1997

Design Considerations For A Small Ultraviolet-Visible Space Telescope For Amateur And Professional Use

Donald Platt

PROCEEDINGS OF SPIE

SPIEDigitalLibrary.org/conference-proceedings-of-spie

Design considerations for a small ultraviolet-visible space telescope for amateur and professional use

Donald Platt

SPIE.

Design considerations for a small ultraviolet-visible space telescope for amateur and professional use

Donald Platt[a]

[a]Florida Institute of Technology, Melbourne, FL

ABSTRACT

Presented is a preliminary design for a small ultraviolet-visible (uv-vis) sensitive space telescope. Included will be optical, sensor, and spacecraft subsystem design considerations. With recent advances in Charge Coupled Device (CCD) technology it is now possible to build a space telescope system with wavelength sensitivity from 200 to 1000 nm. A main objective of this design is to keep costs to a minimum and use Commercial of the Shelf (COTS) components wherever possible. Such a low-cost system could provide a platform for studying such untested techniques as small space-based optical interferometry. The observatory could be open to a guest observer program with proposal for observing time accepted from anyone with a scientifically useful plan. Amateurs and professional astronomers alike. The World Wide Web (WWW) could be used to provide data and images to registered subscribers.

Keywords: small space telescopes, ultraviolet telescopes, small satellites

1. INTRODUCTION

Presented is a preliminary design for a small space telescope. Included in this report will be optical, sensor, and spacecraft subsystem design considerations. This paper was a class project for a space engineering class offered jointly by the Boeing corporation, Florida Institute of Technology and the University of central Florida. It was submitted as a proposal for a small satellite mission to be considered to fly as a hitch-hiker payload on a Boeing Inertial Upper Stage (IUS) communications satellite deployment mission for a space engineering design class. Due to having to design for a spacecraft envelope size of 30 cm x 30 cm x 60 cm the primary mirror is limited in size to 10 inches (25.4 cm). However, if a larger spacecraft envelope is available a larger mirror could be used with only minor modification to the system.

With recent advances in Charge Coupled Device (CCD) image systems it is now possible to build a space telescope with wavelength sensitivity from 200 nm to 1000 nm. The near-ultraviolet region to be studied by this system (200 - ~ 400 nm) is of special interest because this wavelength region is largely absorbed by the atmosphere and is therefore not visible from the ground. A folded optics Cassegrain system is the telescope system design of choice due to the limited space available.

Also studied is the possibility of combining two satellites into an optical interferometry system. With the advances in computers, sensors and communications systems light from two telescopes might be combined into an optical interferometer. This would increase the apparent aperture size of the system. This cheap, small system could be a good test-bed for such unproven technology.

The observatory could be open to a guest observer program similar to several other space-based astronomical observatories. Proposals for observing time could be accepted from anyone with a scientifically useful observing plan from professionals and amateurs alike. Perhaps real-time imaging could be displayed over the World Wide Web to subscribers.

2. OPTICS SYSTEM

Due to the size limitations of the design requirements for this mission a small mirror of 10 inches (25.4 cm) is the maximum aperture available. However, with advances in modern astronomical detectors it is possible to do useful research with such a small mirror and if a telescope can be designed with a 16 inch mirror the system could be as capable as most of the professional ground-based observatories.

A Cassegrain folded optics system is the telescope design of choice for this satellite. For volume considerations an extendable section will hold the secondary mirror which will then be deployed upon orbit (see Figure 1). A telescope of this type has a typical focal ratio of 5 with mirrors made of fused silica with a lithium fluoride-aluminum coating.[1]

The focal length of a Cassegrain telescope can then be found from equation 1:

$$F = F_R * D \tag{1}$$

where F is the focal length, F_R equals the focal ratio and D equals the diameter of the primary mirror. Using this equation we get a focal length, F equal to 1.27 meters. The focal lengths of the primary and secondary mirrors can then be selected to fit within the space available on the satellite from equation 2 where f_1 is the focal length of the primary and f_2 is the focal length of the secondary, which is negative for a hyperbolic secondary and d is the distance between the two mirrors.[2]

$$\frac{1}{f} = \frac{1}{f_1} + \frac{1}{f_2} - \frac{d}{f_1 f_2} \tag{2}$$

The Raleigh Diffraction criteria, θr, defines the smallest angle two closely space images like two stars can be and still just be seen as two separate images. This can be found from equation 3 where λ is the peak wavelength, assumed to be 550 nm and D is the telescope primary mirror diameter:

$$\theta_r = 1.22\frac{\lambda}{D} \tag{3}$$

From this equation θr is found to be 5.44 x 10^{-4} arc seconds. Taking advantage of being above atmospheric turbulence this resolution limit should be possible with high quality optics with this telescope system.

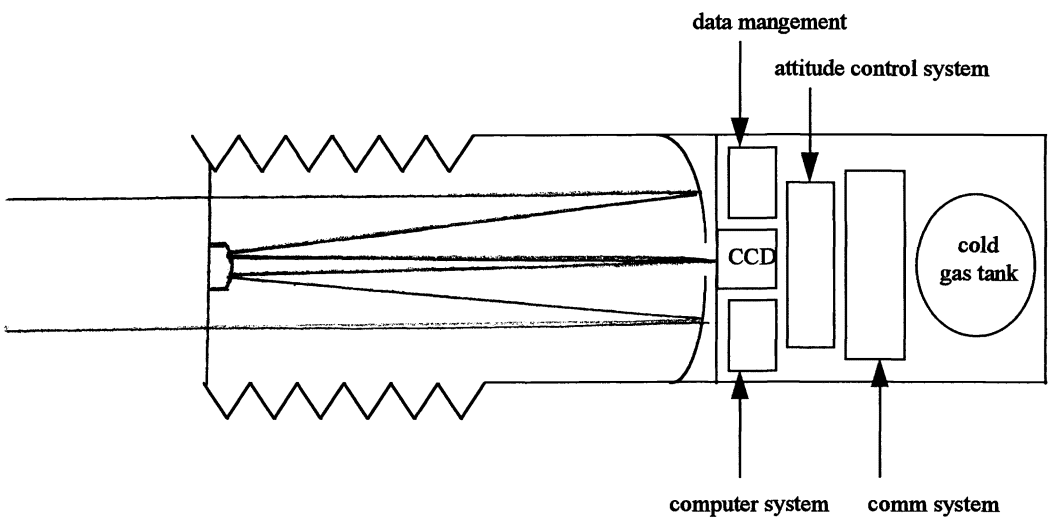

Figure 1. Telescope optical and subsystem layout (estimated)

3. SENSORS

A charge-coupled device (CCD) is the imaging device to be used on this system. A CCD is a solid-state device made from silicon. In crystalline silicon, each atom is covalently bonded to its neighbors. Incident photons of wavelength shorter than 1100 nm can break the bonds and generate electron-hole pairs. The rate of electron-hole pair generation due to thermal energy (dark current) is very dependent on temperature and can be reduced through cooling the system.

When a CCD imager is exposed to light, charge accumulates in the potential wells of parallel registers. Over time a charge can accumulate with the total charge proportional to the intensity of light and exposure time. The serial register then receives the charge packets from the parallel register and the output is sent to an output amplifier. After the serial registers are emptied the next row of charge packets are shifted over in what is called a bucket-brigade.

Quantum efficiency measures a sensor's efficiency in generating electronic charge from incident photons. Electron hole pairs are produced by photons in the region from 400 to 1100 nm. The creation of charge is linear to the light the CCD receives so this allows easy manipulation and processing of multiple exposures with a computer.

Light normally enters the CCD through the gates of the parallel register. These gates are made from very thin polysilicon which is essentially transparent at visible wavelengths but becomes opaque at wavelengths shorter than 400 nm. Light shorter than this wavelength is attenuated by the gate structure when it passes through the frontside of a CCD in what is known as a frontside illuminated CCD. It is possible to create an extremely thin CCD which is about 20 microns thick and then to focus light on the backside of the device. There are no polysilicon gates to absorb light in this system so a back thinned CCD exhibits a quantum efficiency of up to 80 %.

Recent advances in CCD technology has improved the CCD's response to near ultraviolet light. The sensitivity in this wavelength region can be improved using an organic coating called Metachrome. This coating absorbs light between 80 and 400 nm and re-emits it in the visible region of the spectrum. Metachrome is completely transparent to wavelengths longer than 400 nm so the quantum efficiency of the system is not compromised. Other UV coatings reduce the visible light response of the system but since Metachrome is an organic converter it does not exhibit this problem.

Photometrics has developed a METACHROME II phosphor coating that when applied in a 0.4 to 0.6 nm thick film to a frontside-illuminated CCD will greatly improve the quantum efficiency in the 120 to 430 nm range. Photometrics has conducted extensive testing of METACHROME II at both -45 C and -100 C with no cracking or delamination of coatings. When a METACHROME II coated CCD is operated in a vacuum and cooled to at least -40 C, no detrimental effects from exposure to intense ultraviolet or visible light are observed.[3]

Taking advantage of the developments in the field of improving the uv response of CCDs, a possible choice for a uv-visible imaging telescope detector is a back illuminated CCD with 16 bits per pixel, giving 65,535 gray levels. As an alternative a front-illuminated CCD with a METACHROME II coating could be used with a slight loss in UV sensitivity . A spectrograph and filter wheel could also be placed in the light path to greatly increase the scientific value of the telescope's observations.

4. ATTITUDE CONTROL AND POINTING

A zero-momentum reaction control system is selected. In this type of system, a vehicle pointing error produces a signal which speeds up a reaction wheel. This torque corrects the vehicle and leaves the wheel spinning at low speed, until another pointing error makes the wheel either spin faster or slow down.[4] A cold gas thruster would then be used when the wheel reaches saturation to dump momentum.

The three-axis attitude control system allows for accurate pointing and slewing of the system. Cold gas would provide enough energy for momentum dumping and station keeping with a simple, inexpensive system. This type of control system also allows the satellite to point to the same location in space throughout an entire orbital period. This would allow long integration times for faint objects. Pointing accuracy capability of a smallsat three-axis control system should be approximately 0.25 degrees or better.

A new development in attitude control hardware is the flywheel-based Integrated Power and Attitude Control System (IPAC) under development by NASA. These units use flywheels both for attitude control and power systems. Benefits include high usable specific energy, have high efficiency and require less volume than batteries. Energy would be stored more efficiently than rechargeable chemical batteries while also giving pointing control improvements over reaction wheels.[6] Flywheel power/attitude control systems are currently under development and will be flown in space within the next 3 years.

5. STRUCTURAL ANALYSIS

The masses of the various elements of the satellite were estimated. A cylindrical spacecraft structural analysis was carried out which indicates that a monocoque structure will handle the stresses the system will face and will mass about 5 kgs. The analysis followed the method illustrated by Larson and Wertz.[5] In order to fit into a spacecraft envelope of 0.60 x 0.30 x 0.30 meters it was assumed that the spacecraft length was 0.58 m and its diameter was 0.28 m.

Spacecraft cylindrical length (L) = 0.58 m
Spacecraft cylindrical diameter = 0.28 m
Distributed mass (m_B) = 100kg = 981 N

Fundamental frequency:
Axial (from Larson and Wertz, Table 18-9) ~26 Hz (Titan III)
Lateral (from Larson and Wertz, Table 18-9) ~10 Hz (Titan III)

Load factors:
Axial +2.0 G (steady state) +4.0 G (dynamic) Total = +6.0
Lateral 5.0 G

Pressure Differential: 1.1 psi (7,000 Pa)

Material Properties:	7075 aluminum
Young's Modulus (E):	71×10^9 N/m^2
Poisson's Ratio (n):	0.33
Density(r):	2.8×10^3 kg/m^3
Ultimate Tensile Strength (F_{TU}):	524 N/m^2
Yield Tensile Strength (F_{TY}):	448 N/m^2

Axial Rigidity:
Natural Frequency (F_{nat}) = 26 Hz (axial), 10 Hz (lateral)

The natural frequency of the spacecraft structure can be calculated from equation 4:

$$F_{nat}=0.250\sqrt{AE/m_B L} \tag{4}$$

Solving A (axial) = 8.84×10^{-6} m^2

Thickness, t can be solved from equation 5, where R = spacecraft radius, t = 1.00×10^{-3} cm.

$$t = A/(2pR) \tag{5}$$

Lateral Rigidity:

Case c from Fig. 11-41 of Larson and Wertz:

$$10 = 0.560\sqrt{EI/m_B L^3} \therefore I = \frac{10^2 * 100 * 0.58^3}{0.560^2 * 71x10^9} \tag{6}$$

I = 8.76 cm^4 and thickness, t = I/(pR^3) = 1.01x10^{-3} cm

Therefore with this information it is apparent that a minimum wall thickness of 1/8th inch (3 mm) aluminum will be sufficient to provide the necessary rigidity, see Table 1.

Type of Load	Force (Newtons)	Distance (meters)	Load Factor	Limit Load
Axial	981	----------------	6.0	5886 N
Lateral	981	----------------	5.0	4905 N
Bending Moment	981	0.29	5.0	1422.5 N

Table 1. Cylinder Applied Loads

Equivalent axial load:

$$P_{eq} = P_{axial} + 2\frac{M}{R} = \frac{5886 + 2 * 1422.5}{0.14} = 26208N \tag{7}$$

Limit load x ultimate factor of safety = ultimate load
26208 N x 1.25 = 32760 N

Sizing for Strength:

Axial stress (s) = P_{eq}/A
A(area) = 2pRt
Solving for t (thickness) results in t=7.108x10^{-3} cm

Yield Conditions

With factor of safety = 1.10 limit load = 26208 x 1.10 = 28829 N
t, thickness can be calculated using equivalent axial load, P_{eq}, radius and F_{TY}:

$$t = \frac{P_{eq}}{2\pi * R * F_{TY}} = 7.315x10^{-3}cm \tag{8}$$

Sizing for Stability:

Assuming a thickness of 3mm, geometric parameters φ and γ need to be calculated using equations 9.1 and 9.2 for a cylinder in order to calculate cylindrical buckling stress.

$$\varphi = \frac{1}{16}\sqrt{R/t} = \frac{1}{16}\sqrt{\frac{0.14}{0.003}} = 0.427 \tag{9.1}$$

$$\gamma = 1.0-0.901(1.0-e^{-\varphi}) = 0.687 \tag{9.2}$$

Cylindrical buckling stress is calculated from equation 10:

$$\sigma_{cr} = 0.6\gamma\frac{Et}{R} = 0.6(0.687)\frac{71x10^{9}*0.0003}{0.14} = 6.271x10^{8}N/m^{2} \tag{10}$$

Cross sectional area = 2pRt = 2p(14)(0.3) = 26.4 cm^2

$P_{cr} = As_{cr}$ = 0.00264*6.271x10^8 = 1.656x10^6N (ultimate)

Margin of Safety (MS) = Allowable Load / Applied Load - 1.0

MS = 1.656x10^6 / 32760 - 1.0 = 49.6

<u>Internal Pressure</u>:

hoop stress, σ_n, gives the cylinder hoop stress (equation 11):

$$\sigma_{n} = \frac{pR}{t} = 32.7x10^{4}N/m^{2}\ (limit)\ x\ 1.25 = 40.8x10^{4}N/m^{2}\ (ultimate) \tag{11}$$

<u>Mass</u>:

$$\text{Mass} = \text{r2pRtL} = (2.8\text{x}10^{3})(2\text{p})(0.14)(0.003)(0.58) = 4.3\ \text{kg} \tag{12}$$

Therefore, a monocoque shell will work for this satellite's shell. It provides enough strength, stiffness and low total mass. Table 2 shows how an estimated mass budget breaks down.

Subsystem	Mass (kg)
Structure	4.3
Communications	10
Payload	10
Thermal	2
Power	15
ADCS	15
Propulsion	30
Margin	5
Total Mass	92

Table 2. Spacecraft estimated mass budget

6. OTHER SUBSYSTEMS

Power for the system would be provided by solar cells around the outside of the satellite and commercially available re-chargeable batteries. The total power provided by the solar cells can be calculated assuming two arrays with each array being a rectangle of 0.40 x 0.70 meters,. The surface area of the cells would then be 0.18 m^2. The power output from the two solar arrays would be approximately 53 watts at the beginning of the mission. Backup power would be provided by rechargeable batteries during eclipse periods.

Communications will be through a deployable antenna. An option that should be explored is the use of an inflatable antenna to gain a larger aperture while still fitting within the spacecraft launch configuration envelope. Research has been done recently on using high power processor chips and commercially available transmitters in space applications and these components should be explored for communications and data handling subsystems. Space-rated components will be avoided whenever it can be done.

A first approximation of the configuration of the small space telescope system is illustrated in Figure 2. The salient features demonstrated are the extendible secondary mirror assembly in deployed position in this picture, dual solar arrays, and deployable communications antenna.

7. OPTICAL INTERFEROMETRY

An astronomical interferometer combines the light gathering of two or more telescopes separated by a baseline distance. This type of instrument can achieve the resolution of a telescope with an aperture equal to the baseline. The interferometer preserves the phase relationships between beams from different telescopes. The optical train of the interferometer must be continually adjusted to equalize the optical path lengths from the star, through the telescopes and to the detector. Interference fringes will appear if the path lengths are equal to within a few wavelengths. The amplitudes and phases of the fringes are measured and related through a Fourier transform to the distribution of brightness across the sky.

An optical interferometer is much more difficult than a radio interferometry system for two reasons. First, a low-noise, phase coherent amplifier is readily available at radio wavelengths but has yet to be built at optical wavelengths. Secondly, atmospheric turbulence is much less of a factor at radio wavelengths than at optical wavelengths.

A space-based interferometer would then not have to deal with the problem of atmospheric turbulence. It is for this reason that it may be profitable to look at developing a space-based optical interferometer. Two or more spacecraft could be flown in close formation and the light that they both receive could then be digitally added and combined. Having the spacecraft at geostationary altitude would allow one ground control and data processing center to operate the interferometry pair at

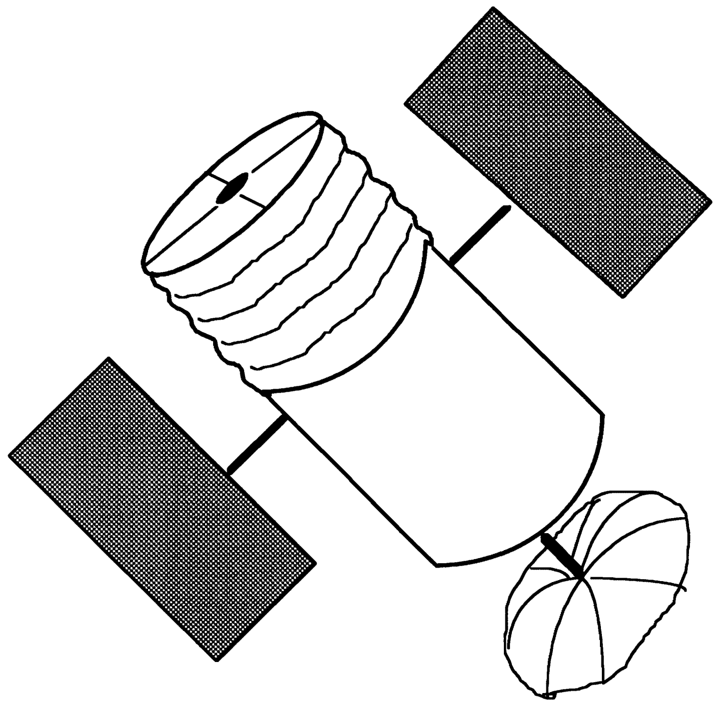

Figure 2. A possible configuration for a small space telescope

all times, and Also, space could provide a large baseline, essentially the separation of a series of spacecraft on opposite sides of the Earth in orbit.

The two key elements of the interferometer are the system that equalizes the path lengths taken by the light waves collected by the separate telescopes and the system that correlates the waves once they have been brought together. The resulting data of fringe amplitudes, phases and positions are used to produce the image.[7] It is also possible that today's advanced computing power could be used to re-construct the fringes in an interferometer. A high-speed computer system could be used to produce these images.

8. OTHER POSSIBLE USES

Small space-instruments are essential for scanning large areas of the sky, for discovering and cataloging new objects, and for monitoring variable or transient sources of radiation.[8] With this telescope being above the atmosphere its images should be virtually diffraction limited, offering excellent image quality and research potential. Professional observatories are currently vastly over booked for observing time so a series of these cheap small space telescopes could be used to do many observing projects that currently require large ground-based telescopes.

This system could be used for long-period observing programs that currently can not be done with large earth-based systems due to limited observing time. The type of projects that the small space telescope could tackle include observations of variable stars over extended time frames, observations of binary stars, ultra-violet photometry and spectroscopy and extra-galactic supernova observations.

Observations could be planned and controlled over the internet similar to robotic observatories currently being used by the Bradford Robotic Telescope project at Yorkshire, England.[9] Users could register and then be allowed to use the system after paying an observation fee.

9. CONCLUSION

Flying above the atmosphere, even a small space telescope can produce valuable research. By using commercially available components, whenever possible, cost of the system can be kept down. Several satellites could be placed in orbit for redundancy, multiple observations and possibly to develop an optical interferometer system. Observing time can be made available to both professional and amateur groups and access can be offered through internet to subscribers and registered users.

With new advances in CCD technology, the telescope can use an inexpensive CCD detector with its linear response and computer capabilities to observe from the near ultraviolet to the near infrared. By placing a spectrograph on board the satellite an even more capable system can be developed.

ACKNOWLEDGMENTS

The author wished to thank Professors Mark Moldwin of the Florida Institute of Technology (FIT) and Roger Johnson of the University of Central Florida (UCF) for their technical support and advise. Also gratitude is expressed to the Boeing/IUS Titan IV integration team and US Air Force at Cape Canaveral Air Station (CCAS). This project was a class project for the joint Boeing, FIT, UCF spacecraft processing class conducted at CCAS during the fall, 1996.

REFERENCES

1. B. Thompson, and R. Shannon, *Space Optics-Proceedings of the Ninth International Congress of the International Commission for Optics,* National Academy of Sciences, Washington, D.C., 1974.
2. G. Walker, *Astronomical Observations,* Cambridge University Press, Cambridge, 1987.
3. Photometrics World Wide Web internet site, *Metachrome Reference Page*, http://www.photomet.com/ref/ref.metachrome.html, 1997.
4. J. Eterno, R. Zermuehlen, H. Zimbelman, "Attitude Determination and Control", *Space Mission Analysis and Design,* Larson, Wiley J. and Wertz, James R., eds., pp. 340-366, Microcosm, Inc., Torrance, CA, 1992.
5. P. Doukas, J. McCandless, T. Sarafin, W. Britton, "Structures and Mechanisms", *Space Mission Analysis and Design,* Larson, Wiley J. and Wertz, James R., eds., pp. 430-460, Microcosm, Inc., Torrance, CA, 1992.
6. NASA World Wide Web internet site, *Aerospace Flywheel Development Program*, http://godzilla.lerc.nasa.gov/ppo/flywheel
7. J. Armstrong,, D. Hunter, K. Johnston, D. Mozurkewich, "Stellar Optical Interferometry in the 90's", Physics Today, May, pp. 42-49, 1995.
8. F. Dyson, "Benefits of Small and Large Space-based Astronomy", Y. Kondo, Y., ed., *Observatories in Earth Orbit and Beyond*, Kluwer Academic Publishers, The Netherlands, 1990.
9. J. Baruch,. "Ready for Robotic Astronomy?", Sky & Telescope, December, pp. 92-94, 1996 .