guidance Sensors because these instruments are inside a sealed compartment. To evacuate the dissipated heat from the instruments, a radiator is required (and maybe a heat-pipe to connect the radiator with the instruments). The outer part of the instrument compartment is wrapped in MLI to reduce temperature variations of the instrument compartment.

Assessment

The thermal design concept uses flight proven elements, which minimizes risk. The mass of the thermal control for the telescope is included in the construction /structure mass.

The science assessment has proposed the option to point closer to the Sun, to about 40 degrees, which is about 5 degrees closer to the Sun. This would lead to a modification to the Sun shade. This change should be minor, but would impact the cost of the system as there will be additional design and fabrication, as well as possible testing (deployment tests, structural and thermal tests).

TELECOMMUNICATIONS SYSTEM

Overview

The X-Band telecommunications system includes a 0.8 m diameter high gain antenna with a 3dB bandwidth of 2.6 degrees and an antenna gain of 30dBi, and three low gain antennas each with 3-dB bandwidth with more than 160 degrees and a gain of 04dBi. The transponders are X-band SDST, redundant, with a mass of to 5.4 kg each. The high power amplifiers are X-band SSPA, redundant, with 30Watt RF output power, and 150Watt DC input power (efficiency 20%) with a unit mass of 2.7 Kg.

During the launch phase, when the spacecraft is operating on battery power, the telecommunications system can perform initial acquisition using exciter power (only 12 dBm), bypassing the high power TWTA. For this purpose a switch was added to the output of one of the STM transponders.

The table below contains a summary of the telecommunications hardware for the X-band and UHF systems.

TELECOMMUNICATIONS SYSTEM MASS AND POWER

Subsystem Totals	9		28.552	162.3	162.3
Components	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Average Power per Unit (W)
X-band 0.8m HGA	1	6.000	6.000		
X-band Omni LGA	3	0.301	0.902		
SDST X-up/X down	2	2.700	5.400	12.3	12.3
X-band SSPA 30 Watt RF	2	6.075	12.150	150.0	150.0
Additional Hardware	1	4.100	4.100		

- The cabling mass from the telecommunications bay/box to the antennas are book kept under the Cabling subsystem.

Assessment

The telecommunications system seems to be reasonably robust and a good amount of margin for the telecommunications links exists. There are 3 LGA with –3 dB gain which allows easy operation during the low data rate operation. The downlink includes the coding that needs about 2.5 dB of bit energy to noise density ratio, which allows choices of codes. The emergency and the uplink modes are comfortably handled using the HGA as well as the LGA. The low rate link with the data rate of 0.5 Kbps seems to be fine assuming a residual carrier modulation and directly modulated carrier. No pointing losses were included in the budget, the pointing loss is considered in the antenna gain.

The High Gain Antenna (HGA) is supposed to be 0.8 to 1.0 meter parabolic reflector antenna. The gain of the HGA is assumed to be 30 dBi. With a 0.8 meter antenna the gain of 30 dBi is obtained if the antenna efficiency of about 23% is assumed. The ESA report assumes 50% efficiency and 34 dBi nom gain with 3 deg 3db beamwidth. The report also considered 3dB pointing losses in the gain figure to minimize antenna pointing that could interfere with observations.

In the event of an emergency, the simultaneous reception via two LGA and transmission via one LGA is not the optimum configuration but is the best compromise for other considerations.

The ranging used is the ESA multipurpose ranging system (MPTS). This ranging system is not compatible with suppressed carrier schemes. Ranging could be done with the low data rate (residual carrier). If needed, an intermediate data rate with modulation SP-L/PM supported with the HGA could be considered with simultaneous ranging.

The masses of the telecommunications hardware shown in the table are without margin. The 30W RF SSPA has 20% efficiency rather than 27% (normally). The additional hardware total mass was calculated to be 4.1 kg. This mass included 3 Diplexer (X-band, 0.28kg each), a Coax RF Transfer Switch (0.1kg each), a Coax 2-Position RF Switch (0.055kg each), a Coax Hybrid Coupler (0.02kg each), a Bandpass filter (0.38kg each), and a Notch filter (0.1kg each).

POWER SYSTEM

Overview

The subsystem must provide 1692 watts for instruments and housekeeping loads at end of life. This power must be produced with the arrays at 45° to the sun. In addition, 3.22% degradation per annum is used for radiation effects. It should also be designed to be rigid enough to resist inducing movement into the telescope.

The system is designed to have redundancy in the battery and battery electronics. The battery capacity is derived from the launch and deployment. The case being considered is the transfer to L2. During this transfer operation, 1/3 of the panels is exposed, and some power is produced so that the batteries must not carry all loads.

The power subsystem is designed for a 50 volt regulated bus. Regulation is accomplished with shunt regulation and/or solar array switching. There are capacitor banks on the bus to stabilize the voltage. With a regulated bus, both charge and discharge controllers are required for the batteries.

The system is a regulated, 50-volt, direct energy transfer system. Solar array switching or shunt regulators are composed of eight channels. The system carries redundant two string batteries and three sets of charge/discharge regulators. The total mass of the subsystem is listed as 98.6 kilograms.

The fixed solar array provides 1642 watts at EOL while tilted to 45° off the sun. BOL power is listed as 2433 watts with 45° offset. The array is built from rigid substrate panels with OSR's on the backside. Each wing is 10.5 square meters, for a total of 21 square meters. They are folded into three segments each. The panels are populated to 94% with 17% efficient advanced silicon cells. Typical operating temperature is 70° Celsius. A temperature coefficient of 0.4%/degree is assumed. Radiation is assumed to be 1x 10^{15} Mev electrons/cm^2. The radiation damage is assumed to be 16.1% over the five-year life. The mass of the solar array is listed to be 61 kilograms, including all components.

There are two 12 ampere-hour lithium-ion batteries. The 12 ampere-hour batteries are fully redundant for worst case discharges. Three battery charge/discharge controllers control them. The DOD for the deployment case is 35% for two batteries and 70% for one battery. The weight of the batteries carried is 3.86 Kg. Each. This corresponds to 85Watt-Hours/Kilogram at the battery level.

The power electronics in broken down into the following components:

Power Conditioning Unit - 16.83 kilograms
    Shunt Regulator or Solar Array Diode Switching

Two units are described, each with four shunt channels, for a total of 8 channels.
Capacitor banks on the Bus Mass in PCU?
Power Distribution – 12.86 Kilograms
Four foldback Current Limiters for stay alive circuits in a single module?
52 low power latching current limiters in four modules (redundant?) for user loads
18 Heater driver circuits in a single module for thermal control
Three battery charge controllers, One Spare,
Three battery discharge controllers, One Spare
Pyro Drivers – 5.00 Kilograms

The main sizing condition for the battery is the launch, deployment, and transfer to orbit. The power listed for the LDT mode is listed as 280 watts. The battery is assumed to be an 8-cell unit, with a nominal 28 -volt operation. The two battery system with 12 Ampere-hours each would contain 672 Watt-hours to 100% DOD. To 35% DOD this provides only 235 watt-hours, which allows 50 minutes of operation without solar array power. In the case of a 70% DOD, 100 minutes is available without solar power. A single exposed panel would provide a maximum of 400 watts. If the spacecraft were spinning, it would produce approximately 200 -watts on average, and this would extend the time to recover attitude to approximately 200 minutes.

Assessment

The power design should meet the WSO mission requirements with adequate margin. The radiation damage of 16.1 % for 5 years seems too high, and more information is needed for advanced silicon designs. The solar array is the same as XMM and Chandra and was picked for its low cost. The array weighs more than a triple junction array, but the higher weight is acceptable on a Proton launch vehicle.

PROPULSION

Overview

- A total velocity increment of 95 m / s, consisting of 75 m / s required to reach station at L2 and 2 m / s per year for 10 years for station keeping at L2.
- An additional 2 kg was assumed for reaction wheel unloading over a 10-year period.
- An initial spacecraft mass of 3356.34 kg
- Functional redundancy.

Most of the information available about the ESA WSO propulsion system design is implicit in the following table that contains information from the ESA equipment list:

Parameter	Quantity	CBE Mass (kg)	Mass Uncertainty (%)	Maximum Mass (kg)
Thrusters	12	3.28	5	3.45
Tanks	2	13.62	5	14.30
Prop Feed	1	2.00	5	2.10
Pipes&Harness	1	0.76	5	0.79
Total Dry Mass		19.66		20.64
Fluid Masses		136.20		
Total Wet Mass		155.86		

The ESA report also states that the thrusters are 5 N thrusters. Recently, thrusters have been developed for Cluster II and Rosetta which incorporate a dedicated Latch Valve for each unit. This allows more flexibility in the operation of the thrusters and increased reliability. The thrusters are located at the base of the spacecraft as far from the telescope as possible. The pitch and yaw thrusters face in the -X direction and are also used for trajectory correction maneuvers. The roll thrusters are located so as to produce the maximum torque with the minimum of plume impingement. A possible location for the roll thrusters is on the plus and minus Y faces of the bus pointing away from the solar panels. It should be possible to find a suitable location for the roll thrusters. The hydrazine is contained in two tanks for configuration reasons. It is easier to fit two tanks into the spacecraft bus than one tank that is twice the size and the two tanks weigh approximately the same. The pitch and yaw thrusters use most of the hydrazine for trajectory correction maneuvers. A small amount (assumed to be 2 kg) is used by all six thrusters to unload the reaction wheels. The thrusters are also be used for initial tipoff rate reduction and for attitude control when safing occurs. The catalyst bed heaters are normally off. They are only turned on for launch and safing. Valve and line heaters may also be necessary to prevent hydrazine freezing. They would be included in the thermal control system. Purified hydrazine would be used as the propellant. The tanks are conventional titanium tanks with diaphragms.

Assessment

This propulsion system design is appropriate for this spacecraft. Monopropellant hydrazine systems have long lives and do not contaminate the spacecraft or science instruments. A spacecraft with a ten-year design life should have redundant thrusters even if the usage of the thrusters is low, as is the case here.

The Advanced Projects Design Team design tool gives the following values for the above propulsion system design:

Parameter	Mass (kg)
Thrusters	3.96
Tanks	14.59
Prop Components	3.04
Lines	1.8
Total Dry Mass	23.39
Fluids	144.7
Total Wet Mass	168.1

The Advanced Projects Design Team design tool yields a total wet mass that is 12.24 kg or 7.9% heavier than the comparable ESA mass. The fluid (propellant and pressurant) mass is higher because the ESA design used a total velocity increment of 85 m/s rather than 95 m/s. The Propulsion section of the ESA document states that the station keeping velocity increment is 1 m/s per year while the other sections use 2 m/s per year. The ESA propellant value also may include less momentum wheel unloading hydrazine than the conservative 2 kg used in the Advanced Projects Design Team estimate. The Advanced Projects Design Team thruster, line, and component mass estimates are all slightly higher than their ESA counterparts. This may be because the values in the Advanced Projects Design Team tool database are heavier Cassini-class components rather than lighter weight later technology components. The Advanced Projects Design Team tool tank mass estimate is higher because of the higher Advanced Projects Design Team propellant mass estimate. This makes the tanks larger and heavier.

The Advanced Projects Design Team tank mass estimate, like the ESA estimate, assumes that conventional titanium propellant tanks with diaphragms are used. However, this is not recommended because the proposed Proton launch vehicle has so much excess launch capability. Also a custom design composite propellant tank has a TRL of 4 whereas an existing design titanium tank has a TRL of 6.

It might be wise to deliberately carry an extra 20 to 50% of excess propellant if the spacecraft bus can accommodate the larger tanks. This would take advantage of the Proton's excess launch capability. This is a higher level decision and not a propulsion system decision.

It is not clear from the ESA proposal whether or how the hydrazine thruster catalyst bed heaters will be used. (Some information is given on Page 57 for the batteries and propulsion combined.) Normal practice is to preheat the catalyst beds of all thrusters that may be used for a particular operation. Thus we would preheat the thrusters that may be used for the tipoff rate reduction, trajectory correction maneuvers, momentum wheel unloads, and safe mode. The catalyst bed heaters may be off at other times.

The 5 N thrusters used may be qualified for a certain number of cold starts but it is always preferable to preheat the thruster catalyst beds before firing the thrusters.

The ESA proposal says (page 66) that the hydrazine tanks will be isolated at launch and will be opened by firing pyro valves to supply the thrusters. Pyro valves are not needed in the proposed design and are not shown in the block diagram. Normal practice is to control flow from the hydrazine tanks to the thrusters by latch valves and to launch with the propellant lines to the thrusters primed with hydrazine.

The use of a dual mode or solar electric propulsion system was briefly considered. Both are much more complex and expensive than a monopropellant hydrazine system but would save some spacecraft mass. The total impulse required is too large to permit the use of a cold gas system. Mass reduction is not important in this case and a monopropellant hydrazine propulsion system is clearly the correct choice.

STRUCTURES

Overview/Assessment

This was intended to be a review of the proposed design and configuration of the WSO spacecraft. Thus, the furnished Mass and Equipment List (MEL), as updated by the customer during the sessions, was replicated in the TeamX spreadsheets.

The spacecraft structure concept seems straightforward and robust. The mass estimate for the major bus structure seems conservative, even considering the complexities of mounting the telescope. Since it is heritage from the XMM mission, some economies of design and construction might be realized.

Of the 61.0kg total listed for the solar arrays, TeamX estimated 30.6kg for solar cells, panel wiring, etc., leaving 30.4kg for the mechanical aspects of the $21m^2$ “2x3” solar arrays. Since this also has to include the solar array hold-down/release and deployment mechanisms, TeamX would consider this total as rather light, especially since the panels must be exceptionally rigid to avoid thermally induced vibrations - recall the Hubble telescope’s solar panel problems. But apparently these panels are heritage from the XMM mission, so the established design is presumably satisfactory.

The HGA 2-axis pointing mechanism at 12.0kg and the hold-down/release hardware at 6.0kg seem perhaps generous, even including the drive electronics at 3.0kg.

GROUND SYSTEM & MISSION OPERATIONS

Overview

During operations, the Ground System will consist of:

- The ground stations and communications network
- The mission operations center (MOC)
- The science operations center (SOC)
- External observers

The MOC will include the infrastructure, computer hardware, flight control system, data processing, and flight dynamics software. Communications is based on 15-meter receivers at the ESA stations at Perth, Kourou and Vilspa. Users' stations may also supplement this communications network.

The MOC will be staffed with operations personnel, provide expertise in spacecraft control and flight dynamics, and interface with the SOC. The SOC will plan observations, calibrate instruments, and process, archive, and distribute data. The flight control system will be a distributed hardware and software architecture providing spacecraft monitoring and control.

A joint Time Allocation Committee (TAC) will evaluate the scientific time requests of the observers. Once the time has been allocated and scheduled, the individual SOCs will further define the observing sequences. There is one primary coordinating MOC and SOC that has central authority. Back-up/alternate MOCs exist. The primary MOC will integrate and uplink the sequences. In case of an anomaly the primary MOC will have overriding authority. On the downlink side the individual SOCs will have common data analysis software, so any SOC can process the data. The primary SOC will give permission to release the data.

Assessment

The operations concept needs further discussion in the WSO Science Working Group. The method by which observation requests are made and priorities are established has not been defined in detail yet.

A distributed operations concept is attractive since it allows the spacecraft to be operated like a ground observatory (i.e. scientists work autonomously). There are, however, many inherent issues with the distributed approach. Cost effectiveness and reliability of the distributed approach need to be evaluated.

Authorization to control the spacecraft needs to be coordinated centrally. Distributed operations require a larger effort for training, tool, and procedure development than a centralized operation. System engineering and coordination is needed to ensure that there is limited or no duplication of effort (e.g. multiple SOCs processing the same data.

The ground stations and communications network approach is sound. The baseline tracking plan employs ESA stations at Perth, Kourour, and Vilspa and the Maspalomas station for nominal operations, with 2–3 hours per day from each station for a total of approximately 8 hours/day of tracking. The cost effectiveness of this scheme should be evaluated -- it might be less expensive to schedule more time from one station rather than small amounts of time at several stations. This, of course, needs to be coordinated with the data-taking activities. User ground stations might also be used (in addition to ESA stations and Maspalomas). The method by which the data will reliably be delivered to the project's primary SOC must be defined.

Network traffic requirements need to be studied to ensure there are no bottlenecks. It is assumed that there are no security issues related to the use of commercial networks to the user SOCs. Apparently, the mission operations system has large amount of inheritance, which reduces cost and risk.

PROGRAMMATICS

Overview

The start of the project phase A will be early 2001 with a launch in the middle of 2006 followed by a three month orbit transfer phase before entering the final L2 orbit. The mission will last for five years with a possible five-year extension. Structural, thermal, and electrical models will be built for testing. A protoflight model will be built with flight standard parts and subjected to qualification tests.

Assessment

The lack of radiation shielding on the UV spectrometer should be studied. There is a likelihood of high radiation levels during part of the transfer phase from LEO to L-2, and it should be confirmed before launch that the expected levels would not damage the instrument. Fortunately, this is a short-term exposure, and it should be easy to establish shielding requirements experimentally.

All technologies are assumed to be at TRL 6 by the end of phase B, in December 2002. As noted in the report, technologies potentially affected by this limit are:

- ACS Guidance Sensors
- DHS Processing Unit
- Isostatic mountings between the Service module, the telescope module, and the Optical Bench.

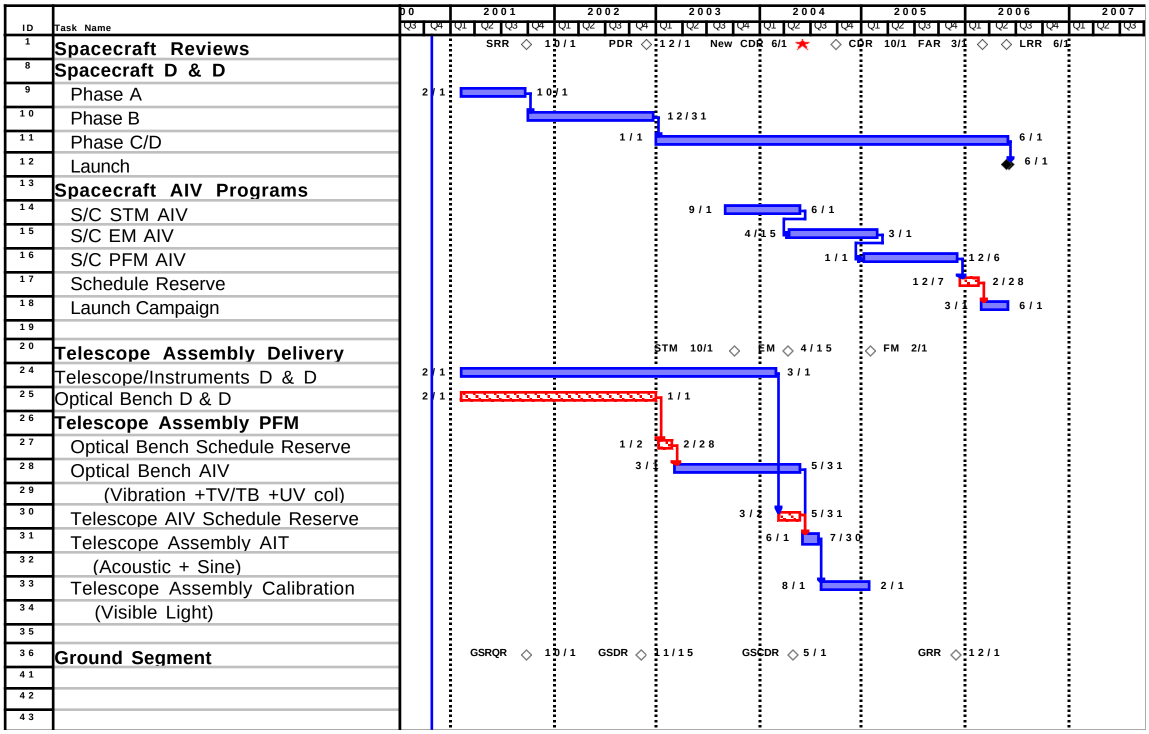
An additional concern is the sensitivity of the scientific payload to contamination. Care must be taken during assembly, integration, verification, and test to maintain high cleanliness standards.

The schedule proposed for this project includes 8 months for phase A, 12 months for phase B, and 38 months for phase C/D. This may be slightly ambitious for phase C/D of a mission with so many suppliers. However, it can be accomplished with careful management attention to schedule. The 20- month planning phase (A/B) should be adequate to prepare for the challenges in phase C/D.

The proposed date for CDR in October of 2004 is late relative to the PFM and Telescope AIVs. We recommend moving this review to early June (red star on schedule). This has the advantage that CDR comes at the end of spacecraft STM integration, and the beginning of Telescope Integration. The Ground Segment CDR needs to be moved to be consistent with the general CDR.

We also recommend approximately one month of schedule reserve per year of development. In the schedule below, this reserve (red hatched bars) is book kept between spacecraft PFM AIV and Launch Campaign, and between Telescope/Instruments D&D, and Telescope Assembly AIV. The preceding tasks were shortened in each case to make room for these reserves.

The software inheritance and development process remains to be developed. An integrated dependency schedule showing hardware and flight software development, ground operations development and AIV is recommended.



PROGRAMMATICS -PRELIMINARY DEVELOPMENT SCHEDULE
(Changes to the Assessment Study Report Schedule are shown in red)

APPENDIX—PROJECT-SPECIFIC

SYSTEMS

WSO EQUIPMENT LIST - ACS, CDS, INSTRUMENTS, POWER, PROPULSION

Mission: World Space Observatory Red Team 10-00
Element: WSO Orbiter

System Level - Summary of Equipment List

Attitude Determination and Control System

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Sun Sensors	3	0.570	1.710	0.0	0.0	Customer Supplied		9	Used for Safe Mode
Fine Guidance Sensors	3	5.780	17.340	21.2	13.0	Customer Supplied		6	1-sigma accuracy 1/25th of 0.054 arcsec pixel, which is 0.0022
Star Trackers	2	19.100	38.200	36.5	36.5	Alenia	SAC-C	6	3-axis autonomous using 3 heads; accuracy 2 to 4 arcsec in all 3
Attitude Anomaly Detector	3	0.150	0.450	0.0	0.0	Customer Supplied		6	None
IMU	2	1.000	2.000	5.0	5.0	Sagem		9	Used for failure detection and during safe mode; high precision
Reaction Wheels	4	9.075	36.300	174.0	58.0	MMS	XMM	9	40 Nms momentum storage per wheel; 4 wheels in a tetrahedron.
Thruster Control Electronics	2	6.820	13.640	7.0	7.0	Customer Supplied		9	For control of twelve 5-N hydrazine thrusters
Failure Detection & Correction Elect.	1	6.260	6.260	11.0	11.0	Customer Supplied		9	None
Antenna Pointing Control Electronics	1	3.000	3.000	10.0	2.0	Customer Supplied		9	For high gain antenna pointing; from Mechanisms equipment list;

Command and Data System

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
ICDS	2	16.000	32.000			TBD			None
Mass Storage	1	18.880	18.880			TBD			None

Instruments

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Telescope	1	1444.000	1444.000		300.0	TBD		8	None
Optical Bench	1	20.000	20.000			TBD		5	None
UV Spectrometer	1	141.000	141.000		125.0	TBD		6	None
UV Imager	1	51.500	51.500		25.0	TBD		6	None
Visual Imager (PR)	1	17.000	17.000		6.0	TBD		5	None

Power

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Solar Array Adv Si, 17% @ wings 10.5	1	29.995	29.995			TBD		5	91 Watts EOL
Power Electronics System	1	35.210	35.210			TBD		3	None
Lithium Ion Batteries 7 AH	2	3.964	7.928			TBD		5	None

Propulsion

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Fuel Tanks	2	6.809	13.618			TBD		6	None
Lines, Fittings, Misc.	1	0.760	0.760			TBD		9	None
Propellant Isolation Assy.	1	2.000	2.000			TBD			None
Thrusters, 22N	12	0.273	3.280			TBD		9	None

WSO EQUIPMENT LIST - STRUCTURES, TELECOMM, THERMAL, SUMMARY

Structures

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Primary Structure	1	237.990	237.990			TBD		6	None
Secondary Structure	1	27.000	27.000			TBD		6	None
Telescope Interface	1	41.730	41.730			TBD		6	None
Solar Array Structure	2	15.200	30.400			TBD		6	None
Antenna Articulation Mechanism	1	15.000	15.000			TBD		6	None
Adapter, Spacecraft side	1	48.260	48.260			TBD		6	None
Adapter, Launch Vehicle side	1	100.000	100.000			TBD		6	None
Cabling	1	32.880	32.880			TBD		7	None

Telecomm

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
0.8m HGA	1	6.000	6.000			TBD		9	0.8
X-band Low Gain Omni	3	0.301	0.902			TBD		9	0
SDST X-up/X down	2	3.000	6.000	12.3	12.3	TBD		9	None
X-band SSPA 30 Watt RF	2	6.000	12.000	150.0	150.0	TBD		9	None
Additional Hardware	1	4.000	4.000			TBD		0	None

Thermal

Component	Flt Units	Mass/ Unit (kg)	Total Mass (kg)	Peak Power per Unit (W)	Aver. Power per Unit (W)	Vendor	Heritage	TRL	Description/Comments
Multilayer Insulation			26.54			TBD		6	None
OSR's			1.88			TBD		7	None
Miscellaneous			3			TBD		6	None
Grolunding Provision			1			TBD		7	None
Heaters/Thermostats			3.5	540	270	TBD		6	None

Element: **WSO Orbiter**

System Level - Summary of Equipment List

Subsystem	Flt Units		Total Mass (kg)	Total Peak Power per Unit (W)	Total Aver. Power per Unit (W)	Note: The accompanying powers listing reflect the artificial situation when all of the devices on the spacecraft are on or at peak power. Please refer to system sheet for proper modes.	Lowest TRL		Description/Comments
Instruments	5		1673.5	0.0	456.0			5	
Attitude Determination and Control System	21		118.9	264.7	132.5			6	
Command and Data System	3		50.9	0.0	10.0			0	
Power	4		73.1	0.0	0.0			3	
Propulsion	16		19.7	0.0	0.0			6	
Structures	9		533.3	0.0	0.0			6	includes cabling and s/c adapter
Telecomm	9		28.9	162.3	162.3			0	
Thermal	0		35.9	540.0	270.0			6	
System Total	**67**		**2534.2**	**967.0**	**1030.8**			**3**	without contingency

APPENDIX—TECHNOLOGY READINESS LEVEL DEFINITIONS

Table D-1: TRL Definitions

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