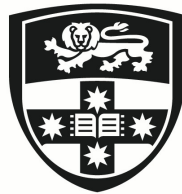


The Thermo-Mechanical Control and Analysis of the TOLIMAN Space Telescope

MARK A. GEORGE

BEng (Hons)/BSc



THE UNIVERSITY OF
SYDNEY

Supervisor: Dr. Xiaofeng Wu
Associate Supervisor: Professor Peter Tuthill
Associate Supervisor: Dr. Christopher Betters

A thesis submitted in fulfilment of
the requirements for the degree of
Bachelor of Engineering Honours and Bachelor of Science

School of AMME
Faculty of Engineering
The University of Sydney
Australia

14 November 2021

Contribution of Work

I, Mark A. George, declare that this thesis titled "The Thermo-Mechanical Control and Analysis of the TOLIMAN Space Telescope" and the work presented within this thesis is in compliance with The University of Sydney policy on academic honesty. All information obtained from the work of others has been acknowledged and referenced.

The following is a list of my contributions to this thesis:

- I carried out the literature review
- All research, thermal design, and analysis was done by myself
- I wrote and prepared this thesis report
- All conclusions are my own, with influence from my supervisors

The above represents an accurate summary of my contribution to this thesis.

Acknowledgements

I would like to thank my supervisors Dr. Xiaofeng Wu, Professor Peter Tuthil, and Dr. Christopher Betters who have guided me throughout my work and provided me with all necessary resources.

I also wish to show my gratitude to Fred Crous who attended meetings in the later stages of my work and provided further engineering guidance.

Finally, I would like to thank my family and friends for their encouragement and support.

Abstract

The Telescope for Orbital Locus Interferometric Monitoring of our Astronomical Neighborhood (TOLIMAN) is a microsatellite mission that aims to detect habitable exoplanets in the Alpha Centauri star system. TOLIMAN is designed around a 13 cm Ritchey–Chrétien telescope in low Earth orbit that uses a reformulation of an optical configuration known as a “diffractive pupil” to measure the micro-arcsecond angular deflections between stars. TOLIMAN aims to address a critical blind spot in our astronomy capability; answering the basic question of whether there are Earth-analog exoplanets orbiting our nearest-neighbour star systems. This innovative mission concept requires an optical system with thermo-mechanical stability previously unseen in this class of telescope.

This thesis introduces the design and analysis of a “cold bias” thermal control system for TOLIMAN. Through extensive simulations, candidate orbits are studied, exploring the trade space between thermal performance and viewing time on the science target. Parametric thermal models and optimisation routines are used for the determination of worst case orbit conditions and optimal design choices. Reduced order thermal models are also created based on finite element models for the design of an active heater control system. These thermal models and their results are integrated with structural models to assess the dimensional stability of the primary mirror. The proposed design can achieve on orbit temperature stability of $20 \pm 0.05^\circ\text{C}$ on the primary mirror, and $20 \pm 0.25^\circ\text{C}$ on the metering structure through a synthesis of active and passive control techniques. The final outcome of a predicted RMS dimensional stability of better than 5 nm on the primary mirror is highly encouraging for the precision optical measurements required to accomplish the core mission science.

Contents

Contribution of Work	ii
Acknowledgements	iii
Abstract	iv
Contents	v
List of Figures	ix
List of Tables	xv
Chapter 1 Introduction	1
Chapter 2 Literature review	4
2.1 The TOLIMAN Mission Concept and Other Space Telescope Missions	4
2.1.1 The TOLIMAN Space Telescope	4
2.1.2 Past Small Space Telescope Missions	5
2.2 Basic Theory of Thermal Heat Transfer	8
2.2.1 Thermodynamics Basics	8
2.2.2 Conduction	9
2.2.3 Contact Resistance	10
2.2.4 Radiation	11
2.3 The Design of Thermally Stable Space Telescopes	13
2.3.1 Effect of Orbit Geometry	13
2.3.2 Thermal Control Technologies	16
2.3.3 Examples of Thermal Control in Small Space Telescope Missions	20
2.4 Methodologies for Thermal Analysis and Design	23
2.5 Approach Taken in this Thesis	25
2.6 Summary	26
Chapter 3 Thermal Modelling	27

3.1	Introduction	27
3.2	Thermophysical Properties	27
3.3	Surface Optical Properties	28
3.4	Modelling Thermal Contact Resistance	29
3.4.1	Uniform Contact Pressure	29
3.4.2	Bolted Contacts	30
3.4.3	Adhesive Contacts	31
3.5	Orbital Heating Environments	31
3.6	Detailed Spacecraft and Telescope Model	33
3.6.1	Spacecraft Model	34
3.6.2	Detector Model	36
3.6.3	Telescope Model	37
Chapter 4	Passive Thermal Control and Cold Bias	39
4.1	Introduction	39
4.2	Passive Control Design	39
4.3	Study of Potential Orbits	41
4.4	Worst Case Orbit Determination	44
4.5	Baffle Length Study	48
4.6	Sensitivity Analysis	50
Chapter 5	Active Heater Control Design	52
5.1	Introduction	52
5.2	Heater Placement Optimisation	52
5.3	Model Order Reduction	54
5.3.1	Reduced Order Models	55
5.3.2	Parameter Fitting	57
5.4	Feedback Controller Design and Simulation	61
Chapter 6	Structural Stability Analysis	66
6.1	Introduction	66
6.2	Structural Finite Element Model	66
6.3	Calculation of Grid Sag Profile	67
6.4	Uniform Temperature Soak	70
6.5	On Orbit Temperature Profiles	72

Chapter 7	Conclusions and Future Work	74
Bibliography		76
Appendix A	Further Results	84
A1	System Identification	84
A2	PI Control Simulation	88
A3	On Orbit Mirror Sag Profiles	92
Appendix B	Material and Surface Optical Properties	95
Appendix C	Calculation of Thermal Conductances, Lumped Properties, and Effective Conductivities	97
C1	Spacecraft Model	97
C1.1	Lumped Component Properties	97
C1.2	Component Power Dissipation	97
C1.3	Contact Conductances	98
C1.4	Solar Array Honeycomb Insulation Thermal Conductivity	98
C2	Detector Model	100
C2.1	Lumped Component Properties	100
C2.2	Component Power Dissipation	100
C2.3	Contact Conductances	100
C3	Telescope Model	101
C3.1	Uniform Pressure Contact Conductance	101
C3.2	Adhesive Contact Conductance	101
C3.3	Bolted Joint Contact Conductance	102
C4	Patch Heater Modeling	102
Appendix D	Mesh Convergence Study of Telescope Thermal Model	104
Appendix E	Mesh Convergence Study of Telescope Structural Model	105
Appendix F	Orbit Propagator	106
F1	Orbital Mechanics	106
F1.1	Coordinate Systems	106
F1.2	Orbits as a Function of Time	107
F1.3	Perturbations	109

F1.4	Solar Zenith Angle	109
F1.5	Calculation of Earth Vector	109
F1.6	Calculation of Sun Vector	110
F1.7	Eclipse Determination	111
F2	Orbit Propagator Algorithm	111

List of Figures

- 2.1 A star and planet orbiting each other will orbit about the centre of mass of the system – the source of reflex motion for the host star. 5
- 2.2 Tuthill et al. [3]. Candidate point-design pupil for the TOLIMAN mission (A). The black/white regions indicate discrete phase steps of $0/\pi$, resulting in a diffraction pattern due to relative differences in optical path length that light takes. The monochromatic Point Spread Function (PSF) generated (B) has a complex/fine featured pattern which extends radially from the centre, creating many sharp fringes. 6
- 2.3 Heat flows from the high to low temperature regions according to Fourier’s law (A), if there is no internal heat generation and material properties are homogeneous, an analogous 1D thermal resistance can be defined (B) for the object. 10
- 2.4 Contact between two objects (A). Microscopically, the contact is not perfect due to roughness of the two surfaces, and contact is only achieved in certain locations, heat can travel through the contact points $q_{contact}$, or through the gas (if present) between q_{gap} (B). 11
- 2.5 Schematic of surface dA_1 emitting radiation to surface dA_n , which is normal to the emitted rays, the z is the normal to the surface dA_1 . 12
- 2.6 Illustration of various beta angles, where $\hat{S}$ is the solar vector – the unit vector pointing from the sun to the Earth. Beta angles for $\beta = 0^\circ$ (A), $0^\circ < \beta < 90^\circ$ (B), and $\beta = 90^\circ$ (C) are shown. 14
- 2.7 Spacecraft orbital plane (blue) relative to the sun during the Earth’s orbit (red) around the sun for the JASMINE mission. Observations of the primary target were only made during the Spring and Autumn seasons due to positioning of the sun during the year. 15
- 2.8 Heat transfer that occurs between a spacecraft and its environment. Flows include direct heat flux from the sun, reflected solar heat flux from the earth (albedo), and planetary radiation emitted from the Earths surface and atmosphere. The surface of the satellite will radiate heat back to the environment. 17

2.9	The basic operating principle of MLI. Incident heat is continually reflected by layers of low emissivity film, resulting in very small heat transfers.	18
2.10	from Meseguer et al. [38], a typical thermal strap (A). One end is bolted to the hot surface, and the other is bolted to the cold surface, heat is transferred between the two ends via strap of high thermal conductivity. Typical mounting (B).	19
2.11	Knapp and Seager [8]. Thermal control diagram for the ASTERIA telescope assembly.	20
2.12	Utsunomiya et al. [26]. Structural design of JASMINE telescope.	21
2.13	Scowen et al. [23]. Payload assembly for the SPARCS telescope. The payload occupies 3U volume within the satellite.	22
2.14	Scowen et al. [23]. SPARCS telescope tube assembly with materials labelled for each component.	23
2.15	A map of the processes and software packages used for thermal and structural analysis, and their integration.	26
3.1	Typical mapping of Earth albedo (left) and Earth IR flux (right) from the top of the atmosphere, on the Spring Equinox (21/03/2011) obtained from Peyrou-Lauga [70]	33
3.2	Internal view of spacecraft thermal model (A) and dimensions (B). Dimensions are in mm.	35
3.3	Top (A) and bottom (B) side of the FLIR BFS-U3-200S6M-BD2 detector thermal model.	37
3.4	Thermal model of telescope assembly.	38
4.1	Passive control design and interface with spacecraft. Some details are omitted due to confidentiality agreements with other parties involved in the design and manufacturing of the telescope.	40
4.2	Sun angle $\theta_{\odot}$ and Earth angle $\theta_{\oplus}$ measured between the respective vectors and the optical axis.	41
4.3	Alpha Centauri viewing times a given Solar Right Ascension (SRA) and orbital Right Ascension of the Ascending Node (RAAN) whilst in sun (A) and in eclipse (B) for minimal optical axis angles $\theta_{\odot} = 90^{\circ}$ and $\theta_{\oplus} = 110^{\circ}$, and a 7° inclination orbit.	43
4.4	Alpha Centauri viewing times a given Solar Right Ascension (SRA) and orbital difference between SRA and Right Ascension of the Ascending Node (RAAN) whilst in sun (A) and in Eclipse (B) for minimal optical axis angles $\theta_{\odot} = 90^{\circ}$ and $\theta_{\oplus} = 110^{\circ}$, and an SSO orbit.	44

- 4.5 Cold case (A) and hot case (B) orbit for 7° inclination orbit as obtained from worst case search. 46
- 4.6 Temperatures after four orbits for 7° inclination orbit cold (A) and hot (B) case in Table 4.1, and total heat transfers out of telescope structure for cold (C) and hot (D) cases. 47
- 4.7 Cold case (A) and hot case (B) orbit for SSO orbit as obtained from worst case search. 47
- 4.8 Temperatures after four orbits for SSO orbit cold (A) and hot (B) case in Table 4.1, and total heat transfers out of telescope structure for cold (C) and hot (D) cases. 48
- 4.9 Example temperature distribution of spacecraft (A) and telescope (B) at a single point in orbit for the 7° inclination cold case orbit. Note the temperatures shown are for the structure beneath any MLI that is used. 48
- 4.10 Average orbital temperature (A) and maximum orbital temperature gradient (B) of telescope structure for various baffle lengths at each orbital case. 49
- 4.11 Sensitivity analysis parameter sweep. Panels (A), (C), (E), (G), and (I) give orbital average temperatures as the telescope IR emissivity, enclosure to spacecraft conductance, telescope tube to enclosure contact conductance, detector heat generation, and spacecraft heat generation is varied over a range of uncertain/possible values for all four worst case orbits. Panels (B), (D), (F), (H), and (G) respectively give the maximum temperature gradient achieved throughout an orbit. 51
- 5.1 Patch heater placement on telescope, where the location of the patch heaters is parameterized by h_1 and h_2 describing their position as explored by the optimisation routine. 53
- 5.2 Steady state temperature distributions on the telescope structure obtained after heater placement optimisation routine using orbital average heating rates for 7° inclination cold case orbit (A) and SSO hot case orbit (B). Both figures use the same temperature scale. 54
- 5.3 The two thermal capacitance-resistor model used in creation of the reduced order model. Variables u_1 and u_2 represent heater inputs to each node, d is the disturbance input, a is the fraction of the disturbance on each node, $C_{th,1}$ and $C_{th,2}$ are the heat capacities of each node, and R is the thermal resistance between each node. 55

- 5.4 Example heater input sequence used for system identification. A 3rd order Butterworth low-pass filtered white noise signal, with a cutoff frequency of 0.4 mHz, mean signal value of 5 W, and standard deviation of 3.25 W. 58
- 5.5 System identification results for 7° inclination Cold Case orbit. Note all orbital cases were used in the fitting of parameters. Panels (A), (C), (E), (G), and (I) show the nodal temperatures, heater inputs, and disturbance inputs for the training data. Panels (B), (D), (F), (H), and (J) show the same results for a validation case. 59
- 5.6 Linearised disturbance model from Equation 5.6 for 7° inclination cold (A) and hot (D) cases, and SSO cold (B) and hot (D) cases for coefficients in Table 5.1 compared to simulation data obtain from Thermal Desktop. 61
- 5.7 PI controller simulation for 7° cold case orbital condition. Panels (A), (C), and (E) show the sensor temperatures, heater input, and heat transfers for the Thermal Desktop model. Panels (B), (D), and (F), show the equivalent simulation using the reduced order state space models, with (F) being the disturbance input. 63
- 5.8 Telescope tube (A), primary Mirror (B), and secondary mirror (C) temperatures for the 7° inclination cold case orbit from Thermal Desktop model using PI controller. 64
- 6.1 Finite element structural model of the telescope (A) and primary mirror (B) in ANSYS Mechanical used for deformation analysis. 67
- 6.2 Sag s of primary mirror and radius of curvature R . 68
- 6.3 A displaced node on the mirror surface will correspond to a different point on the mirror due to displacement components in the radial direction. 69
- 6.4 Sag profile of undeformed primary mirror. 69
- 6.5 Δs profiles of primary mirror for uniform temperature soak for temperature changes of $\Delta T = -0.25^\circ\text{C}$ (A), $\Delta T = -0.1^\circ\text{C}$ (B), $\Delta T = +0.25^\circ\text{C}$ (C), and $\Delta T = +0.1^\circ\text{C}$ (D) from the set point temperature $T = 20^\circ\text{C}$. 70
- 6.6 Von-Mises equivalent stress on the telescope structure (A) and total deformation magnitude (B) for a temperature soak of $\Delta T = -0.25^\circ\text{C}$ 71
- 6.7 Temperature distributions for 7° inclination cold case orbit at times corresponding to 812 min (A), and 867 min (C) in simulation results from Figure 5.8 and their respective Δs profiles in (B) and (D). 72
- A.1 System identification results for 7° inclination Cold Case orbit. Note all orbital cases were used in the fitting of parameters. Panels (A), (C), (E), (G), and (I) show the

- nodal temperatures, heater inputs, and disturbance inputs for the training data. Panels (B), (D), (F), (H), and (J) show the same results for a validation case. 85
- A.2 System identification results for 7° inclination Cold Case orbit. Note all orbital cases were used in the fitting of parameters. Panels (A), (C), (E), (G), and (I) show the nodal temperatures, heater inputs, and disturbance inputs for the training data. Panels (B), (D), (F), (H), and (J) show the same results for a validation case. 86
- A.3 System identification results for 7° inclination Cold Case orbit. Note all orbital cases were used in the fitting of parameters. Panels (A), (C), (E), (G), and (I) show the nodal temperatures, heater inputs, and disturbance inputs for the training data. Panels (B), (D), (F), (H), and (J) show the same results for a validation case. 87
- A.4 PI controller simulation for 7° hot case orbital condition. Panels (A), (C), and (E) show the sensor temperatures, heater input, and heat transfers for the Thermal Desktop model. Panels (B), (D), and (F), show the equivalent simulation using the reduced order state space models, with (F) being the disturbance input. 89
- A.5 Telescope tube (A), primary mirror (B), and secondary mirror (C) temperatures for the 7° inclination hot case orbit from Thermal Desktop model using PI controller. 89
- A.6 PI controller simulation for SSO cold case orbital condition. Panels (A), (C), and (E) show the sensor temperatures, heater input, and heat transfers for the Thermal Desktop model. Panels (B), (D), and (F), show the equivalent simulation using the reduced order state space models, with (F) being the disturbance input. 90
- A.7 Telescope tube (A), primary mirror (B), and secondary mirror (C) temperatures for the SSO cold case orbit from Thermal Desktop model using PI controller. 90
- A.8 PI controller simulation for SSO hot case orbital condition. Panels (A), (C), and (E) show the sensor temperatures, heater input, and heat transfers for the Thermal Desktop model. Panels (B), (D), and (F), show the equivalent simulation using the reduced order state space models, with (F) being the disturbance input. 91
- A.9 Telescope tube (A), primary mirror (B), and secondary mirror (C) temperatures for the SSO hot case orbit from Thermal Desktop model using PI controller. 91
- A.10 Temperature distributions for 7° inclination hot case orbit at times corresponding to 785 min (A), and 825 min (C) in simulation results from Figure A.5 and their respective Δs profiles in (B) and (D). 92

A.11	Temperature distributions for SSO cold case orbit at times corresponding to 815 min (A), and 885 min (C) in simulation results from Figure A.7 and their respective Δs profiles in (B) and (D).	93
A.12	Temperature distributions for SSO hot case orbit at times corresponding to 815 min (A), and 885 min (C) in simulation results from Figure A.9 and their respective Δs profiles in (B) and (D).	94
C.1	Insulation layered structure used on spacecraft solar array. Solar cells are bonded with adhesive to one of the two face sheets of the honeycomb insulation layer.	98
C.2	Schematic diagram of honeycomb insulation, W , S , and T are the dimensions of the honeycomb sheet, S is the size of a single honeycomb cell, and δ is the thickness of the ribbon used to construct the honeycomb.	99
C.3	Finite element thermal model for telescope in Thermal Desktop with finite difference surfaces used to represented patch heaters (A) and sensing regions beneath the patch heaters (B).	103
D.1	Mesh convergence study results for thermal finite element model of telescope, maximum (A) and minimum (B) temperatures on the telescope structure are used.	104
E.1	Mesh convergence study results for structural finite element model of telescope, maximum (A) and minimum (B) Δs on the primary mirror are used.	105
F.1	ECI frame defined by reference directions, the vernal equinox and north pole, with right ascension α and declination δ used to define the location of a point on the celestial sphere.	106
F.2	An illustration of six classical orbital elements used to describe an orbit and the position of a satellite within it. The angular momentum vector $\mathbf{h}$ is perpendicular to the orbital plane, the eccentricity vector $\mathbf{e}$ passes through the point of closest approach (the perigee), and the node line vector $\mathbf{N}$ passes through the ascending node and lies within Earth's orbital plane.	107
F.3	The true anomaly θ is the angle measured from the perigee centered at the focus of the orbit, the eccentric anomaly E is the angle measured from the center of the circumscribed auxiliary circle of the intersection of the line project vertically from the satellite position onto the circle.	108
F.4	Illustration of the solar zenith angle θ_{SZ} , the angle between the Earth-Sun vector and the Earth-satellite vector.	109

List of Tables

3.1 Combined hot, cold, and mean cases for thermal radiation coming from the Earth for an 128 second averaging time [69].	33
4.1 Orbital worst cases for 7° inclination and SSO orbits and corresponding viewing times.	46
5.1 Coefficients use for linearised disturbance model given by Equation 5.6	60
B.1 Thermophysical properties for bulk materials used in the creation of thermal models.	95
B.2 Surface optical properties for various surfaces used in thermal models.	96
C.1 Lumped thermophysical and optical properties of electronic components used in spacecraft model.	97
C.2 Nominal heat generation and component power fractions for spacecraft thermal model.	98
C.3 Effective thermal conductivities for honeycomb core and effective emissivity between face sheet material for [98] with $S = 0.01$ m, $\delta = 0.5$ μ m, and $k = 190$ W/m/K.	100
C.4 Nominal estimated heat generation and component power fractions for detector thermal model.	100
C.5 Estimated surface roughness properties used for calculation of bare surface contact conductances.	101
C.6 Bare surface contact conductance coefficients for various interfaces used in thermal analysis for contact pressures of 30 kPa.	102

CHAPTER 1

Introduction

Finding Earth-like exoplanets in our immediate solar neighbourhood has been one of the priorities in astronomy for decades. A 30 year astrophysics roadmap published by NASA in 2013 [1] puts the discovery of Earth like exoplanets as the first stage in the quest for answering the fundamental question; “are we alone?”, and was identified as a priority area to motivate recommended investments in the 2020 Decadal Survey on Astronomy and Astrophysics [2]. The Telescope for Orbital Locus Interferometric Monitoring of our Astronomical Neighborhood (TOLIMAN - also a name for the Alpha Centauri star system from antiquity) is a Sydney University led mission, funded by a grant from the Breakthrough Watch Initiative, with the aim of detecting habitable exoplanets in the near solar environment, specifically, the Alpha Centauri star system. The mission is designed around a microsatellite platform hosting a 13 cm Ritchey–Chrétien (RC) telescope as its main payload.

TOLIMAN’s ambitious mission objective will be achieved using a novel optical architecture – a reformulation of a configuration known as a “diffractive pupil” – to measure the angular deflection between the binary stars that results from gravitational perturbations induced by planets orbiting the host star [3, 4, 5]. In order to make the required micro-arcsecond level angular measurements, a telescope structure with extreme thermo-mechanical stability and small footprint is required. Traditionally, lower cost space telescopes developed by smaller institutions used for astronomy are scarce. In more recent times, with the miniaturisation of technology, their benefits have become clear [6, 7], and their mission objectives more ambitious. Missions such as ASTERIA [8], SPARCS [9], and MOST [10] have begun paving the way for small space telescopes in astronomy. The TOLIMAN mission is another step in the same direction, requiring an optical system of thermo-mechanical stability not yet seen in this class of telescope.

Currently, the TOLIMAN mission is progressing into the late design phase. The spacecraft bus is designed and manufactured by OHB Sweden [11], with ground station control and mission operations supported by Saber Astronautics [12]. The 13 cm RC Silicon Carbide (SiC) telescope is manufactured by Aperture Optical Sciences (AOS) [13], with the diffractive mask being fabricated at JPL. The all-SiC telescope structure forms the base of the thermally stable optical system. The optical design, thermal control system, and mechanical interface for the telescope is being designed by the University of Sydney.

This thesis proposes the preliminary thermal control structure for the TOLIMAN space telescope, carries out an in depth thermal analysis for two candidate orbits, and studies the resulting structural stability of the proposed control approach. There are two main innovations here. Firstly, the sub-Kelvin temperature, and nanometer level dimensional stability requirements place TOLIMAN in a category of its own in terms of small space telescopes, particularly for missions led by small institutions and universities. The novelty here is not only in the thermal design itself, but also in the methodology of thermal analysis, doing away with the traditional point design approach typically seen and instead making use of parametric thermal models and optimisation routines to allow thermal analysis to participate earlier in the design process. Secondly, this work serves to outline and provide a more complete resource in the main theory, methodologies, and technologies used in thermal analysis for space telescope missions.

Chapter 2 is a review of the literature, with a focus on past and current space telescope missions similar to TOLIMAN, and the various control approaches taken. Basic theory in thermal analysis is introduced, as well as commonly used thermal control technologies and methodologies. Chapter 3 introduces in detail the creation of numerical thermal models in Thermal Desktop (a comprehensive software package used for the thermal analysis in space). The underlying concepts such as calculation of thermal heat transfer parameters, contact conductances, and modelling environmental heating loads are covered. TOLIMAN will use a “cold biased” thermal design, and so Chapter 4 covers the passive thermal control approach. This begins with the proposed passive control design, followed by a study of the potential orbits in detail using an orbit propagator created in MATLAB to understand the mission’s thermal environment. Worst case orbital conditions are determined using optimisation routines, and the chapter concludes with parametric studies and a sensitivity analysis to assess the effect of uncertain design variables on thermal performance. Chapter 5 describes the active heater control approach, beginning with the determination of optimal heater placement, followed by

the creation of a state space reduced order model for the design of a feedback controller. This feedback controller is simulated in all orbital worst cases. Finally, Chapter 6 integrates thermal models with structural finite element models in order to assess the mechanical stability of the telescope. The work in this chapter enables further optical analysis to be performed by the rest of the TOLIMAN team, but which lies beyond the scope of this work.

Literature review

2.1 The TOLIMAN Mission Concept and Other Space Telescope Missions

2.1.1 The TOLIMAN Space Telescope

To date, over 4000 exoplanets have been discovered [14], and the rate of exoplanet discovery is increasing. This fact however masks a blind spot in current observation techniques in that some degree of luck is required for the bulk of these detections. Most often, the discovery is made due to the orbital alignment of an exoplanet with its host star and the Earth, meaning that a transit across the face of the star can be measured by a dip in the stellar light. Also, this type of detection is biased towards planets with short orbital periods due to the smaller required observational times, resulting in a higher rate of observations. As a result, we have little knowledge on the planets which may orbit stars in our galactic neighbourhood of 32 light-years [3].

The TOLIMAN mission will use astrometric detection to measure the changes in stellar positions due to orbiting planets. The idea behind astrometric detection is depicted in Figure 2.1. The gravitational “tugging” from a planet orbiting its host star results in reflex motion of the star. With sufficiently accurate measurements, the relative angular deflection between the binary stars can yield a solution for the planet’s mass and orbit. A more detailed comparison is given by Tuthill et al. [3], but in summary astrometric detection gains greater sensitivity for F, G, and K-type stars compared to Doppler Radial Velocity for the detection of true Earth analogs.

The angular deflection induced by Earth-like exoplanets is very small, on the order of one micro-arcsecond. The TOLIMAN space telescope is built around a novel optical architecture

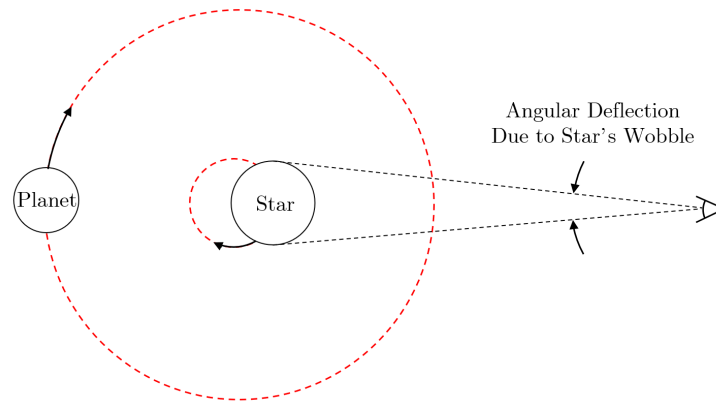


FIGURE 2.1. A star and planet orbiting each other will orbit about the centre of mass of the system – the source of reflex motion for the host star.

which is a reformulation of the diffractive pupil [3, 4, 5]. The idea is that a complex pattern of phase offsets, the diffractive pupil, is incorporated as part of an optical element with a telescope with extreme mechanical stability. This pupil diffracts the incoming starlight to form a pattern whose features are known, and as long as the pupil remains stable, these features can be used to make measurements. TOLIMAN reformulates the pupil to have a sharp, fine featured structure particularly in the radial direction (Figure 2.2). The fringes produced by diffraction are used as a “ruler” to measure the relative deflection between binary stars over time. Instead of measuring the deflection in terms of pixels on a sensor, the deflection is measured by using the fringes as the “grid”. The result is the ability to measure the astrometric deflection of binary star systems at the sub-micro-arcsecond level. Mechanical distortions in this pupil mean that the “ruler” is distorted, adversely affecting the measurements made; hence mechanical stability is critical to mission success.

2.1.2 Past Small Space Telescope Missions

The TOLIMAN mission concept is not only unique from a scientific point of view, but also with regards to the engineering tolerances and requirements imposed on the science payload (and indeed the rest of the spacecraft). Over the past few decades, there have been a number of highly successful large space telescope missions such as Hubble, Chandra, Spitzer, with the soon to be successor to Hubble, the James Webb Space Telescope (JWST) slated for imminent launch. However, smaller space telescopes of the scale of TOLIMAN are relatively few. Shkolnik [6] and Genet et al. [7] highlight that there are several gaps that smaller sized telescopes could fill in astronomical research. As the miniaturization of technology advances, the cost of

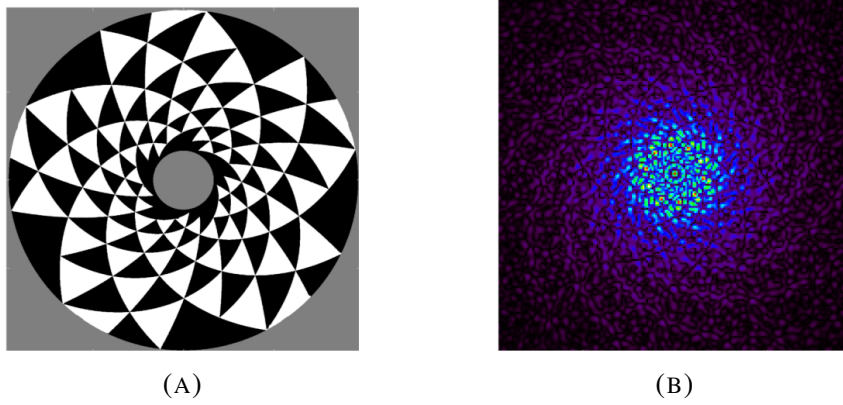


FIGURE 2.2. Tuthill et al. [3]. Candidate point-design pupil for the TOLIMAN mission (A). The black/white regions indicate discrete phase steps of $0/\pi$, resulting in a diffraction pattern due to relative differences in optical path length that light takes. The monochromatic Point Spread Function (PSF) generated (B) has a complex/fine featured pattern which extends radially from the centre, creating many sharp fringes.

these small telescope missions will decrease, especially with regards to the CubeSat standard. Furthermore, these smaller telescopes, which are generally developed by universities or smaller science programs (as is the case with TOLIMAN), have the advantage that their instruments do not need to be shared among many research groups. Moreover, they are able to focus on more specific problems, whereas large programs need to appeal to a large range of research areas. Of course, these smaller telescope missions will not replace the need for larger missions such as JWST, but they provide more accessible and more agile complimentary approaches to accomplishing unique science. This review will focus on past and current small space telescope missions, their technical challenges, how they were overcome, and lessons/implications for the TOLIMAN program.

Discussion begins with the Arcsecond Space Telescope Enabling Research in Astrophysics (ASTERIA), a 6U CubeSat developed at the NASA Jet Propulsion Laboratory (JPL) in collaboration with the Massachusetts Institute of Technology (MIT). It was designed with the primary mission objective of demonstrating precision pointing and thermal control in a nanosatellite platform [15, 8]. The optical telescope was a 5-element refractive design. It and the payload electronics occupy approximately half the internal volume of the satellite. The mission has successfully met its Level 1 Requirements by demonstrating optical line-of-sight stability of 0.5 arcsecond RMS, and focal plane temperature stability better than ± 10 mK over 20 minutes [16]. In fact, on a per-unit-mass basis, ASTERIA boasts better pointing stability than

most major missions (e.g. JWST, Kepler, Transiting Exoplanet Survey Satellite (TESS), and the Characterising Exoplanets Satellite (CHEOPS)), [17]. Also on a per-unit-mass basis, the focal plane temperature stability of ASTERIA is better than much larger programs (Kepler [18], Wide-Field Infrared Survey Explorer (WISE) [19], TESS [20], and CHEOPS [21] to name a few). Although the payload differs optically and geometrically from that proposed for TOLIMAN, ASTERIA was a pathfinder for demonstrating that extremely high pointing and thermal stability can be achieved on the nanosatellite platform with size constraints comparable to those for TOLIMAN. Details on how ASTERIA achieves this is discussed in Section 2.3.

Much more similar in design to TOLIMAN is the Star-Planet Activity Reserach Cubesat (SPARCS), a 6U CubeSat based mission devoted to the study of ultra-violet (UV) time-domain behaviour in low mass stars [9]. The SPARCS telescope uses a 9 cm RC telescope, which is designed to function over the temperature range -40°C to 60°C , while having dimensional stability down the optical axis of 0.14 mm [22, 23]. It is deployed in a Sun-Synchronous Orbit (SSO). Compared to TOLIMAN and ASTERIA, stability requirements are not as stringent, largely due to differing mission objectives and greater mass constraints. Nonetheless, interesting design decisions regarding material selection were made to achieve high dimensional stability at a broad range of temperatures using a passive thermal design.

The Japan Astrometry Satellite Mission for Infrared Exploration (JASMINE) is a set of two satellite missions by the Institute of Space and Astronautical Science (ISASS)/The Japanese Aerospace Exploration Agency (JAXA). It aimed to study the core of the Milky Way by measuring the distance and motion of stars 26 thousand light-years away with high-precision astrometry observations in the near-infrared band (NIR) [24]. The missions consist of a small 35 kg satellite, called Nano-JASMINE [25], and Small-JASMINE a larger 400 kg satellite [26]. The focus here will be on the Small-JASMINE telescope, which has more publicly available information and is more akin to the TOLIMAN mission design. It will simply be referred to as JASMINE hereafter. JASMINE's telescope was a modified Korsch system with a 30 cm primary mirror – just over twice the diameter of TOLIMAN – and also with very stringent stability requirements. JASMINE's structural thermal stability requirements were to have no more than a 10 nm change of the stellar positions on the focal plane, which meant keeping the telescope structure within 278 ± 0.1 K [24]. Preliminary estimates for the TOLIMAN mission give similar thermal stability requirements, but with a smaller RC telescope, and a lower cost.

To meet these requirements, JASMINE had developed an advanced version of the material “Super Invar”, called “Super Super Invar” [24, 27].

There are a number of other small telescope missions which offer valuable inspiration in the design of small, highly thermally stable telescope systems. Some examples include the Microvariability and Oscillations of Stars (MOST) satellite, launched in 2003 by the Canadian Space Agency (CSA), which was the first “small” telescope with a 15 cm aperture, on a 54 kg spacecraft [28, 10] - almost identical in mass and volume to TOLIMAN. It was the smallest telescope for 10 years until the Bright Target Explorer (BRITE) constellation was launched in 2013. BRITE was a group of six 10 kg nanosatellites, with 30 mm aperture telescopes [29]. Emphasis was placed on ASTERIA, SPARCS, and JASMINE due to their various similarities with the proposed TOLIMAN mission in terms of thermal management and structural design of their telescopes. TOLIMAN is a mission which differs greatly from these, however, the many design choices made in these missions are useful to inform the thermal-structural scheme employed for the payload.

2.2 Basic Theory of Thermal Heat Transfer

2.2.1 Thermodynamics Basics

For the analysis presented hereafter, there are two main concepts in thermodynamics which are essential to understand. These are *energy conservation* and the *heat capacity of solids*. There are many ways to state energy conservation, or the first law of thermodynamics, but the most convenient here is as a rate equation [30]

$$\dot{E} = \dot{E}_{in} - \dot{E}_{out} + \dot{E}_g, \quad (2.1)$$

where $\dot{E}$ is the rate of energy change of the system, $\dot{E}_{in}$ is the rate of energy being introduced to the system, $\dot{E}_{out}$ is the rate of energy leaving the system, and $\dot{E}_g$ is the rate of energy generated within the system. A topical example application of this equation may be that $\dot{E}_{in}$ may be incident heat flux on a body due to the sun, while $\dot{E}_{out}$ is the heat leaving the body due to emitted radiation, and $\dot{E}_g$ is the internally generated heat from a heater. $\dot{E}$ is then the change in energy of the body over time from temperature can be calculated.

Heat capacity relates the amount of energy required to increase the temperature of an object, while the specific heat capacity is the heat capacity per unit mass. The heat capacity of a substance depends on whether energy is added to it while keeping the pressure or volume constant. Here, only solids are considered, in which they are assumed to be incompressible. In this case, the specific heat at constant volume and constant pressure can be assumed to be the same. Hence, we have the specific heat capacity of a solid c_p with the following definition [30]

$$c_p = \frac{1}{m} \frac{\partial E}{\partial T}, \quad (2.2)$$

where m is the mass of the solid, E is the energy, and T is the temperature.

2.2.2 Conduction

In space, conduction and radiation are the two modes of heat transfer. Heat conduction is the movement of heat through a solid, occurring due to the neighbouring interactions between adjacent atoms. The most fundamental law in conduction is Fourier's law [31]

$$Q = -kA\nabla T, \quad (2.3)$$

where k is the thermal conductivity of the material, A is the cross-sectional area which heat flows through, T is the temperature of the solid, which is a function of space, and Q is the heat. Equation 2.3 states that the heat flux is proportional to the temperature gradient and thermal conductivity.

In the special case of 1D heat transfer, with no internal energy generation, Fourier's law gives a useful analogy between heat transfer and electric charge. Just as electrical resistance quantifies the conduction of electricity, thermal resistance describes the conduction of heat. In this case, the driving potential is the temperature difference, which gives rise to the flow of heat. Figure 2.3 depicts this analogy, where the thermal resistance is defined as the ratio of driving potential to heat flow

$$R = \frac{T_2 - T_1}{Q} = \frac{L}{kA}, \quad (2.4)$$

where R is the thermal resistance, and the second equality is true for an object with constant cross-sectional area A . One can also define the thermal conductance (not to be confused with conductivity), which is the inverse of thermal resistance. However, more commonly, the conduction coefficient is used, which is the conductance per unit area. In this work, the

conduction coefficient is denoted by h . It is often useful to introduce an effective thermal resistance for an entire component when temperature distributions are not required; this dramatically improves the computational cost and complexity of thermal models.

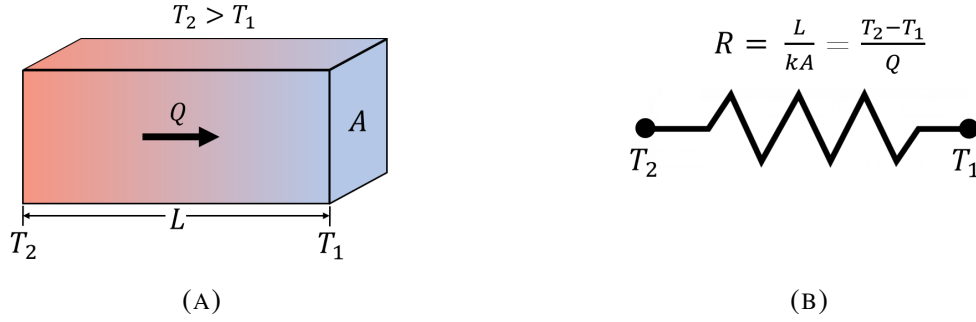


FIGURE 2.3. Heat flows from the high to low temperature regions according to Fourier's law (A), if there is no internal heat generation and material properties are homogeneous, an analogous 1D thermal resistance can be defined (B) for the object.

Transient heat conduction through an object is governed by the heat diffusion equation (or just the heat equation), and can be derived from Fourier's law and the conservation of energy [31]. In 3D, it is given by

$$\frac{1}{\alpha} \frac{\partial T}{\partial t} = \frac{\partial^2 T}{\partial x^2} + \frac{\partial^2 T}{\partial y^2} + \frac{\partial^2 T}{\partial z^2} + \frac{\dot{q}}{k} \quad (2.5)$$

for constant thermal conductivity k , where $\alpha = k/\rho c_p$ is the thermal diffusivity, and $\dot{q}$ is the heat input per unit volume. This Partial Differential Equation (PDE) can only be solved analytically for a number of simple cases, so is generally solved numerically for real-world geometries and heating cases.

2.2.3 Contact Resistance

When objects come into contact, the heat transfer across the interface is less than the heat transfer that would occur in a continuous medium. This decrease in heat transfer is described by the thermal contact resistance. Figure 2.4 (b) shows microscopically the contact between two real rough surfaces. The actual surface area of the contact is reduced, so the thermal resistance is increased. Heat can also travel via the gas in the gaps between the two contacting solids. In space, where pressure of the surrounding gas is extremely low, this can be neglected.

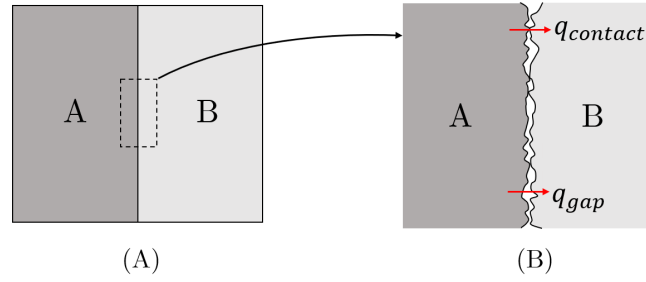


FIGURE 2.4. Contact between two objects (A). Microscopically, the contact is not perfect due to roughness of the two surfaces, and contact is only achieved in certain locations, heat can travel through the contact points $q_{contact}$, or through the gas (if present) between q_{gap} (B).

The conduction coefficient of the contacting joints is in general very difficult to predict and depends on the material and surface properties of the contacting materials, as well as the local contact pressure. Various theoretical, empirical, and semi-empirical models exist for different contact joints, whether they are bonded, bolted, or uniform pressure [32, 33, 34]. Such models will be discussed in Chapter 3, where the spacecraft/telescope thermal model is described.

2.2.4 Radiation

Alongside conduction, radiation is the predominant method of heat transfer in space. Radiant heat transfer occurs both between the satellite and its environment (due to heating from the Sun and Earth), and between the spacecraft components. In all analysis here, it is assumed that the radiation behaves as a ray. Firstly, referring to Figure 2.5, we define the spectral radiation intensity I_λ , as the energy emitted per unit time dq , at wavelength λ , per unit wavelength interval $d\lambda$, in direction (θ, ϕ) , per unit area of emitting surface dA_1 , per unit solid angle $d\omega$ about this direction [31]

$$I_\lambda(\lambda, \theta, \phi) = \frac{dq}{dA_1 \cos \theta d\omega d\lambda}. \quad (2.6)$$

With this, we can define the spectral emissive power E_λ , which is the energy per unit time dq , at wavelength λ per unit wavelength interval $d\lambda$, per unit emitting surface area dA_1 , in all directions. The emissive power is calculated by integrating Equation 2.6 over a hemisphere, and therefore represents the total energy emitted into a hypothetical hemisphere surrounding the area dA_1 . For the above quantities, it is often useful to look at the total radiation intensity, or

total emissive power, over all wavelengths. This is denoted by I , and E , where the λ subscript is removed. The total emissive quantities are found by integrating over all wavelengths $d\lambda$.

In many cases, it is a reasonable approximation to assume the intensity of the emitted radiation is independent of the direction, such a surface is called a *diffuse emitter*. With this approximation, the following relationship between emissive intensity and power can be made

$$E = \pi I, \quad (2.7)$$

which also holds true for the spectral quantities I_λ and E_λ .

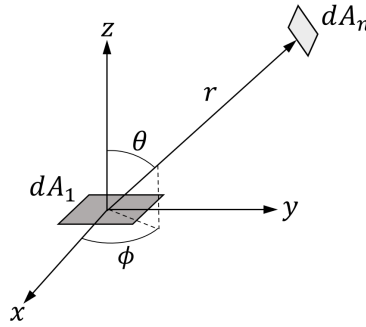


FIGURE 2.5. Schematic of surface dA_1 emitting radiation to surface dA_n , which is normal to the emitted rays, the z is the normal to the surface dA_1 .

The most fundamental concept in radiant heat transfer is that of a black body. By definition, a black body is a perfect absorber, and hence a perfect emitter [35]. The spectral intensity of a black body is well known and is given by the Planck distribution, which is not be stated here however details can be found in Seigel and Howell [35]. The Stefan-Boltzmann law gives an expression for the total emissive power of a black body, and is given by

$$E_b = \sigma T^4, \quad (2.8)$$

where $\sigma = 5.670 \times 10^{-8} \text{ W/m}^2/\text{T}^4$. True surfaces are not perfect emitters or absorbers. For the analysis here, there are two quantities which describe the radiative properties of real surfaces which are most important. These are the emissivity and absorbtivity, which quantify how well a surface emits or absorbs radiation relative to a black body. In general, these properties will depend on the temperature of the emitting body, as well as the direction and wavelength of the emitted radiation. However, for most engineering calculations, the direction and temperature dependence is neglected, and averaged quantities are used. The spectral hemispherical intensity is defined as the ratio of the emitted radiation from a surface to that of a black body, at the

wavelength λ

$$\varepsilon_\lambda = \frac{E_\lambda}{E_{\lambda,b}}.$$

Similarly, the spectral hemispherical absorptivity is defined as

$$\alpha_\lambda = \frac{G_{\lambda,abs}}{G_\lambda},$$

where G_λ is the irradiation, and is the total incident radiation energy per unit time, per unit area, and $G_{\lambda,abs}$ is the total irradiation absorbed by the body. This absorbed radiation is then the energy which contributes to the heating of the body. As before, an average emissivity and absorptivity can be defined over all wavelengths, or over a subregion of wavelengths.

2.3 The Design of Thermally Stable Space Telescopes

2.3.1 Effect of Orbit Geometry

When designing an optical telescope for use in space, thermal stability is generally one of the greatest challenges. Changes in the thermal environment of the telescope result in dimensional changes of the metering structure, thereby affecting the mirror figures and alignment, in turn impacting the optical performance. As Feinberg et al. [36] points out, the thermal environment for a space telescope is largely determined by the chosen orbit. In a Low Earth Orbit (LEO), for example, the spacecraft will undergo thermal swings throughout its orbit due to eclipse periods of the Sun. Furthermore issues of stray light and communication scheduling are associated with these orbits. Encompassing this full complexity is not the focus of this work. There is always some trade between science performance and cost which is often a key deciding factor on the choice of orbit. Open telescope missions such as the JWST orbit in Lagrange Point 2 (L2) with a large sun shade to avoid these issues [37]. The TOLIMAN mission will be in LEO, with candidate orbits being a low inclination orbit, or a Sun-Synchronous Orbit (SSO). Hence, the design must consider the thermal challenges of these orbits and explore the design trade space opened.

Solar eclipse periods are the biggest thermal concern impacting orbit choice. One useful parameter for thermal analysis of a spacecraft is the beta angle, β , which describes the relative orbit orientation with respect to the sun [38]. It is the angle between the solar vector $\hat{S}$, which

is the vector from the Sun to the Earth, and the plane of the orbit. This is illustrated in Figure 2.6. The beta angle will depend on the orbital parameters and the position of the Sun relative to the Earth.

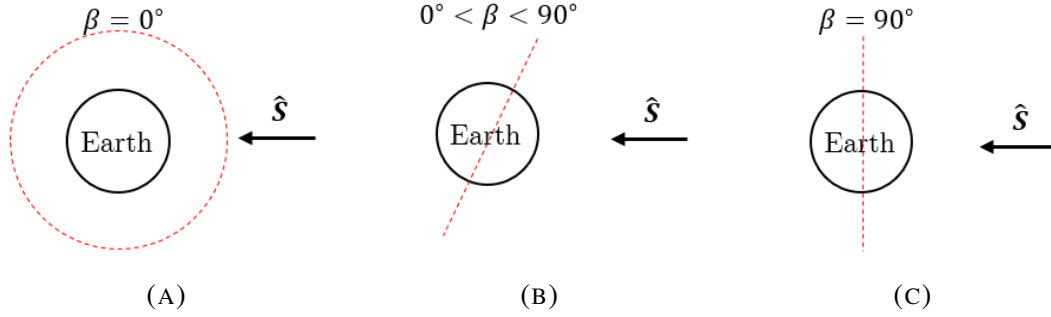


FIGURE 2.6. Illustration of various beta angles, where $\hat{S}$ is the solar vector – the unit vector pointing from the sun to the Earth. Beta angles for $\beta = 0^\circ$ (A), $0^\circ < \beta < 90^\circ$ (B), and $\beta = 90^\circ$ (C) are shown.

The study by Sumanth [39] uses the JPL DE405 Ephemeris data to compute the eclipse duration of small satellites in LEO. The results show that the eclipse fraction of a satellite can vary from an average of 0.4 at a 300 km LEO, low inclination orbit, to 0.29 at 1500 km. Furthermore, this eclipse period will vary throughout the year, with two maxima and two minima of the same magnitude each year. Variation of inclination angle has little effect on the maxima of eclipse duration, however, higher inclination orbits will spend less time in the eclipse away from these periods of maximum eclipse. Similar results were obtained by Ismail et al. [40]. This means with higher inclination orbits, at certain times of the year, there will be little to no eclipsing. From a thermal point of view, thermal swings are reduced thus yielding more stable satellite temperatures throughout the orbit, but with higher overall temperatures.

The report by Smith [15] on the on-orbit results from the ASTERIA spacecraft explains the effect of the orbit geometry on the thermal performance of the spacecraft. ASTERIA was deployed from the International Space Station (ISS) (altitude of approximately 400 km and inclination of 51.6°). The beta angle varied between 0° and 75° over several weeks due to orbital precession, the result is eclipse fractions varying between 0.39 and 0. To mitigate stray light, observations were only made during eclipses. During these eclipses, the lack of sunlight meant a decrease in temperature on board the satellite, which required management by the thermal control system. Periods of low eclipse result in higher component temperatures due to the constant incident solar flux, which can also be problematic in terms of thermal control.

The SPARCS telescope mission, on the other hand will be launched in a SSO terminator orbit, with eclipse periods being estimated as approximately 25 minutes per year [9]. This allows for longer observation periods that will have to be interrupted periodically for slew maneuvers to point the radiators to space for thermal management. Similarly, the JASMINE satellite is placed in SSO. [24]. The JASMINE mission, however, is only able to image during the spring and autumn seasons (northern hemisphere) due to the positioning of the sun. Figure 2.7 illustrates the orbital plane with respect to the sun as the Earth rotates around the sun. During the winter, the Sun is in the line of sight of the target, so observations cannot be made. During the summer period, the thermal radiator situated on the rear of the payload would be facing the sun, and so the thermal environment is not suitable. During the spring and autumn periods, the radiator can face deep space, allowing it to radiate the necessary amount of heat to maintain thermal requirements.

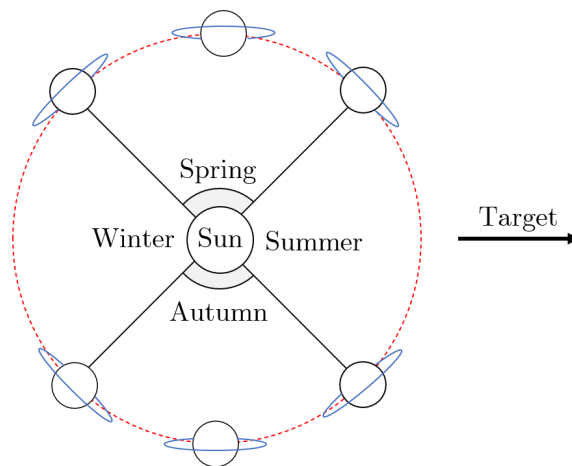


FIGURE 2.7. Spacecraft orbital plane (blue) relative to the sun during the Earth's orbit (red) around the sun for the JASMINE mission. Observations of the primary target were only made during the Spring and Autumn seasons due to positioning of the sun during the year.

Clearly, there is no “perfect” orbit which is optimal for both scientific observation and thermal control. What might be better for thermal control may not work in favour of the scientific objective. The examples of ASTERIA, SPARCS, and JASMINE illustrate there are always trade-offs to be made. For any orbit chosen, there will be a window where the thermal requirements can be met, and observations of the target can be made. Determining these periods, and designing around them is one of the challenges of a successful mission. Indeed, this is the case for TOLIMAN. The primary target for TOLIMAN, Alpha Centarui, results in a

situation very different to the ones discussed above, and hence the mission must develop its own way of dealing with stability requirements.

2.3.2 Thermal Control Technologies

This section initiates an exploration of thermal management that pertains to the design of the spacecraft and the payload itself. Bely [41] defines “athermalization” as the design of an optomechanical system such that changes in bulk temperature do not affect the optical performance. Note that bulk temperature changes do not refer to temperature gradients in the system. Bely puts the methods of athermalization into three categories:

- **Passive:** The materials or optics and components are chosen such that the system is insensitive to temperature.
- **Passively Compensated:** The alignment is maintained by passive systems using thermal expansion itself to compensate for temperature effects.
- **Actively Compensated:** Sensors and electromechanical actuators maintain alignment using feedback control loops.

For many missions, some combination of the above is used. Actively compensated optics are not of concern here, as they are more suited to larger telescope missions with segmented mirrors, such as the JWST. Therefore, by only using passive means, athermalization of a structure becomes largely an issue of material selection. An example of passive compensation is offered by the Spitzer Space Telescope [42], where the expansion of the telescope tube compensates for the change in the radius of curvature (and hence focus) of the optics. Indeed, the all-SiC telescope manufactured by AOS used on board TOLIMAN is an athermal design. This solution, however, will not compensate for changes due to temperature gradients. This is accounted for by thermal control, where the aim is to prevent changes in temperature and eliminate temperature gradients across the structure. Thermal control can be done passively, through coatings, insulation, and radiators to control the amount of external heat input and heat lost, or actively by using heaters.

Thermal control technologies used in spacecraft are based on the types of heating environments a spacecraft encounters. Meseguer et al. [38] outlines the different thermal environments a spacecraft may encounter throughout its lifetime, this includes the launch and decommissioning

of the spacecraft. In the design of this telescope, and the focus of this thesis, the concern is on-orbit performance only. In space, the only mode of heat transfer to or from the external environment is through radiation heat transfer. The different sources of heat transfer are summarised in Figure 2.8 [33].

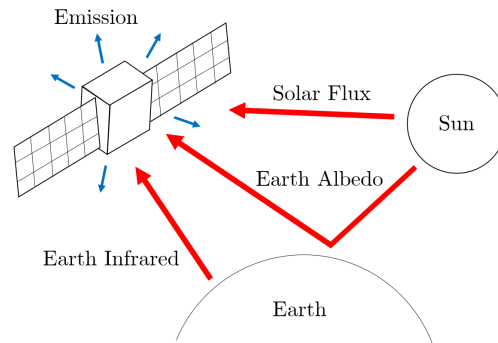


FIGURE 2.8. Heat transfer that occurs between a spacecraft and its environment. Flows include direct heat flux from the sun, reflected solar heat flux from the earth (albedo), and planetary radiation emitted from the Earth's surface and atmosphere. The surface of the satellite will radiate heat back to the environment.

There are many forms of passive and active thermal control technologies currently available and under development. Different mission requirements will call for the use of different technologies. Here, only a brief outline of some of these technologies which may be most useful to TOLIMAN will be given. A more detailed discussion of current thermal control technologies is given by Miao et al. [43] or Gilmore [33].

There are two ways in which passive thermal control techniques work. Firstly, they control the amount of heat which enters or leaves the system from the environment. Secondly, they transfer/distribute heat around the system from one place to another. Multilayer Insulation (MLI) is one of the most commonly used passive thermal control technologies on spacecraft. Like any insulation, MLI minimizes the flow of heat between a system and its surroundings, thus reducing the amount of energy required to maintain it at a given temperature [44]. A basic diagram demonstrating the operation of MLI is shown in Figure 2.9. MLI uses layers of low emittance films, usually thin polymer films with vapor-deposited metal on one or both sides. Each layer will typically reflect 90% - 99% of radiation, and the result is a barrier which is nearly 100% effective. Being used in the near vacuum of space, well-designed MLI will have very little conductive heat transfer. As a result, when modelling MLI mathematically, the conductivity is neglected, and an effective emissivity ϵ^* is defined, which describes the

heat transfer according to a radiative transfer between the two outer layers. The guidelines by Finckenor and Dooling [45] give a thorough treatment of the material selection, design, and layering of MLI blankets for use on spacecraft.

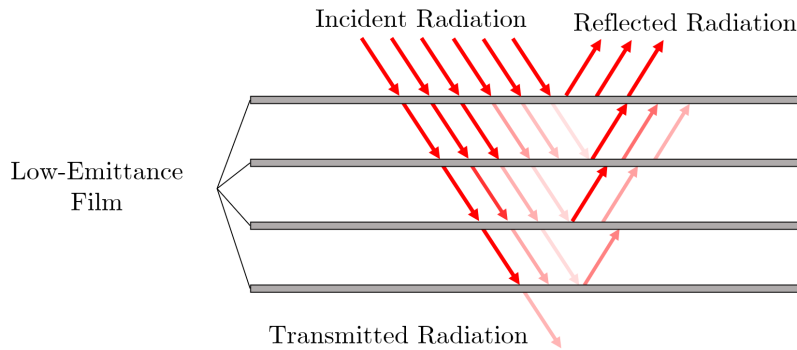


FIGURE 2.9. The basic operating principle of MLI. Incident heat is continually reflected by layers of low emissivity film, resulting in very small heat transfers.

Since radiation is the only way in which heat can enter or leave a spacecraft, the surface optical properties dictate the thermal coupling of a spacecraft to its thermal environment. By choosing surface properties carefully, a spacecraft can be made to emit or absorb light at certain rates to control the amount of heat flux in or out of the system. In general, surface optical properties can be put into three categories [44]. Firstly are solar reflectors, which have low absorptivity, but high emissivity; second, are flat coatings, in which the surface reflects and absorbs the incident radiation nearly equally; and finally solar absorbers, which have high absorptivity relative to their emissivity. An extensive list of coatings, their properties, and mathematical descriptions is given by Kauder [46] and Gilmore [33]. Surface coatings may also have secondary purposes such as glint prevention, electrical grounding, stress handling, and micro-meteoroid protection.

An important consideration for coatings (and less so MLI) is the degradation of thermal properties [33]. These coatings are affected by charged particles, ultraviolet (UV) radiation, and high vacuum. The result is an increase in solar absorptivity and little effect on the infrared (IR) emittance. In terms of thermal control, this means any radiating surface must be oversized to meet the end-of-life requirements for a mission. The oversized radiators then mean that the components may run much cooler in the early years of the mission. The NASA Long Duration Exposure Facility (LDEF) was launched in 1984, and recovered in 1990. It contained over 10 000 specimens to test the effects of the LEO space environment on materials, components and systems. As an example, A276 paint, commonly used on radiator coatings, with an initial absorptivity of 0.282 was measured to have an absorptivity of 0.552 in regions that were

exposed to incident solar radiation, but had not degraded at all due to atomic oxygen erosion [47]. Anvari et al. [48] numerically modelled the effect of the degradation of A276 paint on a small cubical satellite which used the coating on its radiating surfaces. For the case of their small satellite, they found that temperatures varied over 10°C from start to end-of-life. While material degradation studies are a critical design consideration in thermal control, they are beyond the current aims of this work.

Materials with high thermal conductivity can be used to transport heat around a spacecraft system. These thermally conductive materials can be incorporated into the existing structure of the telescope, or, where this is insufficient, thermal straps may be added [38]. Such strategies can transfer heat rapidly around the system. Metallic materials such as aluminium and copper can be used, or non-metallic materials such as aluminum-based SiC or graphite. The choice depends on the mass and conductivity requirements. One important consideration for the use of conductivity to manage the flow of heat is an understanding and analysis of thermal contact resistance.

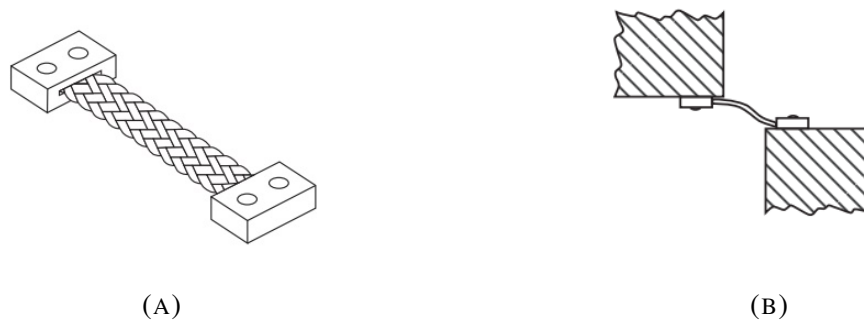


FIGURE 2.10. from Meseguer et al. [38], a typical thermal strap (A). One end is bolted to the hot surface, and the other is bolted to the cold surface, heat is transferred between the two ends via strap of high thermal conductivity. Typical mounting (B).

In ideal situations, the thermal control of a telescope (or any spacecraft component) would be done entirely passively. In many cases however, large fluctuations in the thermal environment, or strict requirements on the operating temperatures of the components mean that active thermal control measures are needed. Here, only active thermal control via heating is explored. There are many heater types which can be used on a spacecraft, but the most common type of heater is a patch heater [33]. These heaters consist of a resistive heating element sandwiched between an electrical insulator, such as Kapton. The advantage of patch heaters is that they are flexible and

can be made in different shapes. Other heaters such as armored electrical heaters or ceramic-metal integrated heaters exist for applications which require very high operating temperatures or large power densities [43]. In order to use a heater to maintain a required temperature set point, feedback control must be implemented.

2.3.3 Examples of Thermal Control in Small Space Telescope Missions

We now examine the thermal control methods that some of the previously described space telescope missions have employed, starting with ASTERIA. A diagram illustrating the thermal control on ASTERIA is shown in Figure 2.11. The thermal control on the ASTERIA payload was accomplished using a multi-layered approach. Firstly, the entire telescope and sensor assembly was thermally isolated as much as possible from the rest of the spacecraft. Besides electrical wiring, the only connection was through the titanium bipods. It is estimated that the total conductance of the bipods is 0.015 W/K.

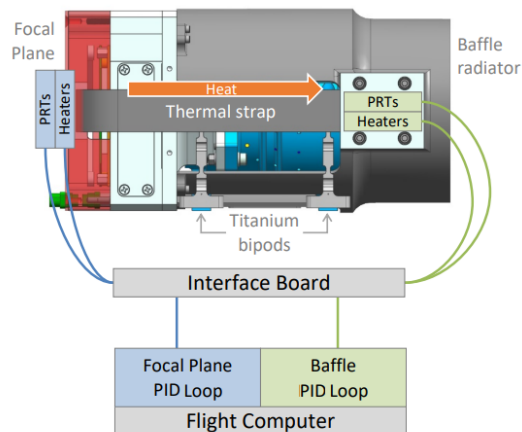


FIGURE 2.11. Knapp and Seager [8]. Thermal control diagram for the ASTERIA telescope assembly.

The imaging sensor is shown in the red region of Figure 2.11, and produces approximately 1 W of power. The heat produced by this imager is transported to the telescope baffle using a single side mounted thermal strap. This baffle then acts as a radiator to space. To achieve precise temperature control, the assembly has two active thermal control loops. Both control loops use a Proportional-Integral-Derivative (PID) controller for temperature control [49]. Two independent control loops are used, one “coarse” control loop at the baffle, where the main source of thermal disturbance originates, and one “fine” control loop at the imager to provide a more precise thermal control at the focal plane. The temperature is measured using Platinum

Resistance Thermometers (PRTs), and patch heaters are used for control. The result is a temperature control of ± 0.5 K at the baffle and ± 0.01 K at the imager over a 20 minute period. ASTERIA is a good example of using both passive and active control techniques to achieve very high thermal control of the telescope. To note is that a single heater on one side of the baffle was used, likely due to the high conductivity of the baffle and its small size. The use of a metal tube is advantageous in that it gives a short, high conductivity (low thermal resistance) path for heat to transfer anywhere around the telescope. ASTERIA is able to achieve such high degree of stability in a small volume because it makes efficient use of existing structures (such as the baffle) and heat sources (the imager) on the telescope assembly as a first layer of thermal control.

The JASMINE mission, on the other hand, with its larger telescope and greater mass and volume budgets, took a different approach. The main structure of the payload is shown in Figure 2.12. The JASMINE telescope uses an Invar truss metering structure, as opposed to a tube assembly like ASTERIA. This is typical for larger telescopes due to mass savings. The design uses a long thermal hood and a sun shield for blocking stray light. The imaging sensor on the JASMINE telescope is required to operate at a temperature less than 180 K. To do this, a peltier cooling system is used. The cold end of the peltier system is attached via a heat pipe to a radiator external to the telescope box. For this system to operate correctly, the radiator must be directed to deep space to allow sufficient heat to flow. It is for this reason that imaging science can only occur during the Spring and Autumn seasons as described in Section 2.3.1.

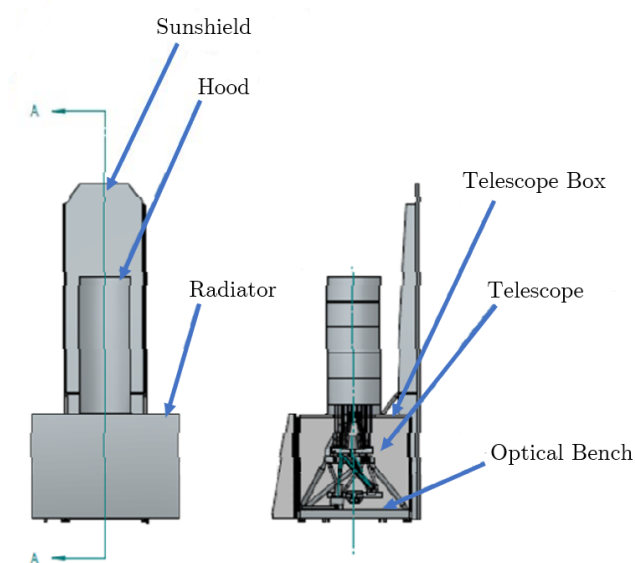


FIGURE 2.12. Utsunomiya et al. [26]. Structural design of JASMINE telescope.

JASMINE requires that the distance between the primary and secondary mirror should change by less than 10 nm. Along with the Invar material used, this also requires active thermal control to maintain temperature fluctuations below 1 K over a 50 minute period. This is done by heating the walls of the telescope box shown in Figure 2.12. Unlike ASTERIA, the telescope structure is not being heated directly. Since a truss structure is used, it is more efficient to heat the walls around the telescope, which then in turn heat the telescope structure through radiation. This type of control relies on the fact that the telescope box is well insulated to ensure the heated walls and the telescope structure have strong thermal coupling. Thermal simulations showed that the telescope structure is kept within 0.1 K if the walls were maintained within 0.2 K of a target temperature. Like ASTERIA, the telescope structure makes minimal physical contact and is well isolated from the rest of the spacecraft.

The SPARCS telescope assembly is shown in Figure 2.13. The thermal requirements on the SPARCS metering structure were much less demanding, both due to the use of low Coefficient of Thermal Expansion (CTE) materials and the science requirements. The optical assembly can maintain the focus between temperatures of -40°C and 60°C . More critical temperature requirements are placed, however, on the detector and a dichroic placed behind the primary mirror. To reduce the dark noise from the observed photometer, the detector must be kept at 238 K. This is achieved with a deployable 1200 cm^2 radiator and a peltier cooler. The optical assembly is mounted on a single post at the centre of mass of the optical assembly, again, to thermally isolate it from the rest of the spacecraft.

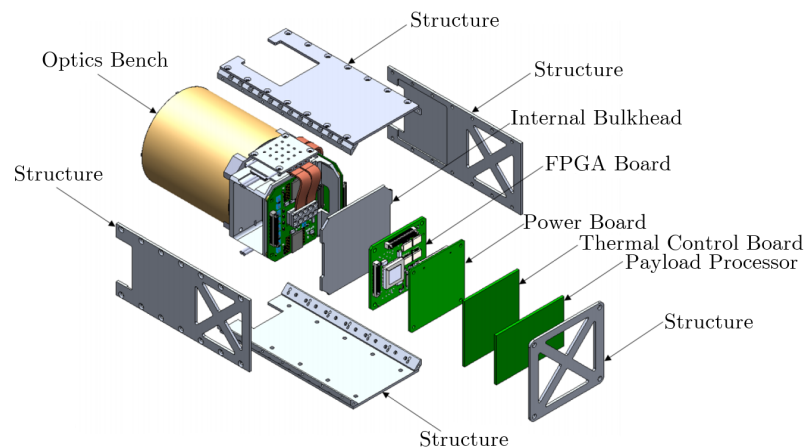


FIGURE 2.13. Scowen et al. [23]. Payload assembly for the SPARCS telescope. The payload occupies 3U volume within the satellite.

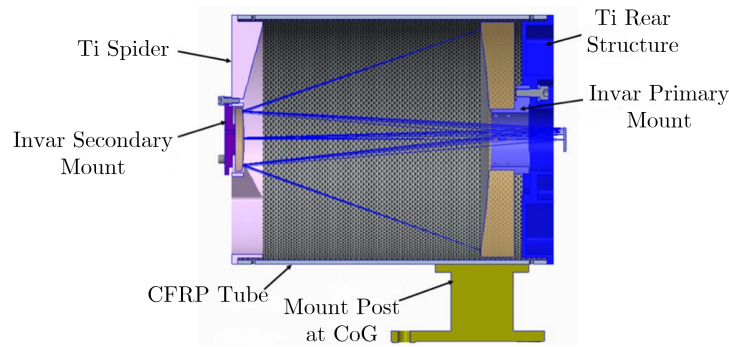


FIGURE 2.14. Scowen et al. [23]. SPARCS telescope tube assembly with materials labelled for each component.

The main tube for the SPARCS telescope is made from Carbon Fiber Reinforced Polymer (CFRP) composite. CFRP is anisotropic in that it can be laid up in such a way that the CTE is nearly zero in a single direction, but larger in others. In the case of SPARCS, the CTE was $0 \times 10^{-6} \text{ K}^{-1}$ (zero) in the axial direction, but $20 \times 10^{-6} \text{ K}^{-1}$ in the radial direction [23]. As a result, Ti6Al-4V was used for radially mounted parts such as the spider to athermalize the system.

In these examples, passive means of thermal control were used as much as possible, this came in the form of heat straps, coatings, and radiators. Active control was only used to achieve very accurate operating temperatures. This active thermal control was only used on components which had these stringent requirements. Also, the telescope payload was thermally isolated as much as possible from the rest of the spacecraft, allowing it to be controlled as its own separate system. This makes thermal control much simpler as the complex, high thermal inertia environment of the rest of the spacecraft no longer has a great influence. TOLIMAN's requirements are unique compared to all of these missions and so will require its own innovative methods for thermal control, however there is undoubtedly much to learn from exploration of these prior designs in this space.

2.4 Methodologies for Thermal Analysis and Design

For optical payloads, thermal control is often a much more significant consideration driving the design process. Hence, the approach to thermal design for TOLIMAN will differ from that typically seen for small satellite missions. Panczak and Cullimore [50], and Martin [51] explain that traditionally, thermal analysis is done using a point design approach. This typically

involves building a thermal design around existing mechanical designs using a trial and error approach. This is a time-consuming process and generally will not lead to an optimal design in terms of performance.

There are a number of reasons why this approach has typically been taken in the past. First, is the uncertainty of the initial design and the effect of design changes on results. For example, the relocation of components and changes in structural features are rarely a great concern from a mechanical point of view, however, the precise thermal implications of such a change may not be clear. The second is due to the time-consuming nature of the development of thermal models. Rapidly changing mechanical designs mean more time will be spent creating thermal models than performing analysis. Engineers may build a model of a candidate design, solve for the temperatures, and compare it to a set of requirements. If the design is not acceptable, it is readjusted and the process is repeated.

Here our suggested alternate path is to take a parametric approach to thermal modelling and use optimisation routines, which are incorporated into the earliest stages of the design process. Such an approach allows engineers to work around the uncertainty that exists in the early design stages of a mission, when components have yet to be selected, or the precise orbit is not yet known. As an example, Cullimore [52] describes the use of Thermal Desktop software to automatically determine the worst case orbit conditions through optimisation routines. In this design approach, recent improvements in computational power are not used to build more complicated models, but instead are exploited to extract more information from a model.

Indeed, there have been a number of works in the past which focus on the thermal analysis and design of small satellites, with emphasis on the methodology and modelling techniques used. The Masters Thesis by Boushon [53] outlines in detail the thermal analysis of the “MR SAT” and “MRS SAT” satellites, which demonstrate close proximity operations and formation flying using an R-134a-based cold gas propulsion system and a stereoscopic imager. Boushon’s work looks at the design process using proven methods, with a focus on technologies for small satellites. It thus aims to provide a resource for the basic principles that are used in thermal design. Similarly, the thesis by Avila de Luis [54] outlines an innovative, standardized approach for a “PocketQube” spacecraft. It looks to reduce design efforts in thermal analysis by taking into account that many components and subsystems of these spacecraft are standard and commercially available. Here, a generalised approach to thermal modelling was taken, which

can be applied to many specific systems which follow the PocketCube standard. Stout's thesis [55] improves the thermal design process by optimizing the thermal paths of a system. Here, a thermal path refers to the path that the heat follows from the point where it enters the system, to the point where it leaves the system. The thesis introduces a collective thermal design optimisation process - an improvement to point designs and designs achieved via component-level optimisation, while specifically looking at the thermal control system for the REgolith X-ray Imaging Spectrometer (REXIS) payload. Similar works by Lai [56] and Escobar [57], to name a few, are other examples of contributions to the spacecraft thermal control process.

This body of research has a focus on overall spacecraft design, and in particular, has not been employed for missions with high precision optical payloads and where high thermo-mechanical stability is required. Furthermore, they do not consider the effects of thermal stability on dimensional stability. In the case of TOLIMAN and any small satellite telescope mission, the thermal performance is tightly coupled to the optical performance of the system. With this in mind, novel approaches to thermal analysis are required.

2.5 Approach Taken in this Thesis

Drawing from the ideas in Section 2.4, this work aims to present a sound methodology for the thermal analysis and design of spacecraft with optical analysis. Parametric thermal models were created with rapid design changes in mind, meaning the effect of various uncertain parameters such as material properties and geometry could be studied through sensitivity analysis. Furthermore, optimisation routines were used to eliminate guesswork in various design choices. These strategies are general and can easily be extended to the thermal control of many different systems.

A logical flow map of the simulation processes is shown in Figure 2.15. Telescope thermal models were created based on existing CAD models of the telescope and various other components. Thermal models for the spacecraft were created directly within Thermal Desktop using primitive geometries. ANSYS Spaceclaim was used to prepare the geometry for both structural and thermal models. These geometrical preparations involve repairing the CAD model and removing unnecessary features such as small fillets and holes for the creation of finite element models. Spaceclaim geometries are imported directly into Thermal Desktop via the TD Direct package, while ANSYS Workbench creates an interface to ANSYS Mechanical.

Temperature distributions calculated in Thermal Desktop were interpolated from the thermal mesh to the structural mesh using Thermal Desktop's mesh importer capabilities. Programs written in MATLAB are used to study orbits and in various aspect of controller design.

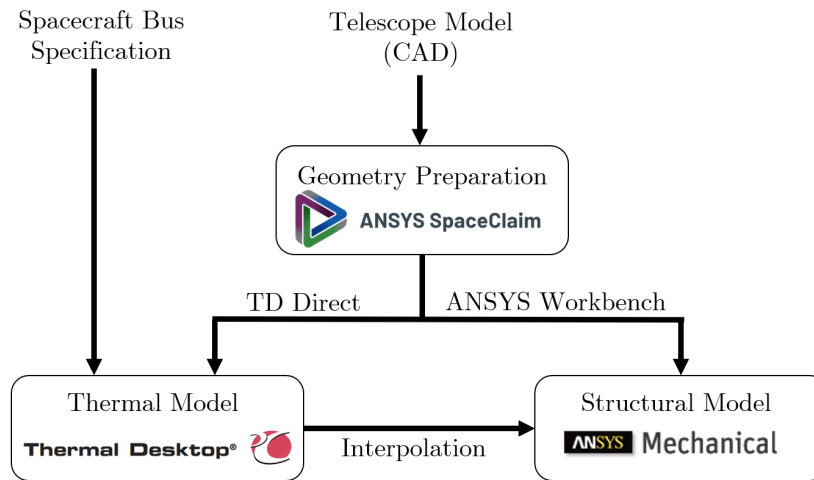


FIGURE 2.15. A map of the processes and software packages used for thermal and structural analysis, and their integration.

2.6 Summary

This chapter gives a brief outline of many of the main concepts and technologies that have been developed in the design of space telescopes with high thermal stability in recent years. As showcased in the various examples of past missions, designing a telescope with high thermal stability has many layers. Many parts of the mission can directly or indirectly inform the design required for stability of an optical payload. The relationship between scientific requirements, cost, and mission design all influence the thermo-mechanical blueprint that is adopted. Specifically, the particular choice of orbit has a large influence on the thermal control techniques used. ASTERIA, SPARCS, and JASMINE all shared the common requirement of extremely high mechanical stability, but each solution to the problem was vastly different due to divergent scientific requirements, orbit geometry, cost, and of course design inspiration. In this regard, TOLIMAN will be no different. TOLIMAN, however, has its own set of requirements governed by the unique diffractive optic, orbit geometry, and mission target. A unique approach to the thermal design process is therefore taken, making use of parametric thermal models to inform design choices. In this way, the thermal design of the payload can begin early in the mission design process, leading to a more optimal and efficient design.

Thermal Modelling

3.1 Introduction

Although the mission stability requirements apply only to the telescope, a full spacecraft model was created to fully capture the heat transfer and the thermal environment that will occur during orbit. This thermal model was divided into three sub-models: the spacecraft, telescope, and the detector. The spacecraft submodel is highly simplified, since detailed temperature distributions are not required. Furthermore, many detailed specifications for the spacecraft in particular were not yet available, so simpler models with reasonable assumptions were used to allow for rapid design changes. In terms of simulating the thermal performance of the telescope, however, these approximations have little impact.

3.2 Thermophysical Properties

In the thermal analysis of solids, the thermal conductivity k , the specific heat capacity c_p , and the mass density ρ are the three critical quantities. Often times, the product of the mass of an object m , with the specific heat capacity, mc_p , is called the “thermal mass” or “capacitance”. A useful picture is that for a given thermal environment, the thermal conductivity determines how the heat propagates through a solid, while the thermal mass determines the magnitude of temperature change.

Many times in thermal models, multiple components are combined into a single “lump” or capacitance – this is useful where precise temperature distributions are not required, and where the geometry of the component is complex, greatly reducing computational cost. Furthermore, the uncertainty in precise material properties, geometry, and thermal characteristics generally means that more complicated models do not necessarily provide more accurate models. For

lumped components, a mass average specific heat capacity is used, defined by

$$c_{p,avg} = \frac{\sum_i m_i c_{p,i}}{\sum_i m_i}, \quad (3.1)$$

where i is the index representing the various materials comprising any lumped component, m_i is the mass of each component, and $c_{p,i}$ is the specific heat capacity. Often it is not possible to detail all component materials and masses required by Equation 3.1, so engineering judgement and experience is used to make appropriate assumptions. The overall thermal mass of the entire spacecraft is important, since this dictates how much energy is left over for the telescope. All bulk material properties used in thermal models can be found in Table B.1.

3.3 Surface Optical Properties

Since the heating of a spacecraft occurs via radiation from the Sun and Earth, the coupling of the spacecraft to its environment is mediated by way of its surface optical properties. As described in Section 2.2, the emissivity ε and absorptivity α depend on the wavelength of radiation, temperature, and the angle of radiation being emitted/absorbed. For most applications in space, it is sufficient to only consider two non-overlapping spectral regions on the black body curve: solar and infrared radiation. The direction and temperature dependence of the radiation is also neglected. This is called the “two-spectral-band” model for grey surfaces [58].

In this model, two properties are defined. First, the solar absorptivity α_S , corresponding to radiation from the Sun, which is treated as a black body at 5800 K. About 40% of the radiant energy is in the visible, 10% is in the Ultra-Violet (UV), and 50% is in the near Infrared (near-IR). This property is defined for the interaction of the surface due to both direct solar heating and Earth albedo. Second is the IR emissivity, ε_{IR} , corresponding to the radiation from a black body at 300 K. This black body has about 90% of its energy in the mid-IR region. This applies to materials which emit, absorb, transmit, or reflect in the temperature range around 300 K— that is all temperatures which the thermal analysis of the spacecraft requires. Since it is assumed that the radiation is due to a black body, Kirchoff’s law applies, and so $\alpha_{IR} = \varepsilon_{IR}$, and the absorptivity does not need to be explicitly given [58]. The solar emissivity is not a relevant quantity since objects on spacecraft do not reach the effective temperature of 5800 K.

This two-spectral-band model has been widely adopted in many thermal analysis programs for years, including Thermal Desktop. In consequence there is a high availability of measured

optical properties for many of the surfaces found on spacecraft and spacecraft thermal control technologies [58, 33, 38, 43]. A list of optical properties used here can be found in Table B.2.

3.4 Modelling Thermal Contact Resistance

Spacecraft and their payloads contain many contacts between components due to the parts being bolted together, held together with adhesive, or simply making bare surface contact. As explained in section 2.2, there is a thermal resistance associated with these contacts, and hence a change in temperature across their interface. No universal models exist that can accurately predict the contact resistance of any two surfaces. A range of empirical, semi-empirical, and analytical models exist for predicting thermal contact resistance. What follows is a summary of the models used in this work and the assumptions made when choosing them.

3.4.1 Uniform Contact Pressure

Contact resistance arises due to the microscopic roughness of surfaces. Due to the reduced contact surface area, the mechanical stress on the contact regions can be large, and so microscopic deformation can occur on the surface. Mikic et al. [59] and Greenwood and Williamson [60] suggest that even at moderate contact pressures, very few points contact with light pressure, so this deformation is largely plastic. Determining whether to use an elastic [61] or plastic model is not clear, however, the review by Sridhar and Yovanovich [62], which used more than 450 thermal contact conductance data points for various engineering metals, showed that for a wide range of surface roughnesses (σ/m) ranging from 8 to 60 μm , the plastic model was a much better fit. The telescope structure, for example, which is constructed from SiC has a significantly lower yield stress when compared to the metals tested in the review by Sridhar and Yovanovich, with similar surface finishes, so it is assumed that for the moderate contact pressures in the construction of the telescope, a plastic model is appropriate.

By treating the location of surface asperities as randomly distributed and their heights as Gaussian, Yovanovich [63, 32] proposed the following dimensionless correlation for plastically deforming surfaces,

$$\frac{h_c \sigma}{k_s m} = 1.25 \left(\frac{P}{H_c} \right)^{0.95}, \quad (3.2)$$

where h_c is the conductance coefficient for the contact, $\sigma^2 = \sigma_1^2 + \sigma_2^2$, where σ_1 and σ_2 are the RMS deviation of the surface roughness for each surface, k_s is the harmonic mean thermal conductivity of the two surfaces in contact, $m^2 = m_1^2 + m_2^2$, where m_1 and m_2 are the effective absolute surface slopes of the microscopic asperities on the surfaces, P is the contact pressure, and H_c is the surface micro-hardness. The values σ and m describe geometrically the surface roughness. The surface micro-hardness H_c in general is not constant with the depth of the surface. Regions close to the surface are harder than the interior due to work hardening. Song and Yovanovich [64] provide the following correlation

$$\frac{P}{H_c} = \left[\frac{P}{c_1 \left(\frac{1.62\sigma \times 10^6}{m} \right)^{c_2}} \right]^{\frac{1}{1+0.071c_2}}, \quad (3.3)$$

which is valid for $10^{-6} < P/H_c < 2 \times 10^{-2}$. Here the constants c_1 and c_2 are correlation coefficients that are obtained through a method of Vickers Hardness testing proposed by Hegazy [65]. These coefficients need to be measured, and complete lists for various materials are not publicly available, and this also goes for the materials being used on the TOLIMAN telescope. Hegazy lists a few for typical engineering materials such as Stainless Steel 304, Nickel 200, Zirconium, and Zirconium-Niobium alloys. In this case, when using Equations 3.2 and 3.3, best estimates based on available properties for similar materials must be made for the Vickers correlation coefficients, as well as for the roughness properties σ and m .

In this stage of the telescope design, the precise surface finishes and material properties are unknown, so the best estimates are made for such properties in discussion with other engineers working on the mission and based on the available data of similar materials. The purpose of these models in this case is to give estimates that inform design.

3.4.2 Bolted Contacts

Plates bolted together will deform elastically at regions near the bolt. This can be thought of as a contact conductance problem with nonuniform pressure. Although various theoretical treatments are outlined by Gilmore [33], these approaches are time consuming and require detailed design knowledge that is not yet available. Instead, the following simple model was adopted,

$$\frac{h_c t_t}{k_s} = 54.7 \left[\frac{A_N}{t_t^3 \tau^{0.5}} \right]^{-0.764}, \quad (3.4)$$

where t_t is the thinnest of the two plates in contact, A_N is the area per bolt (that is total area divided by number of bolts), and τ is the bolt torque. Data for this correlation came from General Electric Inc. reports and from six TRW Inc. reports (the latter by H. A. Pudewa). The correlation included data for plate thicknesses between 1.59 to 25.4 mm, bolt torques between 0.037 to 9.48 Nm, and bolt per area 0.272 to 19.4 cm², which is covering the type of bolted connections used in this thermal analysis. The correlation was found to have an R^2 value of 0.93 with the tested data [33].

3.4.3 Adhesive Contacts

Adhesives are used to bond mirrors of the telescope to the metering structure. An adhesive will act as a thermal interface filler between two surfaces in contact, thus decreasing the thermal contact resistance (see Figure 2.4). The resulting contact conductance will depend on the initial surface roughness properties, the conductivity of the adhesive, the bondline thickness, and the fluid properties of the adhesive, which dictates how well the interface is filled [34, 66, 67]. Many of these properties are not known or unavailable, so as before, the simple dimensional correlation developed by Mirmira et al. [34] was used, given by

$$\frac{h_c \sigma}{k_a} \frac{t_{BL}}{TIR} = 0.15 \left(\frac{h_b \sigma}{k_s} \right)^{0.20}, \quad (3.5)$$

where t_{BL} is the bondline thickness, k_a is the adhesive thickness, TIR is the true indicator reading of the surface profile, and h_b is the bare surface contact conduction coefficient. The correlation was created using data from 12 different epoxy adhesives, with contact pressures ranging from 345 kPa to 2756 kPa, and bondline thicknesses 0.074 mm to 1.549 mm, and gives an average RMS difference of 53% from measured data.

3.5 Orbital Heating Environments

While in orbit, the satellite will experience heating from three sources: direct solar radiation, reflected solar radiation from the Earth (albedo), and IR radiation emitted from the Earth's surface. In this work, the thermal environments were determined in accordance with the "Simple Thermal Environment Model" (STEM) [68] and the "Guidelines for the Selection of Near-Earth Thermal Environment Parameters for Spacecraft Design" [69], both works from NASA. These use data from the Earth Radiation Budget Experiment (ERBE), and various other

sources to develop guidelines for engineers in selecting thermal environment parameters for hot, cold, and median case analysis. This approach means variations in these heating environments can be accounted for.

Radiation from the sun can be treated as parallel rays of constant intensity. Variations in solar flux occur largely due to variations of the Earth-Sun distance throughout the year. The solar constant is defined as $S_0 = 1367 \text{ W/m}^2$, and is taken to be the “median” case solar flux. The solar fluxes for hot and cold cases are given to be $S_{\text{hot}} = 1414 \text{ W/m}^2$, and $S_{\text{cold}} = 1322 \text{ W/m}^2$.

A sizeable fraction of solar radiation incident on the Earth is reflected off into space. Governing this is the albedo, which is highly variable across the globe and depends on the reflective properties of the atmosphere, cloud cover and Earth’s surface. For example it is larger over continents than oceans, and still higher for regions of snow cover. As a result, the albedo is on average larger at higher latitudes. The report by Peyrou-Lauga [70] uses NASA CERES data obtained every day from 2007 to 2011, which has a resolution of $1^\circ \times 1^\circ$ grid over the entire Earth to study this radiation. A typical Earth mapping of albedo and Earth IR flux from top of the atmosphere is shown in Figure 3.1 (left). The definition of albedo is given according to the following equation,

$$a = \frac{E_r}{S \cos \theta_{SZ}}, \quad (3.6)$$

where E_r is the reflected heat flux, S is the direct normal solar heat flux, and θ_{SZ} is the solar zenith angle, illustrated in Figure F.4. Furthermore, E_r itself can depend on the solar zenith angle, making the relationship more complicated.

In addition to reflected solar flux, radiation is also emitted from the Earth directly. This emerges in the infrared and is termed Outgoing Longwave Radiation (OLR) originating from cloud tops, atmospheric gases and the Earth’s surface itself. Because the energy flow is comprised of many steps of absorption and re-emission at different bands (as well as some energy transport by convection) the emergent spectral distribution of OLR is complex. Here it is sufficient to treat the Earth as a greybody, emitting radiation at an effective temperature, which varies around 300 K. Figure 3.1 (right) shows a typical Earth IR emission flux mapping.

Earth albedo and emitted IR flux are not independent; rather they are correlated. Low Earth IR values tend to result in higher albedo, while high Earth IR values tend to be paired with low to moderate albedoes. This means that the orbital hot case is not the combination of the

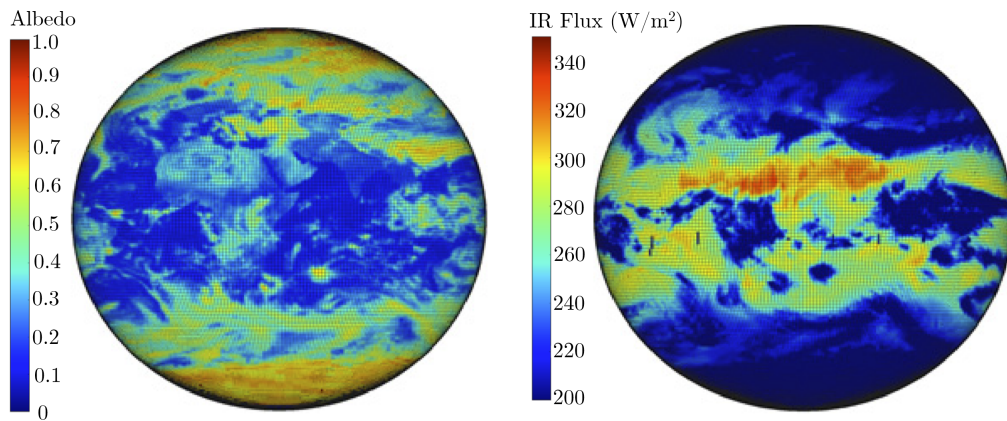


FIGURE 3.1. Typical mapping of Earth albedo (left) and Earth IR flux (right) from the top of the atmosphere, on the Spring Equinox (21/03/2011) obtained from Peyrou-Lauga [70]

highest possible albedo and the highest possible Earth IR. The report by B.J. Anderson et al. [69] provides engineering extreme cases for Earth albedo and Earth IR at varying inclination angles and averaging times, which are the time intervals over which the heat flux was averaged when being measured. Table 3.1 shows the different values for albedo and Earth IR (OLR) for various design cases given by Anderson. An averaging interval of 128 seconds was taken, based on the thermal time constants of the telescope system. For most analyses, the mean case was used, however, hot and cold cases (which correspond to 0.04th and 99.96th percentile worst cases) are used for extreme case simulations.

TABLE 3.1. Combined hot, cold, and mean cases for thermal radiation coming from the Earth for an 128 second averaging time [69].

	Cold Case		Hot Case		Mean Case	
Inclination	Albedo	OLR (W/m ²)	Albedo	OLR (W/m ²)	Albedo	OLR (W/m ²)
Low	0.13	226	0.29	295	0.18	246
Medium	0.15	213	0.30	265	0.22	234
High	0.16	212	0.31	262	0.23	211

3.6 Detailed Spacecraft and Telescope Model

Within Thermal Desktop, each submodel was created in a separate drawing file and combined into a “parent” model using AutoCAD XREFs. This means the parameters defined in the original models could be modified in the parent model when performing parametric analysis and optimization routines. Detailed calculations and numeric tables for all thermal models

discussed below are given in in Appendices B, C, and D. Due to the unavailability of detailed information of various components, estimates were made, and the models will have some level of uncertainty. In these cases, correlation testing should be performed, where the experimental results are correlated with the thermal model to determine the unknown thermal parameters. Correlation testing could not be covered in this work, however its place in developing thermal models should be understood.

3.6.1 Spacecraft Model

The TOLIMAN spacecraft bus is based on the GMS-T (InnoSat) satellite platform by OHB Sweden. Only basic dimensions are currently defined, and preliminary components are selected. Therefore, in the creation of this model, appropriate assumptions and estimates were made. Furthermore, a model was created that was parametric in nature, meaning rapid changes in the spacecraft design that are typical of the early design phase could be accounted for in this model. As the spacecraft model is only used to represent the heat transfer to the payload, and not study the temperature of the entire spacecraft, the model created is simple.

The satellite thermal model in Thermal Desktop can be seen in Figure 3.2. Finite difference surfaces were used to represent the relatively simple geometry of the spacecraft to dramatically reduce the computational cost of the model. These savings were better allocated to the telescope's model and analysis, where more detail is required. Since many details of the spacecraft are unknown at this point, the introduction of finer, more complicated models will have little to no added benefit to the accuracy of the solution.

Due to the large, thin planar walls of the spacecraft bus structure, no through-surface conduction was calculated, however, the thickness was taken into account for in-plane conduction. Furthermore, features in the components which are generally intended for mechanical strength such as holes, fillets, and ribbing, have a negligible effect on thermal heat transfer and were excluded, provided the overall thermal capacitance of the component is preserved. The top and bottom plates of the spacecraft structure were modelled using triangular elements so that the overall shape and telescope hole could be represented geometrically. This was created using the Thermal Desktop TD Mesher utility. Panels were thermally connected to each other by sharing edge nodes, essentially making them continuous.

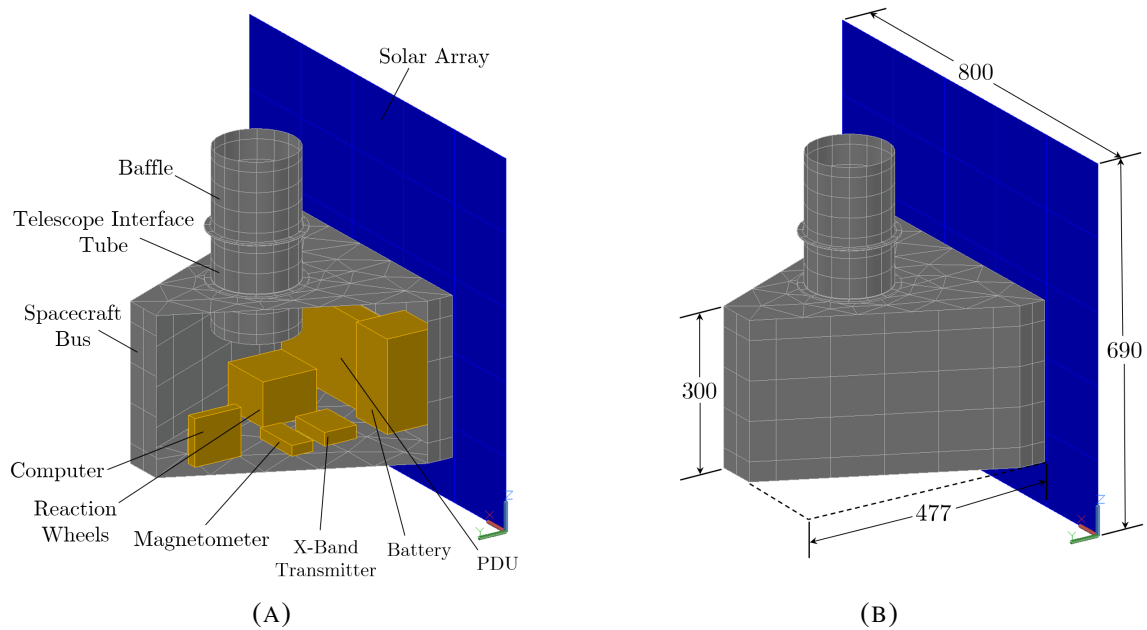


FIGURE 3.2. Internal view of spacecraft thermal model (A) and dimensions (B). Dimensions are in mm.

A thermal conduction model for the solar arrays was constructed to account for the typical layered structures used in spacecraft. Currently, detailed designs for this structure are not available, however, it is typical for a spacecraft to have solar cells held with adhesive on the face sheet of a honeycomb insulating material. The face sheet of this honeycomb on either side can be made from aluminium or CFRP [33]. Other more complicated layering structures are common, however, a simple approach was taken here to obtain a representative model. The face sheets are assumed to be aluminium. Details and calculations for the thermal properties of the solar array are given in Appendix C1.4.

Electronic components were modelled as single-node finite difference solids – nodes with average heat capacities given by Equation 3.1. Precise thermal knowledge of these components is not available, and the calculation of precise temperature distributions is not useful or computationally efficient. Constant heat loads were applied to the nodes to model the heat generation of the electronics. Each component was assigned a “power fraction” – a fraction of the total spacecraft heat generation which the component consumes. This way, the heat generation from the electronics could be controlled by a single parameter. Furthermore, only components in the spacecraft bus which contribute to at least 2% of the total heat load were included in the thermal model. The thickness of the panels was adjusted to account for the loss of thermal mass due to components not included in the model, resulting in a total mass of 35 kg for the

entire spacecraft – in line with mass estimates by the spacecraft provider. Detailed information on heat generation, thermal properties, and their calculations are given in Appendix C1.1 and C1.2. Information on the contact conductance values used for the spacecraft model can be found in Appendix C1.3.

In Thermal Desktop radiative heating due to the Sun and Earth, as well as radiation heat transfer between components and into deep space was calculated using the Monte Carlo ray tracing method. Two radiation analysis groups were used: one for the internal surfaces of the spacecraft, where all electronic components are housed, and one for the external surfaces of the spacecraft. The use of two radiation analysis groups dramatically reduces the computational cost, as unnecessary calculations are avoided between surfaces that will never be in view of each other. Orbital heat loads from the Sun and Earth on each surface were calculated at 16 equally spaced locations around the orbit, with extra locations at the eclipse transitions. Heat rates were interpolated linearly for regions in between.

3.6.2 Detector Model

The detector used to deliver the primary science data on TOLIMAN comprises a major heat disturbance for the telescope due to its mount point behind the primary mirror. In common with the spacecraft model, precise temperature distributions are not the purpose of this model; instead the thermal mass and heat transfer to the telescope were to be modelled. The detector used on TOLIMAN will be a FLIR BFS-U3-200S6M-BD2 [71].

The Thermal Desktop model is shown in Figure 3.3. The Printed Circuit Board (PCB) and Integrated Circuits (ICs) were modelled with finite difference surfaces. Only the main heat generating components such as the FPGA, SDRAM, CMOS Sensor, and power management were included in the thermal model [71, 72, 73, 74]. These ICs were modelled as lumped nodes with average properties given by a 40/30/30 mix of Silicon/Epoxy/Copper, which is detailed in Appendix C2.1. As with the spacecraft, each component is assigned a power fraction that determines the fraction of the total power that is generated by each component. This allowed the detector heat generation to be controlled by a single parameter. These power fraction assignments are given in Table C.4.

The thermal conductivity of the PCB varies both in the in-plane direction and the through-thickness direction; furthermore it depends on the composition of the PCB layers and the copper

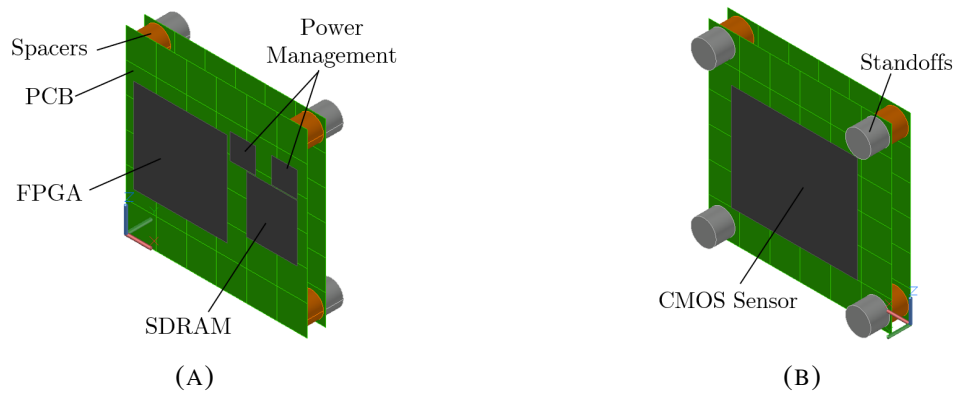


FIGURE 3.3. Top (A) and bottom (B) side of the FLIR BFS-U3-200S6M-BD2 detector thermal model.

trace layout. Thermal conductivity was taken to be isotropic, and through thickness conduction is neglected. Estimates were made based on the works of Nirale [75] and Costa and Vlassov [76], with a thermal conductivity of 8 W/m/K for the PCB being used. The heat capacity was evaluated using Equation 3.1 with a 60/40 mix of Epoxy/Copper. Since passive components such as resistors, capacitors, and smaller semi-conductor devices were not included explicitly in the thermal model, the total mass of the PCB was adjusted such that the mass of the full detector model matches the true mass of the detector (10 g). In this way the thermal mass of these components was accounted for.

Standoffs and spacers were modelled as single node cylinders, made from steel. The mounting screw used for these spacers was lumped into the spacers and standoffs themselves, as they are of the same material. Contact conductances between components and the PCB were estimated based on junction-board conductances given in IC data sheets, and from known contact conductances between metals, which is described in detail in Appendix C2.3.

3.6.3 Telescope Model

Unlike the spacecraft and detector, the telescope thermal model was made from finite elements. This was done using Thermal Desktop's TD Direct feature which can create thermal models from pre-existing CAD models. Finite elements are useful for creating more complex geometries, such as the precise curvature of the mirrors. A more accurate representation of the geometry is also useful when mapping temperatures onto a structural model in Chapter 6.

The finite element model of the telescope can be seen in Figure 3.4, it consists of four separate components: the primary and secondary mirrors, the tube, and the detector mount. Fine details in the structure such as small fillets and holes were removed to allow for a better quality mesh, with negligible affect on accuracy of the model. Ideally, the tube, and most of the structure would be meshed used a hexahedral mapped mesh, however, this was not possible due to software limitations. Swept triangular face meshes were used on the tube, spiders arms, and detector mounts. The size of the mesh was determined through a mesh convergence study, which is described in detail in Appendix D. In general, thermal models can use much more coarse meshes when compared to structural models, which will be evident when the structural model is introduced in Chapter 6.

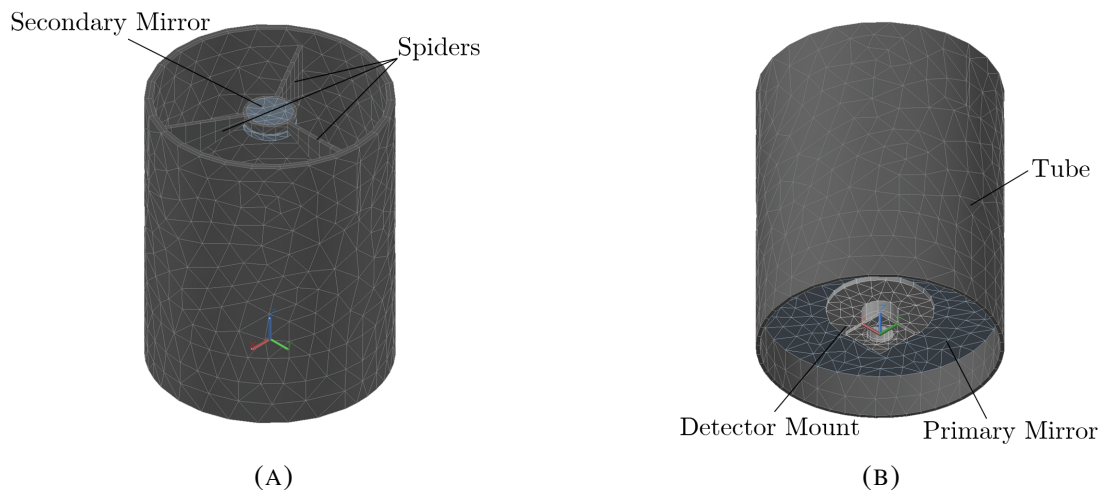


FIGURE 3.4. Thermal model of telescope assembly.

Mirrors are bonded to the structure, and thermal conductance coefficients are calculated as described in Section 3.4.3, with numerical calculations given in Appendix C3.2. The detector mount, made from Invar, is bolted to the rear of the primary mirror, and the contact conductance is calculated using Equation 3.4, which is explained in depth in Appendix C3.3. Material and optical properties for the telescope structure and mirrors can be found in Appendix B.

Passive Thermal Control and Cold Bias

4.1 Introduction

The thermal control approach for the TOLIMAN telescope is a cold biased design, which means that passively, the telescope must be able to maintain temperatures below the operating set point of 20°C. A passive control approach was proposed to achieve a cold bias using common thermal control technologies. The first step in analysing on-orbit performance was to determine the most thermally challenging orbit conditions that the telescope may experience. These worst cases are a function of the possible orbits that may occur during the mission lifetime. Parametric analysis was used to assess the effect of uncertain or unknown design parameters on passive telescope temperatures.

4.2 Passive Control Design

Figure 4.1 shows the proposed control approach for the TOLIMAN mission. The aims of the passive thermal control are to prevent heat escaping the system, as well as reduce external heating disturbances from the rest of the spacecraft, Sun, and Earth. The telescope interface enclosure was wrapped completely with MLI to minimise such heat transfers. The MLI was specified to have an effective emissivity of $\varepsilon^* = 0.02$, which represents the lower end of achievable performance with modern MLI [33]. Lower MLI effective emissivity should be used if possible. The optical properties of the exterior layer of MLI is crucial in dictating net heat flow to the system. Since a cold biased design was adopted, materials which act as solar reflectors (Chapter 2.3.2) were selected to minimize heat absorbed by direct solar radiation and Earth Albedo. These materials have a high IR emissivity, meaning more heat is radiated out to space. By Kirchhoff's law this larger IR emissivity means that IR absorptivity will also be larger, and so more OLR from the Earth is absorbed. OLR accounts for only a small fraction of

the total radiation when compared to other sources, especially considering the Earth's effective temperature is similar to, or lower than the operating temperature of the telescope at 293.15 K. In this design, an exterior coating of Aluminized Teflon was specified, which, for a thickness of 5 mil (0.0254 mm), has $\alpha_S = 0.17$ and $\varepsilon_{IR} = 0.79$ [33]. This surface coating was only specified on the exterior regions of the telescope enclosure due to its relatively high solar reflectivity.

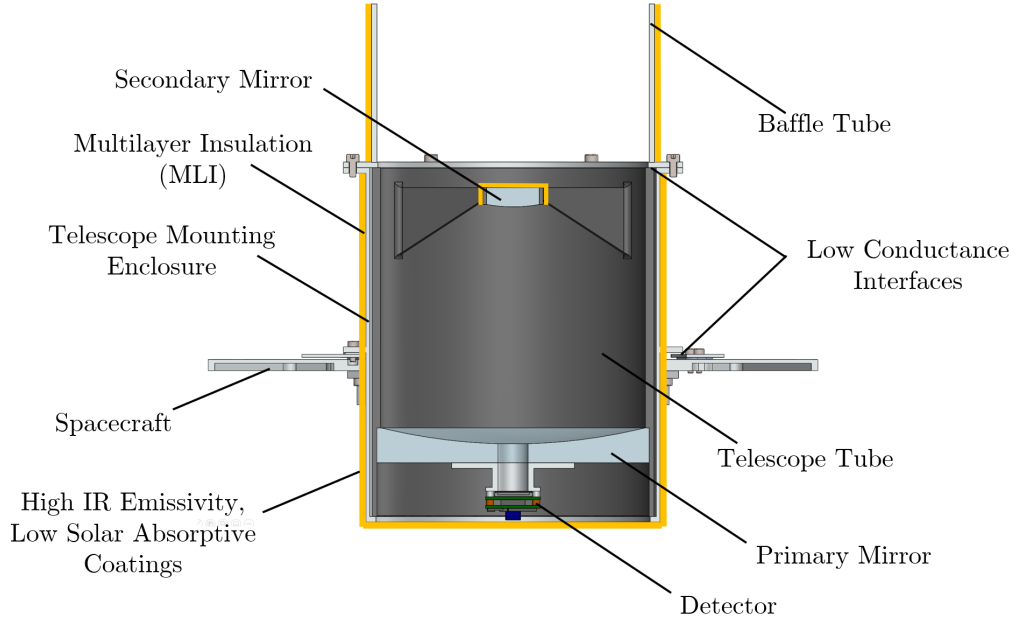


FIGURE 4.1. Passive control design and interface with spacecraft. Some details are omitted due to confidentiality agreements with other parties involved in the design and manufacturing of the telescope.

MLI of the same specification was also placed around the rear of the secondary mirror of the telescope and its mounting structure. The spider arms which hold this mirror are thin, and hence create a very low conductance path for heat to flow from the rest of the spacecraft. MLI at this region decreases heat losses and results in more uniform temperatures on the secondary mirror. Unlike the exterior MLI, a low reflectivity Black Kapton coating was used ($\alpha_S = 0.93$ and $\varepsilon_{IR} = 0.85$), which is common for regions where glint suppression is required [33]. It should be noted that further glint analysis is required to verify this coating is suitable.

A 15 cm baffle tube prevents the direct illumination from heat sources such as the Sun and Earth. Its length is driven by both stray light requirements, as well as thermal stability, which is studied later in Section 4.5. The interior of the baffle was coated in a high absorptivity coating, Chemglaze Z306 ($\alpha_S = 0.96$ and $\varepsilon_{IR} = 0.91$), for stray light prevention [33]. All thermal conductances from the telescope to the enclosure, and from the enclosure to the spacecraft must

be as small as possible to minimise heat transfers in and out of the telescope. This is explored further in Section 4.6.

4.3 Study of Potential Orbits

During orbit, the relative positions of the Sun, Earth, and Alpha Centauri will dictate their thermal loads on the spacecraft and the telescope. For example, for a given orbit, the telescope may be facing away from the Earth while in eclipse during certain times of the year, meaning much lower temperatures will be reached due to reduced heating, while the opposite may be true during other times of the year. The aim here is to first understand the conditions in which the telescope can perform its primary science mission – taking images of the target – to narrow down potential orbital environments.

The positions of the Sun and Earth were studied by considering their angles relative to the optical axis of the telescope, shown schematically in Figure 4.2. These angles give a simple indicator of the position of the two main heat sources. Additionally, the roll of the spacecraft about the optical axis is required to fully define orientation, which is defined by the fact that the solar array will always be facing the sun. There is a minimum allowable angle for $\theta_{\odot}$ and $\theta_{\oplus}$ before the optical system will not be able to obtain useful images due to obstruction of the science target from the Sun and Earth, as well as stray light requirements. These requirements further limit the possible available orbits that will be used for thermal analysis.

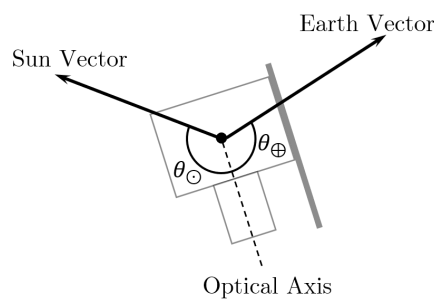


FIGURE 4.2. Sun angle $\theta_{\odot}$ and Earth angle $\theta_{\oplus}$ measured between the respective vectors and the optical axis.

TOLIMAN pointing stability requirements mean that the telescope is assumed to always face the science target. For example maneuvers during the orbit to point the telescope away from heat sources such as the Earth were not considered. This restriction potentially makes cold biasing more difficult: the spacecraft is not permitted to alter its attitude to mitigate varying

thermal loads. An orbit propagator was created in MATLAB to study the variation of these angles throughout the mission lifetime. The orbit propagator simulates Keplerian orbits, with corrections for orbital perturbations given by the J_2 zonal harmonic for the Earth's equatorial bulge if required. In this simulation, the Sun, Earth, and optical axis vectors were tracked throughout the orbit. The detailed implementation is presented in Appendix F. At the current stage of the mission design, there are two candidate orbits, an SSO orbit (which will have an inclination of 98°), and a 7° inclination orbit, both with altitudes of 550 km. Which orbit is used depends on various mission requirements, one of them being the thermal performance.

The precise minimum angles for $\theta_\odot$ and $\theta_\oplus$ are not yet available. For the purpose of this study, these minimal angles were taken to be preliminary estimates of $\theta_\odot = 90^\circ$ and $\theta_\oplus = 110^\circ$. Angles smaller than 90° mean that the telescope is looking “into” the object. Whilst angles less than 90° may be acceptable due to adequate baffling, this estimate is reasonable. A larger angle was used for the Earth since its small distance from the spacecraft results in it filling a large field of view. Given these minimal angles, the time which the telescope can view the target was calculated during an orbit.

For a circular orbit, the Solar Right Ascension (SRA) dictates the position of the sun, the Right Ascension of the Ascending Node (RAAN) of the orbit defines the orientation of the orbit around the Earth, and the inclination describes the “tilt” of the orbit (See Figures F.1 and F.2). The SRA will vary throughout the year as the Earth orbits the Sun, while the RAAN will depend on the initial placement of the orbit, but will also vary throughout the year due to orbital perturbations. In principle, given an initial orbit condition, this can be predicted, and the resulting cases can be studied. This is not very useful however, since the initial RAAN of the orbit is unknown, and predicting its precession will yield very specific results that may not cover possible orbits that the spacecraft will encounter in actuality. Instead, all possibilities were considered. Figure 4.3 shows a heat map representing the orbital viewing times of Alpha Centauri based on the minimum angles $\theta_\odot = 90^\circ$ and $\theta_\oplus = 110^\circ$ as a function of the SRA and RAAN both in the Sun, and during eclipse for a 7° inclination.

Common to all RAANs is the period between SRAs of approximately 190° to 250° , where no imaging can take place. The right ascension of Alpha Centauri is approximately 220° , with a declination of -60° , meaning at these times of the year, the telescope must look in the direction of the sun when not in eclipse, and into the Earth during eclipse. For the most part however,

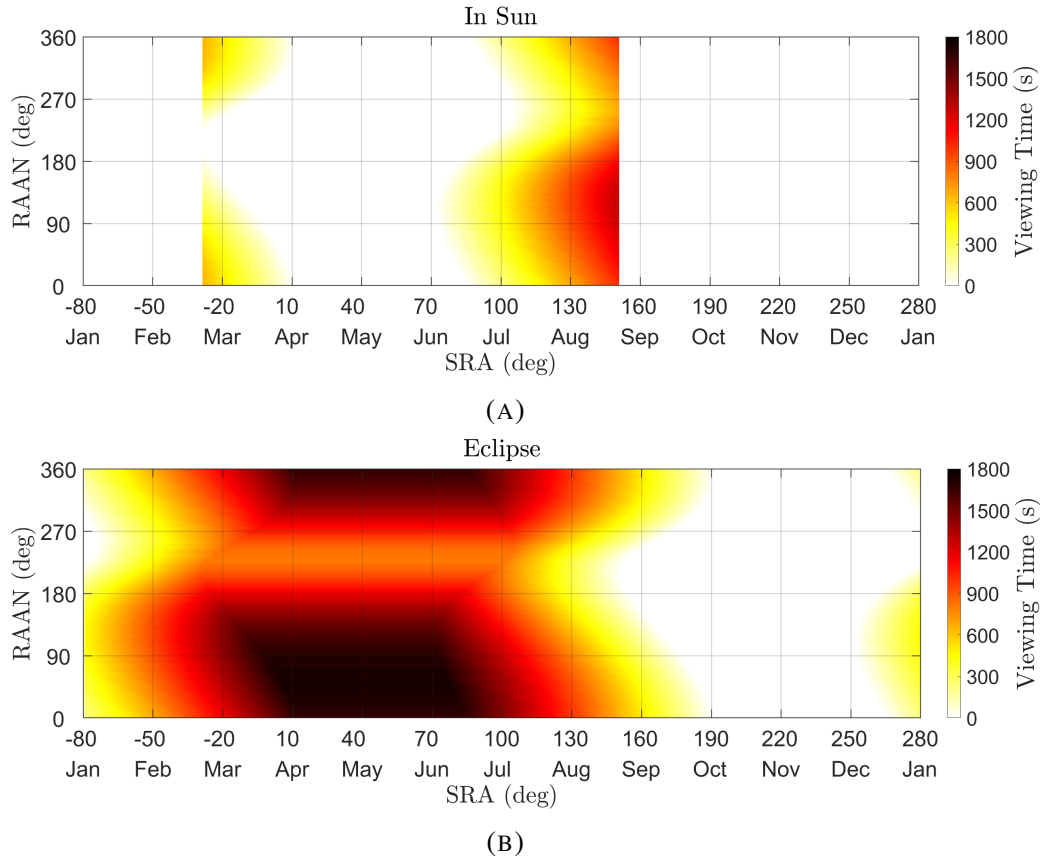


FIGURE 4.3. Alpha Centauri viewing times a given Solar Right Ascension (SRA) and orbital Right Ascension of the Ascending Node (RAAN) whilst in sun (A) and in eclipse (B) for minimal optical axis angles $\theta_{\odot} = 90^{\circ}$ and $\theta_{\oplus} = 110^{\circ}$, and a 7° inclination orbit.

the RAAN does not have a strong effect on viewing times for this low inclination orbit. The period for SRAs between -50° to 120° allows for maximal viewing times of the target since now the target and sun are in the opposite direction. The sharp cutoffs in Figure 4.3A are due to violation of the minimum angle of $\theta_{\odot}$ not being met during these times of the year, so imaging cannot take place in the sun.

Figure 4.4 shows a similar heat map for SSO. Instead of looking at the RAAN however, it is more useful to consider the difference between the RAAN and SRA, since for SSO orbits, the RAAN precesses at a period equal to one year, so this quantity is constant throughout the life of the orbit. In Figure 4.4, horizontal lines represent a single given possible orbit throughout the year. If $\text{RAS} - \text{SRA}$ is 90° or 270° , then the normal of the orbital plane is roughly parallel to the Sun vector. In this type of orbit, eclipse periods are short, but viewing times of the target are long. As before, there is a period of SRAs of 190° to 250° where imaging times are very low.

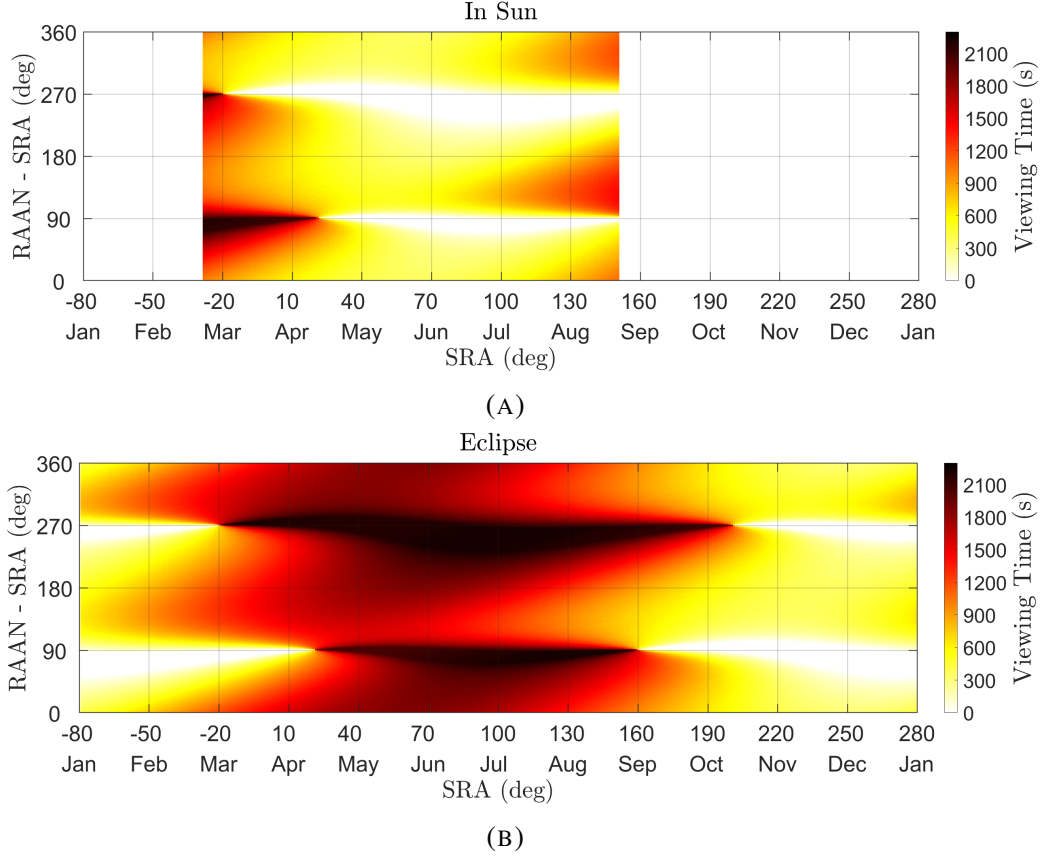


FIGURE 4.4. Alpha Centauri viewing times a given Solar Right Ascension (SRA) and orbital difference between SRA and Right Ascension of the Ascending Node (RAAN) whilst in sun (A) and in Eclipse (B) for minimal optical axis angles $\theta_{\odot} = 90^{\circ}$ and $\theta_{\oplus} = 110^{\circ}$, and an SSO orbit.

Whilst this information can be useful in mission design, for thermal control, viewing times give some understanding to the thermal environment that the telescope will encounter. In particular, the viewing times of thermal worst cases help in understanding the trade-offs between viewing time and thermal performance.

4.4 Worst Case Orbit Determination

The two candidate orbits span a wide range of individual thermal configurations so only selected cases can be simulated. Most informative are the worst cases: those yielding the hottest and coldest telescope temperatures. Note here we do not reference the heating rates in Table 3.1, but rather those orbital geometries yielding the most challenging variations of heat load.

Determining these worst cases is not trivial, and typically in thermal analysis, this is done by estimating based on the basic principles of thermal analysis, orbits, and experience. Instead, optimisation routines within Thermal Desktop were used to search a restricted space of possible orbits, based on the analysis from the previous section to determine the worst possible cases for both the 7° and SSO orbit [52, 50]. In this optimisation problem, the RAAN and SRA which gives the maximal/minimal temperature on the telescope structure over an entire orbit was found. The space of possible orbital SRAs was restricted to the region where imaging time is greater than 200 s. This somewhat arbitrary estimate can be refined when more information is available on the required viewing times – which is determined not only by integration times for imaging, but also pointing stability performance – then these figures can be revised. Placing this restriction ensures that the orbital cases being tested are relevant to the science aims, and avoids irrelevant cases, such as when the telescope is looking into the sun causing excessive heating.

Mathematically, the problem for finding the hot and cold cases can be written as

$$\begin{aligned} \text{maximise: } & \max_{\text{time}} \left\{ \max_{\text{nodes}} \{T_{\text{telescope}}\} \right\}, & (\text{Hot Case}) \\ \text{minimise: } & \min_{\text{time}} \left\{ \min_{\text{nodes}} \{T_{\text{telescope}}\} \right\}, & (\text{Cold Case}) \\ \text{subject to: } & 0^\circ \leq \text{RAAN} \leq 360^\circ, \quad \text{SRA}_{\min} \leq \text{SRA} \leq \text{SRA}_{\max}, \end{aligned}$$

where $T_{\text{telescope}}$ is the temperatures of the nodes of the telescope, as a function of time during an orbit. For the 7° inclination orbit, it was taken that $\text{SRA}_{\min} = -80^\circ$ and $\text{SRA}_{\max} = 160^\circ$, and for SSO, $\text{SRA}_{\min} = -80^\circ$ and $\text{SRA}_{\max} = 200^\circ$. This is solved using a linear sequential iterative method within the SINDA solver of Thermal Desktop. With each iteration, a transient simulation was run, using an orbital average steady state solution as the initial condition. Each iteration of the optimisation routine only used temperatures in the final orbit after four initial orbits are run to clear the memory of the initial condition. Prior to calling the solver for the optimisation routine, a Latin hypercube sampling algorithm was used to scan the design space (by issuing the *DSCANLH* routine to the SINDA solver), this allows the solver to determine a better starting point for the worst case search [77]. The objective function (max/min temperatures) to be minimized/maximized was calculated using a Fortran subroutine created within Thermal Desktop. The orbital parameters of the resulting orbits can be found in Table 4.1.

TABLE 4.1. Orbital worst cases for 7° inclination and SSO orbits and corresponding viewing times.

Orbit	Case	RAAN (deg)	SRA (deg)	Viewing Time (s)
7°	Cold Case	167	53	1035
	Hot Case	221	-25	1485
SSO	Cold Case	237	57	2205
	Hot Case	268	22	2130

Worst case orbits are shown with the spacecraft and eclipse region for the 7° inclination orbit and SSO orbit in Figures 4.5 and 4.7. Orbital temperatures, as well as total heat transfers to other components in the model can be found in Figures 4.6 and 4.8. These results used “mean” case orbital heating given in Table 4.1. Hot and cold extreme cases result in an increase/decrease of orbital temperatures of no more than 5°C on all cases. Given these results, the design proposed in Figure 4.1 is able to maintain temperatures below 20°C for cold bias.

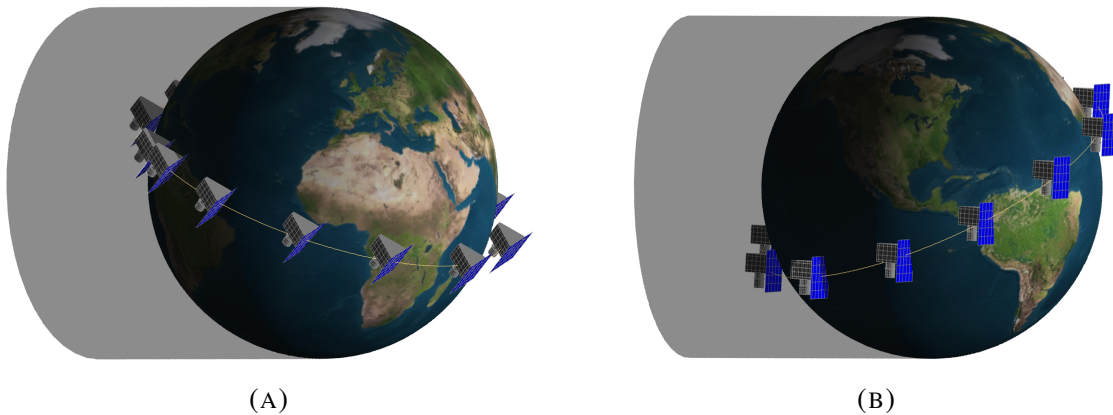


FIGURE 4.5. Cold case (A) and hot case (B) orbit for 7° inclination orbit as obtained from worst case search.

As expected, in general, temperatures for the SSO were higher, since the telescope will be pointed more directly at the Earth during parts of its orbit. Furthermore, temperature gradients (maximum minus minimum temperatures) are greater in the SSO orbit since there are large changes in radiative heating/cooling during an orbit as shown in Figures 4.8C and 4.8D. From a temperature stability point of view, a high inclination orbit is less favourable.

Comparing the worst case on-orbit temperatures with the viewing times in Table 4.1 shows that while an SSO orbit may provide larger viewing times per orbit, this is traded off against poorer thermal stability compared to a lower inclination, 7° orbit. The final orbit chosen will therefore need to take into account the required integration times for adequate imaging of the target, and whether more imaging time is required at the expense of less stable thermal performance.

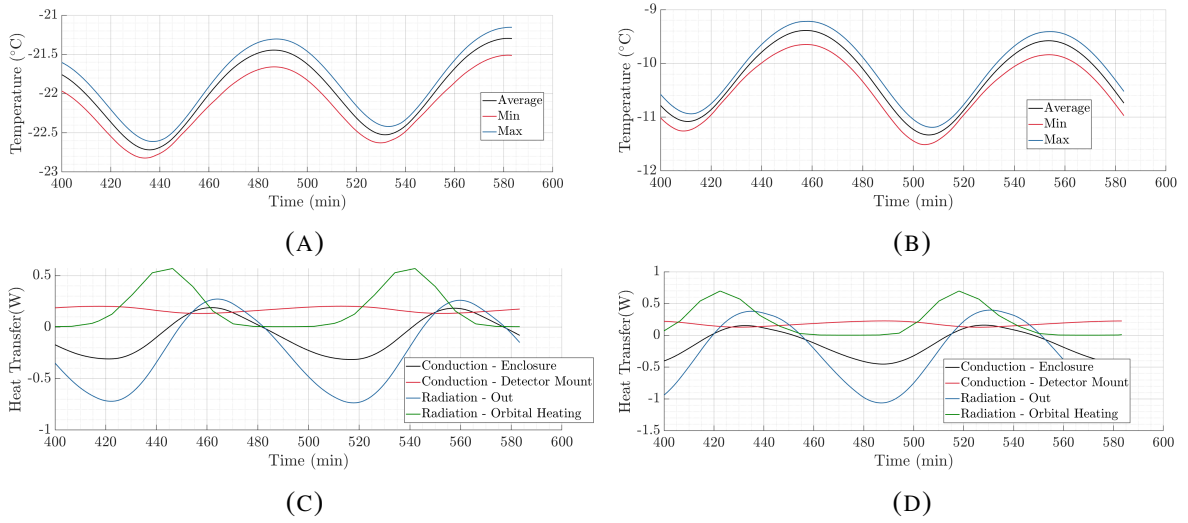


FIGURE 4.6. Temperatures after four orbits for 7° inclination orbit cold (A) and hot (B) case in Table 4.1, and total heat transfers out of telescope structure for cold (C) and hot (D) cases.

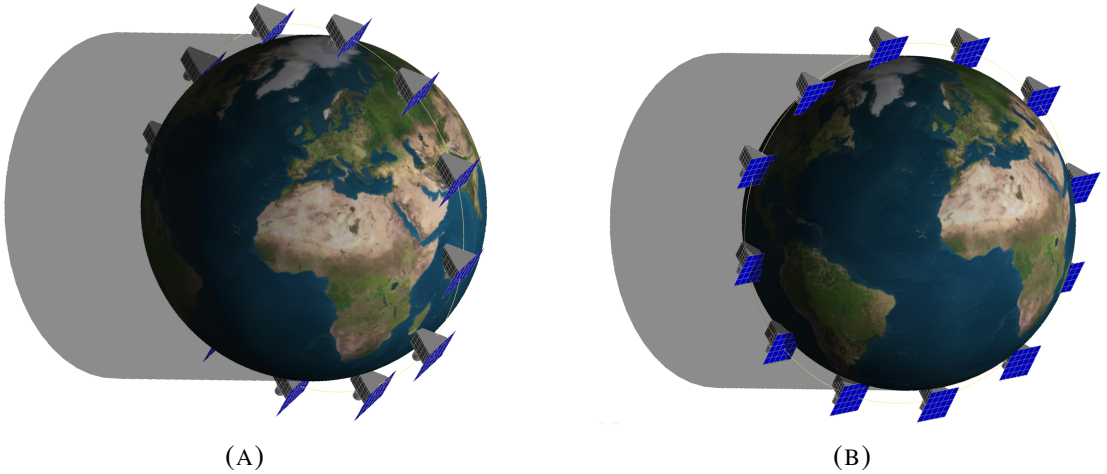


FIGURE 4.7. Cold case (A) and hot case (B) orbit for SSO orbit as obtained from worst case search.

An example temperature distribution of the spacecraft and telescope is shown in Figure 4.9. The MLI and low thermal conductance with the rest of the spacecraft mean that the telescope enclosure is decoupled from the rest of the satellite, maintaining a low, uniform temperature. The largest thermal gradients on the telescope occur in the direction of the optical axis, since the largest heat losses occur at the mounting points. The thin spider arms restrict heat flow to the secondary mirror, and hence a large thermal gradient forms between it and the telescope structure.

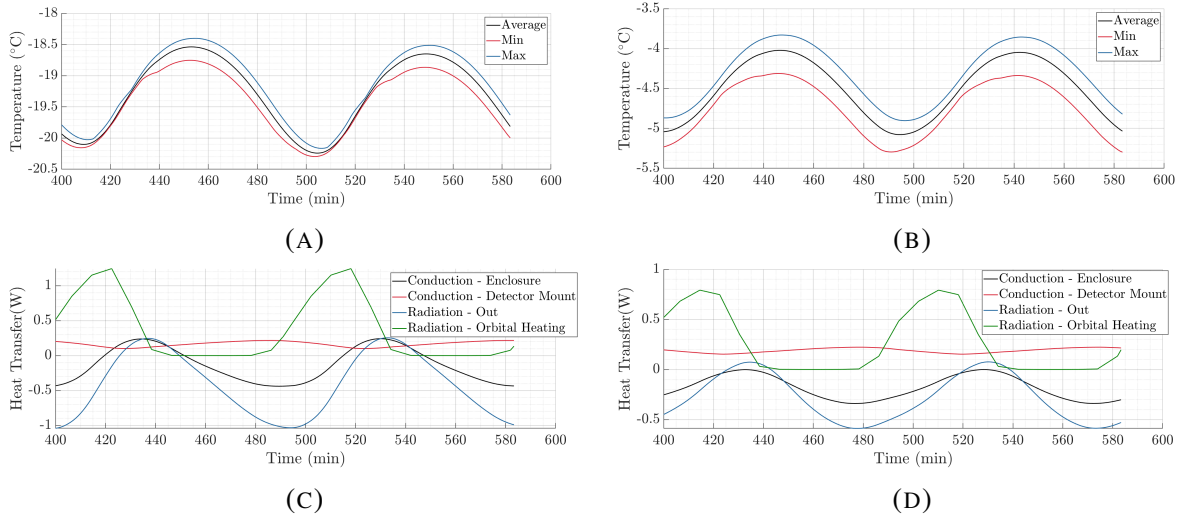


FIGURE 4.8. Temperatures after four orbits for SSO orbit cold (A) and hot (B) case in Table 4.1, and total heat transfers out of telescope structure for cold (C) and hot (D) cases.

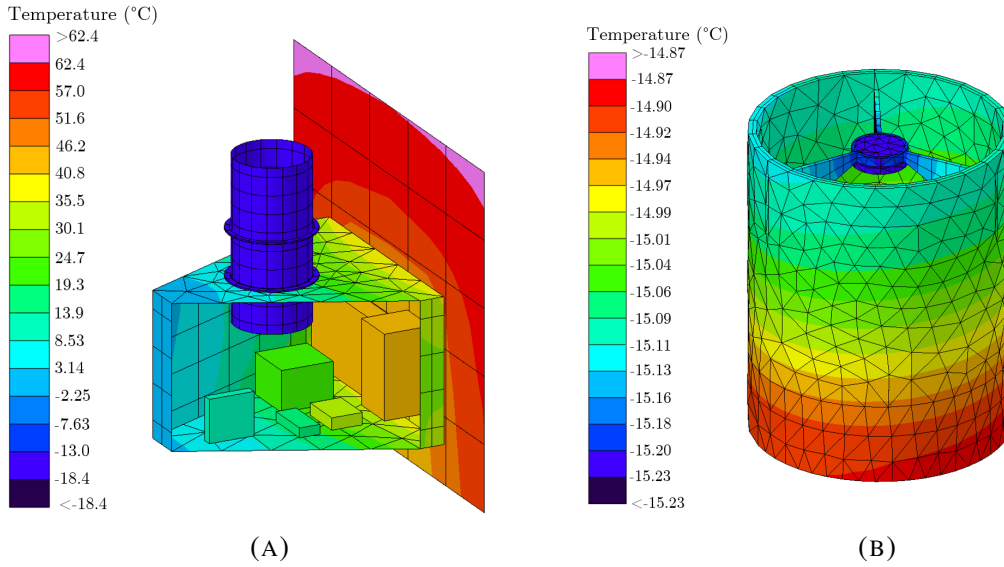


FIGURE 4.9. Example temperature distribution of spacecraft (A) and telescope (B) at a single point in orbit for the 7° inclination cold case orbit. Not the temperatures shown are for the structure beneath any MLI that is used.

4.5 Baffle Length Study

Pointing requirements on the TOLIMAN mission mean that the telescope will not slew away from the sightline to the target throughout the orbit, and as a result, the telescope will experience heating as the Earth comes into view (see Figures 4.5 and 4.7). A baffle tube will help to reduce the direct radiation from external heat sources by reducing the solid angle over which light

can enter. The length of the baffle is in part determined from optical stray light analysis. Here however, a parametric analysis of different baffle lengths was used to understand the effects of baffle length on thermal stability. All parameters are fixed within the model, except the length of the baffle was changed, and a transient simulation was run for each orbital case. As before, the solution was initialised with a steady state solution, and was then run for three orbits to lose memory of the initial state.

A summary of these results is shown in Figure 4.10. Interestingly, the effect of baffle length on the average temperature of the telescope was small, there was only a modest increase in average temperature with longer baffles despite the lowered orbital heating. A longer baffle means any heat absorbed by the system does not leave as easily, and as a result, average temperatures are higher. From Figure 4.10B, it is clear that longer baffles significantly reduce the overall temperature gradient on the structure, with the effect beginning to diminish after a baffle length of 15 cm. For the analysis done here, the baffle tube was selected to be 15 cm, which is is roughly the maximum baffle size that may be used given size constraints on the spacecraft.

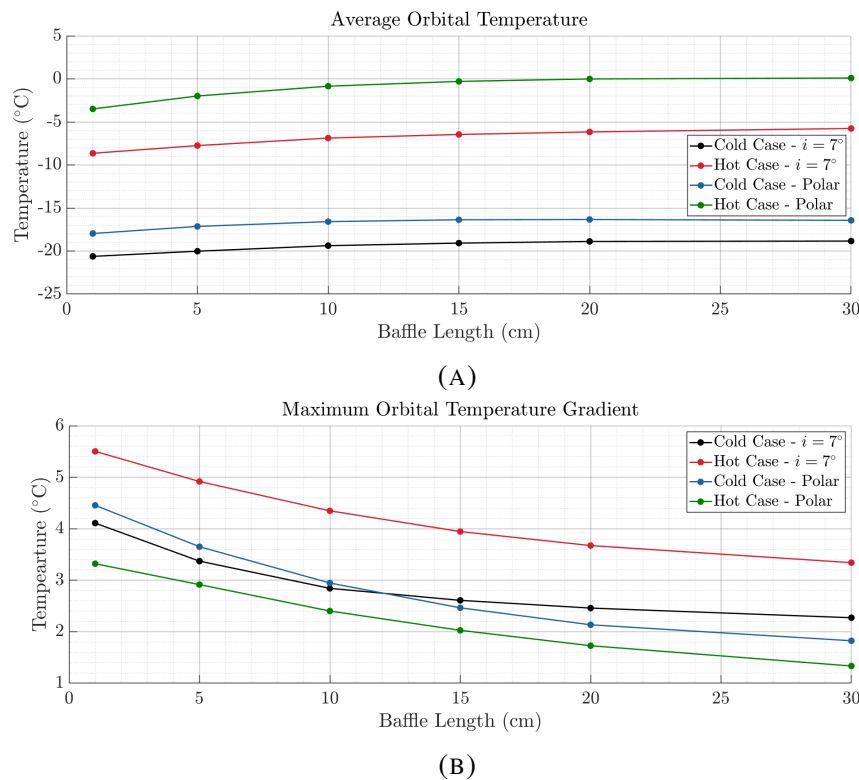


FIGURE 4.10. Average orbital temperature (A) and maximum orbital temperature gradient (B) of telescope structure for various baffle lengths at each orbital case.

4.6 Sensitivity Analysis

Uncertainties are inherent within any engineering design, particularly in the early design phase where components are not yet available for testing, or are not yet fully specified. Moreover, thermal performance and properties of components may change throughout the life of the mission due to degradation for example (See Section 2.3.2). The thermal models created here were used to study uncertain parameters, or parameters that may change with design, and to assess their effect on the telescope's temperatures. The sensitivity analysis was conducted by fixing all parameters at their nominal value, and varying one parameter over a conservative range in which the true value may lie. This was done for the telescope IR emissivity, the conductance between the telescope enclosure to the rest of the spacecraft, the contact conductance coefficient between the telescope tube and the enclosure it is mounted within, and heat generation by the detector and spacecraft. The effect on average orbital temperatures, and the maximal temperature gradient achieved on orbit was studied for each orbital worst case.

The results of the sensitivity analysis are shown in Figure 4.11. Most notably, the telescope emissivity had little effect on average temperatures, and only a small effect on temperature gradients. Figures 4.11C and 4.11D show the importance of minimizing the thermal coupling between the spacecraft and telescope enclosure, as average temperatures and gradients increased with greater coupling. The nominal conductance for the enclosure to the spacecraft was estimated to be approximately 0.05 W/K. The contact conductance between the telescope tube and enclosure only has a large effect on thermal gradients, as greater conductance allows heat to move more freely between the telescope and enclosure, but the enclosure itself has a relatively small thermal mass. Contact conductances of 10–50 W/m²/K represent an interface with a low conductivity material such as ceramic or steel, while conductances of around 800 W/m²/K represent contact with high conductivity materials such as aluminium. Finally, as one would expect, the heat generation from the detector results in an increase of average temperatures and gradients, however the increase is relatively small due to the low thermal conductivity of the Invar detector mount. Spacecraft heat generation has minimal effects due to the good nominal isolation of the telescope to the rest of the spacecraft.

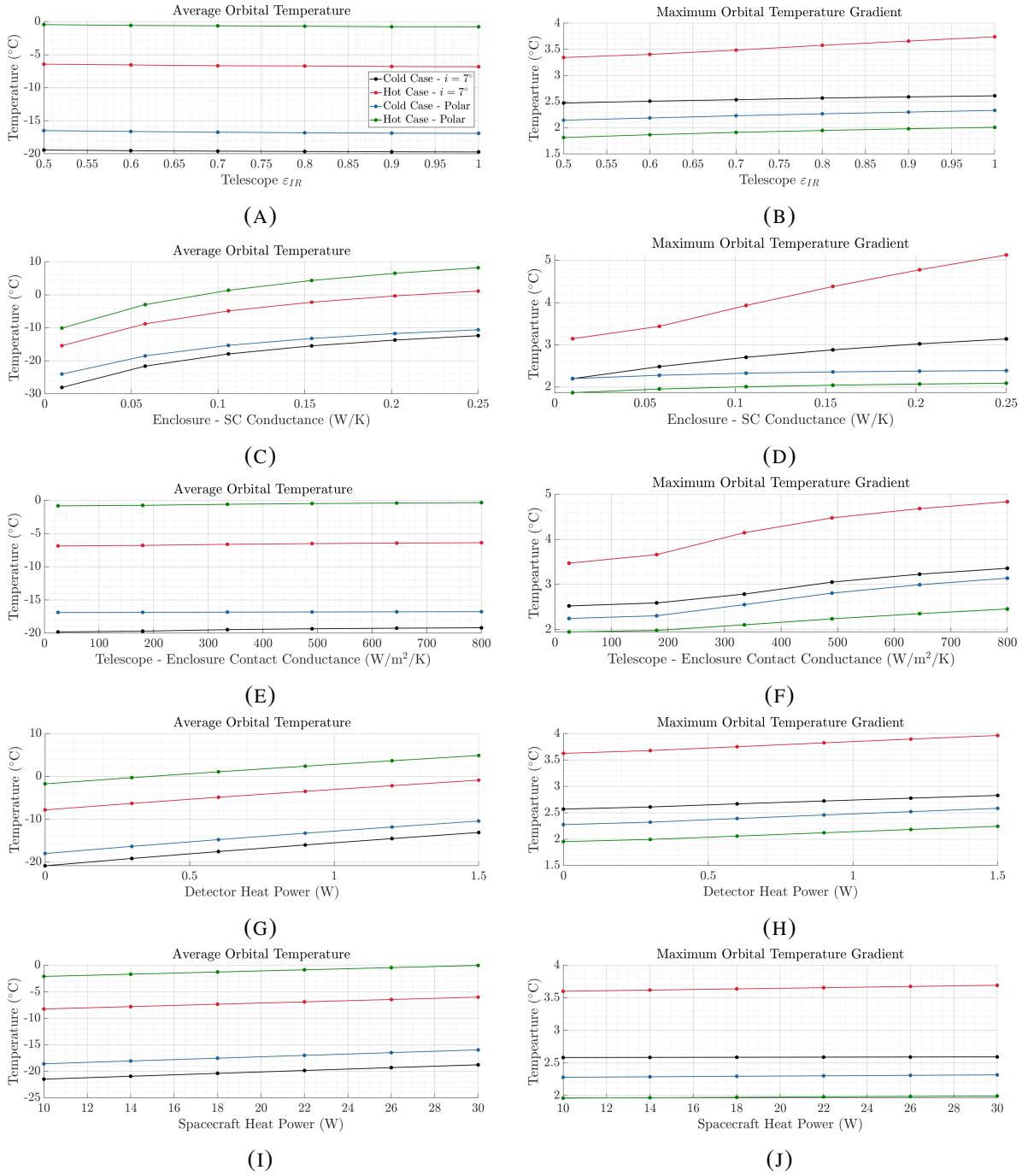


FIGURE 4.11. Sensitivity analysis parameter sweep. Panels (A), (C), (E), (G), and (I) give orbital average temperatures as the telescope IR emissivity, enclosure to spacecraft conductance, telescope tube to enclosure contact conductance, detector heat generation, and spacecraft heat generation is varied over a range of uncertain/possible values for all four worst case orbits. Panels (B), (D), (F), (H), and (J) respectively give the maximum temperature gradient achieved throughout an orbit.

Active Heater Control Design

5.1 Introduction

Having established that a cold bias can be achieved, the next stage – design of an active thermal control system using patch heaters and a feedback controller – is introduced here. This chapter focuses on heater design, and the initial creation of reduced order models for use in feedback controller design. As a starting point, a simple Proportional-Integral (PI) controller was implemented to underscore the utility of reduced order models to deliver estimates of achievable temperature stabilities for the telescope. Future work will involve more detailed analysis and design of feedback control algorithms.

5.2 Heater Placement Optimisation

Two patch heaters placed around the circumference of the telescope are used to control the temperature to 20°C. Such an arrangement is achieved using polyimide thermofoil heaters which can be manufactured to custom sizes, by Minco [78] for example. Modelling in Thermal Desktop of these patch heaters is described in detail in Appendix C4. The heaters are placed around the circumference to minimise diametric temperature gradients across the telescope: the most detrimental impact in terms of system optical performance. The placement of the heaters, shown in Figure 5.1, affects in the way in which heat is distributed around the telescope, and therefore the arrangement strives for the most uniform temperatures. Determining this arrangement is not trivial since the thermal masses, conductances, and heat inputs vary with the length of the telescope.

Optimisation routines previously used to determine worst case orbits in Section 4.4 were repurposed here to determine the optimal heater placement. The location of the heaters was

parameterized as shown in Figure 5.1B, and h_1 and h_2 were determined to give the most uniform temperatures. Here the optimal heater placement is the one which results in the smallest maximum temperature difference from 20°C over the entire telescope structure for an orbital average steady state solution.

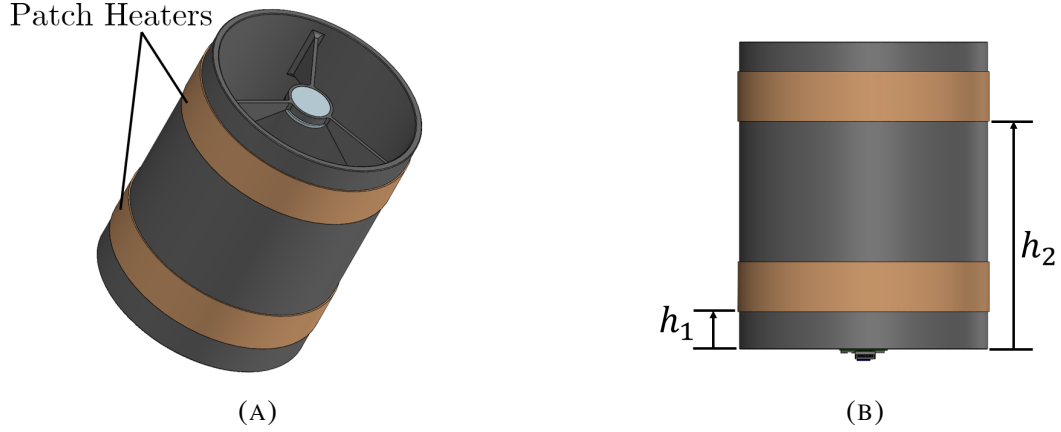


FIGURE 5.1. Patch heater placement on telescope, where the location of the patch heaters is parameterized by h_1 and h_2 describing their position as explored by the optimisation routine.

Mathematically, the optimisation objective is written as

$$\begin{aligned} &\text{minimise: } \max_{\text{nodes}} \{|T_{\text{telescope}} - 20^\circ\text{C}|\} \\ &\text{subject to: } 0 \leq h_1 \leq \frac{h_{\text{max}}}{2} - w, \quad \frac{h_{\text{max}}}{2} \leq h_2 \leq h_{\text{max}} \end{aligned}$$

where $w = 3 \text{ cm}$ is the width of the heater and $T_{\text{telescope}}$ is the nodal temperatures of the telescope. The temperature sensors used in the control of each respective heater were placed in a strip beneath the heater, with the average temperature being used to control the heater. These were idealistically represented as finite difference surfaces in the thermal model (Figure C.3). For this optimisation routine, it was assumed that heaters can maintain the temperature at their respective measuring regions of 20° perfectly. This assumption eliminates the influence of imperfect controller performance on the results. While perfect stability can be achieved easily in a steady state simulation, this is not possible in a transient simulation as it would require a feedback controller that can perfectly track temperatures: if such a thing existed then much of the work here would be done. Furthermore, performing steady state analysis is much less computationally expensive compared to full orbital transient simulations, especially with feedback controllers that require small time stepping. The optimisation was therefore performed

for orbital steady state solutions, spanning all four orbital cases and using the procedure in Thermal Desktop described in Section 4.4 for determining worst case orbits. The resulting optimal patch heater placements and steady state distributions for 7° cold case orbit and SSO hot case orbit are shown in Figure 5.2.

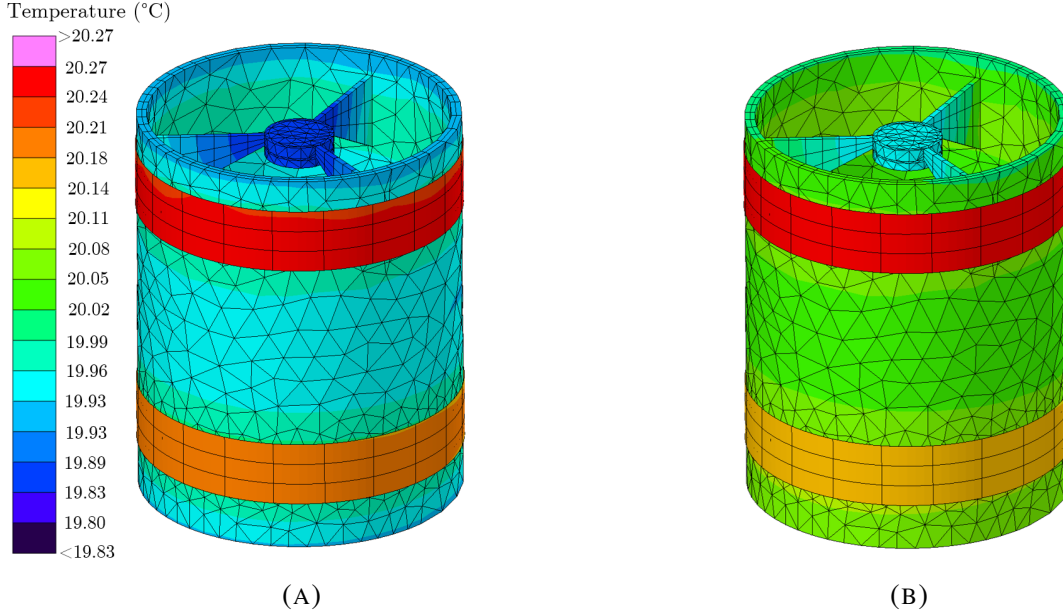


FIGURE 5.2. Steady state temperature distributions on the telescope structure obtained after heater placement optimisation routine using orbital average heating rates for 7° inclination cold case orbit (A) and SSO hot case orbit (B). Both figures use the same temperature scale.

The coldest parts of the entire structure are the spiders, which have very low heat transport due to their thin cross section providing a high resistance thermal path. As a result, the top heater location is directly around the mounting region of the spiders. The lower heater is located around the primary mirror mounting location. This gives maximum heat transport to the primary mirror, which loses heat through radiation to space. Also worth noting are the cold regions due to conductive heat losses at the top and bottom faces of the telescope tube, where it makes contact with the mounting structure.

5.3 Model Order Reduction

The complete spacecraft and telescope thermal models are computationally expensive to simulate, and give little quantitative information on the behaviour of the system. Consequently, using these models for feedback controller design is impractical. A more informative and

efficient approach is to create a reduced order model – a simple approximation of the system that is fast to simulate, and comes in the form of an equation.

5.3.1 Reduced Order Models

The telescope was reduced to a simple mathematical model composed of two lumped capacitances, with a thermal resistor joining them, as shown in Figure 5.3. The symmetry of the telescope about the optical axis means that lumped nodes can be placed in a line along the optical axis. This model is an extreme simplification of the true telescope, however, as will be seen, this proves to be adequate for our purposes. Furthermore, the ideas used to create this simple model can be extended to create more accurate models, with a greater number of nodes. The reduced order model was put in a Linear Time Invariant (LTI) state space form, which is

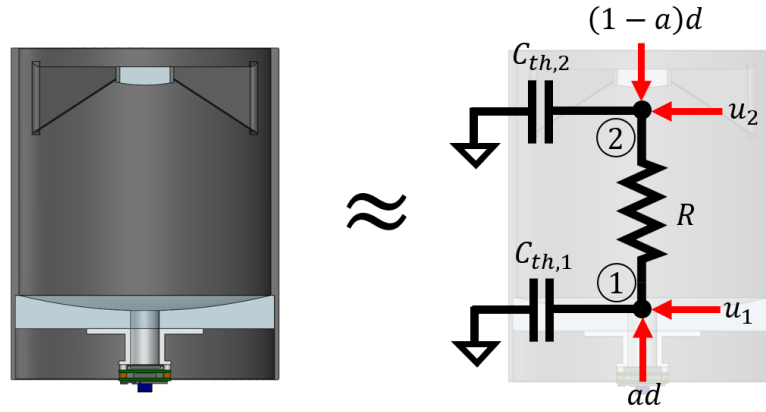


FIGURE 5.3. The two thermal capacitance-resistor model used in creation of the reduced order model. Variables u_1 and u_2 represent heater inputs to each node, d is the disturbance input, a is the fraction of the disturbance on each node, $C_{th,1}$ and $C_{th,2}$ are the heat capacities of each node, and R is the thermal resistance between each node.

natural for this system since the equations for heat conduction are themselves linear. For each node in Figure 5.3, a first order ordinary differential equation (ODE) can be written by taking the time derivative of the definition of specific heats (Equation 2.2) and including conduction terms

$$\begin{aligned}\dot{T}_1 &= \frac{1}{C_{th,1}R}(T_2 - T_1) + \frac{1}{C_{th,1}}u_1 + a\frac{1}{C_{th,1}}d, \\ \dot{T}_2 &= \frac{1}{C_{th,2}R}(T_1 - T_2) + \frac{1}{C_{th,2}}u_2 + (1-a)\frac{1}{C_{th,2}}d,\end{aligned}\tag{5.1}$$

where R is the thermal resistance of the conductive coupling between the two nodes, $C_{th,1}$ and $C_{th,2}$ are the capacitances or heat capacities of each node, u_1 and u_2 are the heat inputs

from the heaters, d is the total disturbance on the telescope, and $0 \leq a \leq 1$ is a factor which determines the relative weighting of the disturbance on the nodes. The disturbance d was treated as an unknown input, and represents all heat transfers besides the heaters: orbital heating, emitted radiation, and conductive transfers with the rest of the spacecraft. A good passive control design will try to minimize the magnitude of d , allowing for better temperature control. The model in Equation 5.1 is a linearization of the true system dynamics about the set point temperature of 20°C. Radiation heat transfer is fundamentally non-linear, having a T^4 dependence, however, over a small temperature range, a linear approximation can be made. This type of model is limited in that it does not capture the phase delay between the heater input and nodal temperature due to the thermal mass of the heaters themselves, and the contact conductance to the telescope structure. This effect was modelled in Thermal Desktop by including extra nodes to represent each heater, the same can be done for the reduced order model.

Equation 5.1 contains four free parameters: R , $C_{th,1}$, $C_{th,2}$, and a . In principle, these parameters could be determined manually by dividing the telescope into two regions with effective thermal properties assigned from calculations. This strategy requires trial and error and will ultimately not result in the optimal solution, especially when trying to account for non-linear terms. Instead, these parameters are fit using simulation data from the Thermal Desktop model. As a state space system, Equation 5.1 can be written as [79]

$$\begin{aligned}\dot{x} &= Ax + Bu + Fd \\ y &= Cx + Du,\end{aligned}\tag{5.2}$$

where $x = [T_1 \quad T_2]^T$, $u = [u_1, \quad u_2]^T$, and the state space matrices are given by

$$A = \begin{bmatrix} -c_1 & c_1 \\ c_2 & -c_2 \end{bmatrix}, \quad B = \begin{bmatrix} b_1 & 0 \\ 0 & b_2 \end{bmatrix}, \quad F = \begin{bmatrix} ab_1 \\ (1-a)b_2 \end{bmatrix}, \tag{5.3}$$

C is the 2×2 identity matrix (full state output), and D is the 2×2 zero matrix. The coefficients in Equation 5.1 were lumped into the parameters c_1 , c_2 , b_1 and b_2 as this allows for better results in the fitting algorithm, and is a more general way for representing the linearization. The form of Equation 5.2 with the included disturbance term Fd is not the standard form of a state space system, and cannot be used directly with the control tools provided by MATLAB. Treating the

disturbance d as another (unknown) input, the system is written as

$$\begin{aligned}\dot{x} &= Ax + \begin{bmatrix} B & F \end{bmatrix} \begin{bmatrix} u \\ d \end{bmatrix} = Ax + \tilde{B}\tilde{u} \\ y &= Cx + \tilde{D}\tilde{u},\end{aligned}\tag{5.4}$$

where $\tilde{D}$ is the 3×3 zero matrix, and the system can be thought of as a three-input, two-output system. Although not considered in depth here, presenting the system in state space form is especially useful in the design of more complex feedback control structures, especially when multiple inputs and outputs are considered [79].

5.3.2 Parameter Fitting

Transient simulations in Thermal Desktop were used as “experiments” for fitting the parameters in Equation 5.3 using the MATLAB System Identification Toolbox. The process of fitting free parameters to a known mathematical model is referred to as “grey-box” model estimation. In terms of the model in Equation 5.4, the output y was taken to be the temperatures measured by the temperature sensors under the heaters. Input-output time series data can then be used to fit parameters.

The input sequence should provide excitation primarily in the frequency range where the model needs to be accurate [80]. There is a limit to how fast the temperatures on the telescope respond to rapid changes in heat input, both due to the thermal inertia and contact conductance of the heaters and telescope structure, as well as the electronics used to power the heaters. As a result, very high frequency inputs, greater than approximately 1 Hz are attenuated by the system. The thermal disturbances that the system will encounter occur over timescales on the order of an orbital period, corresponding to frequencies of 0.17 mHz for a 550 km orbit (orbital periods of ~ 5750 s). A white noise signal, filtered with a 3rd order low-pass Butterworth filter and a cutoff frequency of 0.4 mHz was used as the heater input sequence. These filter parameters were determined through trial and error to give good fitting performance. The mean value of the input sequence was chosen such that it maintains an average temperature of 20°C, which is the temperature range over which the system is linearised, and where accuracy is required. The standard deviation of the signal was set to be 0.65 times the mean value. An example input signal is shown in Figure 5.4.

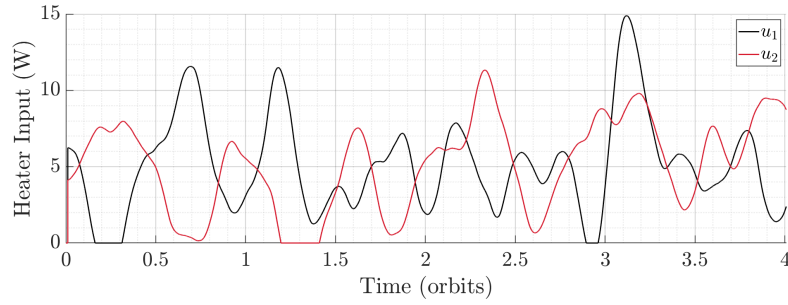


FIGURE 5.4. Example heater input sequence used for system identification. A 3rd order Butterworth low-pass filtered white noise signal, with a cutoff frequency of 0.4 mHz, mean signal value of 5 W, and standard deviation of 3.25 W.

These input signals were given as a time series heat input to the heaters in Thermal Desktop. Nodal temperatures as a function of time, and the total heat disturbance d on the telescope were exported and used as output data in the parameter fitting. The term d is calculated as the sum of heat transfers to the telescope submodel due to conduction with the spacecraft and the detector, and due to radiative terms to space, from Earth, and from the Sun. Parameter fitting was done by first creating an *iddata* object in MATLAB, where all four orbital cases in Table 4.1 are used for parameter fitting. The *greyest* function was used to determine the unknown system parameters in Equation 5.3 given the *iddata* object.

The non-zero set point temperature of 20° about which the system was linearised represents a constant term in the state space system equations, meaning the system is not truly linear. This is dealt with by making a coordinate transformation such that the equilibrium temperature is zero. In the system identification process, this is done by subtracting the mean value from each of the signals [79, 80]. This mean value is then added back to transform back to the true temperatures.

Results for the 7° inclination orbit are shown in Figure 5.5 for both the training data used to fit the parameters, as well as an independent validation data set – created with an input signal different to the one used for parameter fitting. Results for the other three orbital cases are given in Appendix A1. The fitted state space matrices from Equation 5.3 are

$$A = \begin{bmatrix} -6.06 & 6.06 \\ 4.51 & -4.51 \end{bmatrix} \times 10^{-3}, \quad B = \begin{bmatrix} 1.01 & 0 \\ 0 & 1.39 \end{bmatrix} \times 10^{-3}, \quad F = \begin{bmatrix} (0.5)4.62 \\ (1 - 0.5)6.9 \end{bmatrix} \times 10^{-4}. \quad (5.5)$$

Notably, the smaller magnitude of the second row of the matrix A implies that node 1 – the node situated around the primary mirror – has a greater thermal mass. Furthermore, the terms

in the matrix F are smaller than in matrix B , indicating that disturbance inputs are relatively small compared to heater inputs, as expected. Ideal passive control design would drive towards minimization of the matrix F as much as possible.



FIGURE 5.5. System identification results for 7° inclination Cold Case orbit. Note all orbital cases were used in the fitting of parameters. Panels (A), (C), (E), (G), and (I) show the nodal temperatures, heater inputs, and disturbance inputs for the training data. Panels (B), (D), (F), (H), and (J) show the same results for a validation case.

From Figure 5.5, it is clear that the temperature of the telescope and the disturbance input are correlated, since conductive and radiative heat transfers depend on temperature. A simple linear model of this unknown disturbance was created for use in controller design and simulation. We assume the disturbance input will take the following form

$$d = f(t) + \alpha T + \beta, \quad (5.6)$$

where $f(t)$ is a periodic function centered at zero, and represents periodic orbital heating, α is a heat transfer coefficient for linear heat transfer, and β is a constant to account for the fact that there will be linear heat transfers even when $T = 0$. Since $f(t)$ is periodic and centered at zero, its average value is zero, which means taking the average of both sides gives

$$\langle d \rangle = \alpha \langle T \rangle + \beta, \quad (5.7)$$

where $\langle \cdot \rangle$ indicates a time average. Equation 5.7 indicates that the constants α and β can be found using a linear least squares fit between T and d as obtained from the Thermal Desktop simulations. Orbital disturbances are approximated as $f(t) = A \sin(2\pi t/T_{orb})$, where $T_{orb} \approx 5750$ s is the orbital period, and A is the heating amplitude, found through trial and error. Results for this fit can be found in Table 5.1, and a comparison with the disturbance obtained from the full simulation is shown in Figure 5.6 for all four orbital cases.

TABLE 5.1. Coefficients use for linearised disturbance model given by Equation 5.6

Orbit	Case	α (W/K)	β (W)	A (W)
7°	Cold Case	-0.352	-7.51	1.1
	Hot Case	-0.277	-4.72	1.5
SSO	Cold Case	-0.378	-6.69	1.6
	Hot Case	-0.336	-3.74	1.0

The coefficient α has similar magnitude in all cases, as roughly this corresponds to the thermal coupling with the spacecraft, and does not change between orbital cases. The constant term β on the other hand describes the average value of the disturbance, and is larger in magnitude for the colder orbital cases, representing a greater overall heat loss. Figure 5.6 illustrates that the model given by Equation 5.6 is reasonable, and captures the magnitude and general trend of the disturbance. However it fails to replicate the higher frequency components in the disturbance, as expected from a simple linear model. This model only needs to be fit for initial estimates for use in controller design, with the Thermal Desktop model being used for more detailed refinement and analysis.

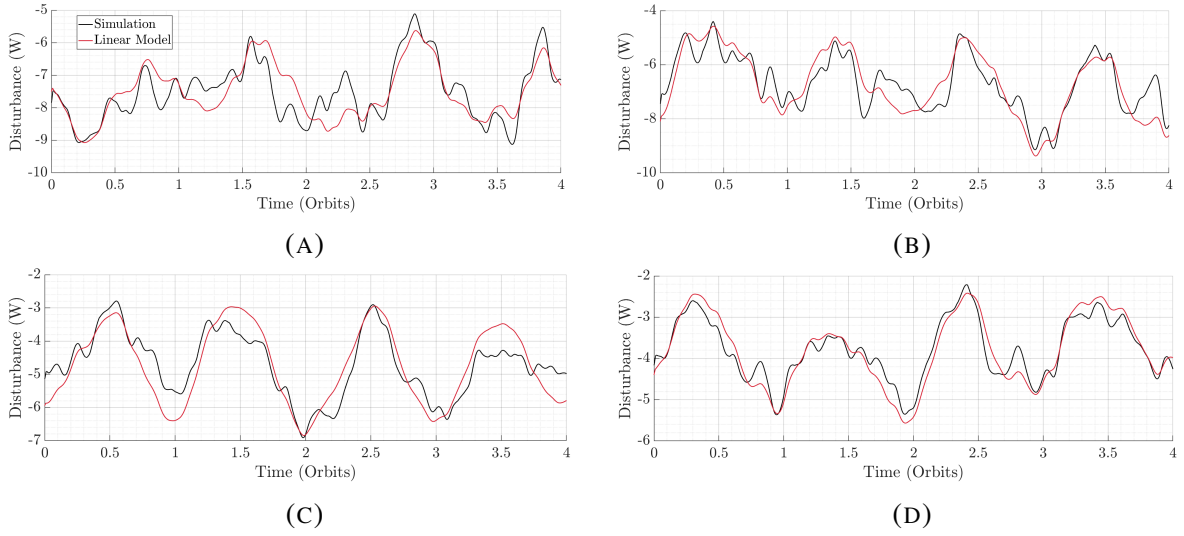


FIGURE 5.6. Linearised disturbance model from Equation 5.6 for 7° inclination cold (A) and hot (D) cases, and SSO cold (B) and hot (D) cases for coefficients in Table 5.1 compared to simulation data obtain from Thermal Desktop.

5.4 Feedback Controller Design and Simulation

Two independent PI feedback control loops are used to control each heater. These two control loops will be coupled – the heat input of one heater will affect the temperatures measured by the other, particularly due to the high thermal conductivity of the telescope structure. Although not implemented here, controllers that account for these loop couplings can be used by designing a decoupling matrix to minimize these interaction effects, essentially turning the system into two non-interacting loops [81, 82, 83]. Alternatively, state feedback techniques such as pole placement, or optimal control through a Linear-Quadratic Regulator (LQR) can be employed [79]. It must be noted however, that since the heaters can only supply a positive limited input, another element of non-linearity is added. Accounting for these non-linearities completely requires techniques such as Model Predictive Control (MPC) or the currently emerging Predictive Thermal Control (PTC) which is in development to enable ultra-thermally stable space telescopes [84]. These ideas lie outside the scope of this thesis, however, the state space reduced order models introduced in the previous section are a necessity when implementing such methods.

In parallel form, a PI controller can be written as [79]

$$u(t) = K_p e(t) + K_i \int_{t_0}^t e(t) dt, \quad (5.8)$$

where $u(t)$ is the control (heater) input, $e(t) = T_r - T$ is the error with T_r being the reference or set-point temperature and T the measured temperature, and K_p and K_i are the proportional and integral control gains. A derivative term can be added (to give PID control), however, thermal systems, which can be written as first order ODEs (Equation 5.1) are highly damped, and derivative control is not necessary. Two PI controllers of this form were used to control each heater independently. The measured temperature is the average temperature measured beneath each heater. In principle, it is possible to use sensors placed in other regions of the telescope, however, due to the slow time constants of a thermal system, this will introduce large phase lags between the control input and measured temperatures, yielding poor results. Phase compensators can be used to mitigate these effects, at cost of increases to system complexity, and decreased reliability [79].

The selection of gains K_p and K_i employed a trial and error approach, making use of the reduced order models from Section 5.3.2, simulated in MATLAB, to obtain rough estimates, and the complete Thermal Desktop model for fine tuning and analysis. PI control was implemented in Thermal Desktop using a PID logic object. Initially only the proportional gain was adjusted to give the desired step response, and then the integral gain was introduced to eliminate steady state error. Integrator anti-windup via clamping was implemented in the controller, which works by stopping integration when the controller input is saturated. As part of the preliminary mission requirements, 20 W of power has been allocated to heater control, which places the limit $0 \leq u_1, u_2 \leq 20$ W, with $u_1 + u_2 \leq 20$ W. The controller was designed with minimising overshoot as the primary requirement, since heater control cannot work to actively reduce temperatures; the controller only has authority over the measured temperature when it is below the set point. Initially the system was run for 35 000 seconds without heaters to clear the memory of the initial condition, then the heaters were activated. Reduced order models were simulated by solving the system in Equation 5.4 numerically with the PI control input calculated at each time step.

The final control gains used were $K_{p,1} = 16.8$ W/K and $K_{i,1} = 5 \times 10^{-5}$ W/K/s for heater 1, and $K_{p,2} = 18.2$ W/K and $K_{i,2} = 7.5 \times 10^{-5}$ W/K/s for heater 2. It was found that heater 2 (the heater around the spiders) required more aggressive control gains to account for greater heat losses. Comparison between the Thermal Desktop and reduced order models are shown for the 7° inclination cold case orbit in Figure 5.7, while the rest of the orbital cases can be found in Appendix A2.

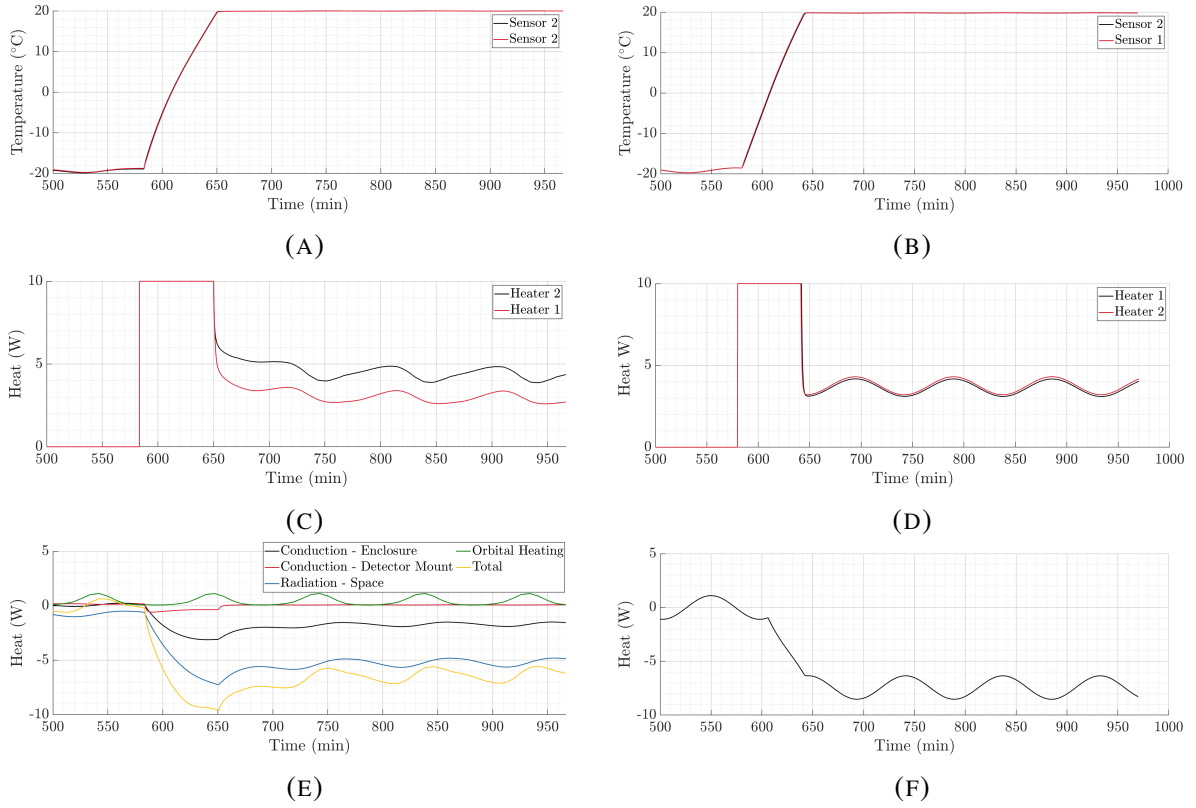


FIGURE 5.7. PI controller simulation for 7° cold case orbital condition. Panels (A), (C), and (E) show the sensor temperatures, heater input, and heat transfers for the Thermal Desktop model. Panels (B), (D), and (F), show the equivalent simulation using the reduced order state space models, with (F) being the disturbance input.

Of particular note is the accuracy of the reduced order model in Figure 5.7B compared to the full simulation in Figure 5.7A. The time from the onset of heater function until both sensors reach a temperature of 19.5°C is approximately 65 minutes for the full simulation, and 58 minutes for the reduced order model. This is an excellent result considering the simplicity of the reduced order model. Arguably, its limitation lies in the accuracy of the disturbance model used. There are considerable differences between the heater inputs of the two models, and the same can be said for the disturbance model, which under-predicts the total heat disturbance on the telescope.

Given the accuracy of the reduced order model with only two nodes, the addition of more carefully chosen nodes will considerably increase accuracy. In particular, the addition of nodes around the circumference of the telescope can be used to capture diametric temperature gradients. In general, more nodes should be added in regions where large temperature gradients

are expected, since these are the regions that must more accurately capture the continuity of the structure.

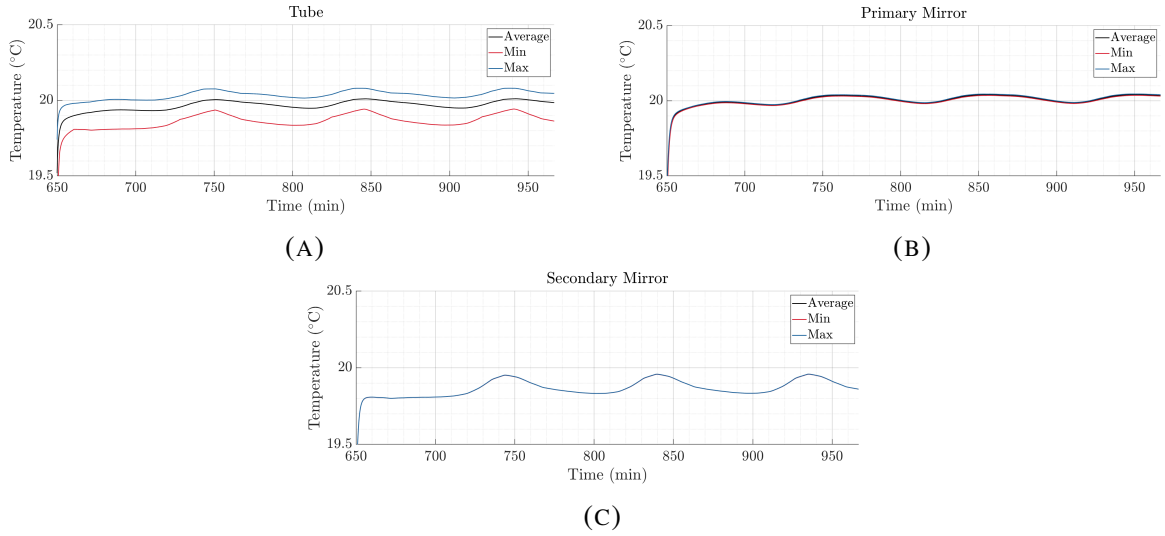


FIGURE 5.8. Telescope tube (A), primary Mirror (B), and secondary mirror (C) temperatures for the 7° inclination cold case orbit from Thermal Desktop model using PI controller.

More detailed orbital temperature data is shown in Figure 5.8 from the Thermal Desktop model for the 7° inclination cold case, with the rest of the cases given in Appendix A2. In all orbital cases, temperature stabilities of $20 \pm 0.05^\circ\text{C}$ on the primary mirror, and $20 \pm 0.25^\circ\text{C}$ on the secondary mirror and tube were achieved. Figure 5.8A shows that the tube structure can have thermal gradients larger than 0.25°C , with most of these gradients coming from the spider structure, as was seen in Figure 5.2. The secondary mirror was consistently below the set point temperature due to the high resistance thermal path created by the spider arms.

As expected, the time for heaters to reach the set point temperature was less for the hot case orbits. Interestingly, both for the 7° inclination and SSO hot case, secondary mirror temperatures on average were lower than the corresponding cold cases. From Figures A.4A and A.8A, the sensor temperatures accurately track the set point, however, the tube temperatures in these cases (Figures A.5A and A.9A) have greater temperature gradients. This arises from the larger swings in heating rates associated with hot cases, due to the more direct heating from the Sun and Earth. This in turn means the telescope has less time to approach uniform equilibrium temperatures. Furthermore, different controller tuning parameters may favour different orbital conditions, yielding better results in some cases. In this case, the controller should be designed with a particular orbit in mind.

When the heaters cause the measured temperatures to exceed the set point, they must switch off and wait until the temperature naturally settles below the set point so they can be actively controlled again. In hot case orbital conditions, the magnitude of the cold bias, that is, the heat loss that reduces the telescope temperatures, is smaller. Hence in these cases, settling times are much longer. The periodic nature of orbital heating means that small overshoots in temperature are unavoidable, particularly when the spacecraft is exiting eclipse. The result is in general poorer control performance in hotter orbital cases. The trade-off is that colder cases require more heater power to maintain the set point temperature.

Structural Stability Analysis

6.1 Introduction

Thermal analysis results need to be integrated with structural models in order to calculate the deformation of optical elements, and hence understand the effects on imaging performance. This chapter introduces the structural model for such analysis, as well as calculation of the sag profile of the primary mirror after deformation. Analysis was restricted to the primary mirror of the telescope since this is the critical component that must remain stable in the optical system. However the methods developed are general and can be extended to other components. The tools and results introduced here enable thermo-mechanical analysis to be integrated directly with optical analysis done by the TOLIMAN team.

6.2 Structural Finite Element Model

The structural finite element model created in ANSYS Mechanical is shown in Figure 6.1. A linear elastic analysis was used to calculate the deformations of the entire telescope. To study deformations of the mirror, it was necessary to model the entire telescope structure to account for differences in relative thermal expansion and contacts between components. Material properties used in this analysis can be found in Table B.1, while the mesh convergence study used in determining the element size for this model is detailed in Appendix E.

All contacts within the model were taken to be bonded, meaning no relative movement can occur between components in contact, including the detector mount. For the current analysis, this assumption was reasonable, particularly since the mirrors are bonded with epoxy to the structure. Boundary conditions applied to the telescope were chosen to represent the ideal mounting condition inside the enclosure. The telescope is mounted by the top and bottom

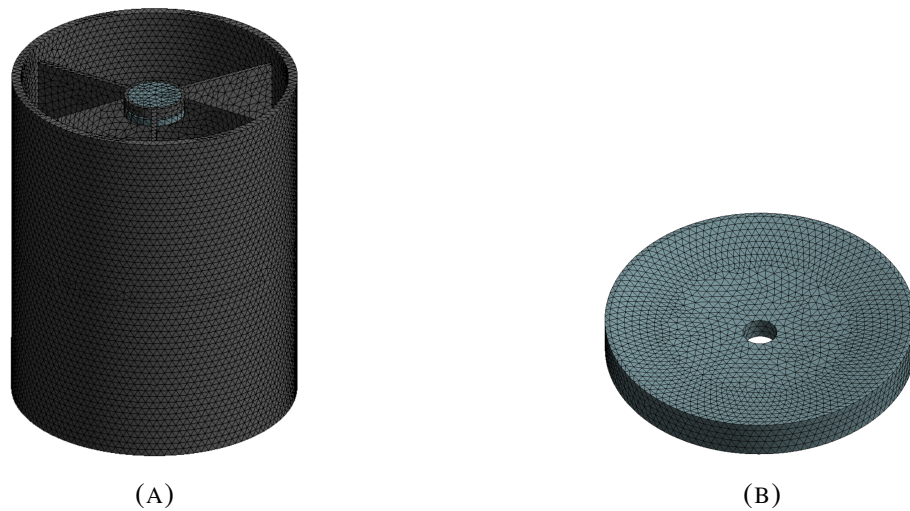


FIGURE 6.1. Finite element structural model of the telescope (A) and primary mirror (B) in ANSYS Mechanical used for deformation analysis.

surfaces within the enclosure, and a flexure mounting system means the telescope is “floating” to minimize further stresses induced by thermal expansion. The bottom mounting surface of the telescope was constrained in motion parallel to the optical axis, but not perpendicular to it (a frictionless support in ANSYS Mechanical). To fully constrain the model, a single, arbitrary point on this surface was fixed from moving in the direction horizontal to the optical axis. This point acts as a datum relative to which all displacements are measured.

Using the Thermal Desktop post processing data mapper functionality, the structural mesh was superimposed over the thermal mesh, and temperatures were interpolated onto the nodes of the structural mesh using the shape functions of the thermal finite elements. This returns an output file containing applied temperatures to the nodes of the structural model in the form of APDL *BF* commands, which were used to apply the thermal load on the structural model in ANSYS Mechanical.

6.3 Calculation of Grid Sag Profile

The mirror sagitta (sag), denoted by s , is a measure of the mirror shape, and represents the “depth of curve” of the mirror, as shown in Figure 6.2. The deformation field of the primary mirror was used to calculate a change in sag, Δs from the undeformed mirror shape – a useful indicator of the dimensional stability of an optical surface. The change in sag was represented

on a uniform grid, which can be used directly in commercial optical design software such as Zemax.

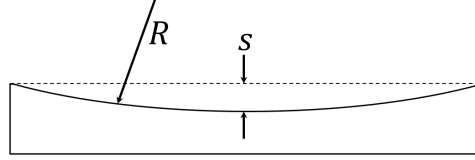


FIGURE 6.2. Sag s of primary mirror and radius of curvature R .

The software package SigFit [85] can be used to link structural deformation analysis to optical design software, however this was unavailable for use here. Instead, interpolation of mirror deformations onto a uniform grid for the calculation of sag profiles was performed using MATLAB. Interpolation can be accurately achieved using the quadratic tetrahedron shape functions used to solve the finite element solution. However, this requires obtaining a relationship between coordinates in physical space to the natural coordinates of the finite elements. Unfortunately there is no closed form solution for variable metric tetrahedral elements as used in this structural model to capture the correct mirror curvature [86]. Another approach which provides good accuracy for high performance optical systems is linear interpolation using Delaunay triangulation [87]. Mirror surface node positions and their displacements were output from the structural model, and interpolated onto a uniform grid using the *scatteredInterpolant* function in MATLAB. Prior to interpolation, the mean value of all displacements was subtracted from the displacements in each respective direction to remove rigid body transitional components of deformation. This has no affect on the shape of the deformed mirror, but is necessary because an (arbitrary) fiducial point was fixed within the finite element model as described in the previous section.

The change in sag cannot be simply taken to be the z component of displacement at each corresponding point, as shown in Figure 6.3. The position of the displaced node corresponds to a different location in the plane of the mirror due to the radial component of displacement. Since the displacements of the mirror are very small compared to the size of the mirror, this effect is accounted for through the following equation [87]

$$\Delta s = \Delta z - \frac{\partial z(r_0)}{\partial r} \sqrt{\Delta x^2 + \Delta y^2}, \quad (6.1)$$

where Δx , Δy , and Δz are the displacements of a point on the mirror, and the second term is the first order correction for Δz to give Δs .

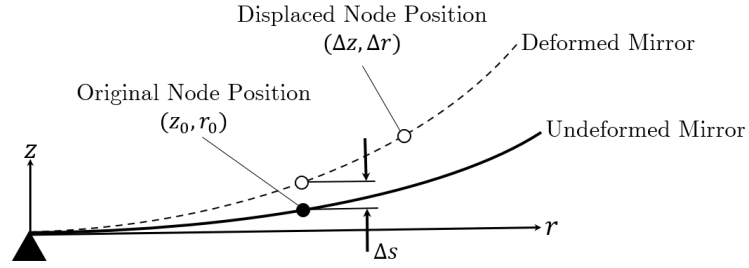


FIGURE 6.3. A displaced node on the mirror surface will correspond to a different point on the mirror due to displacement components in the radial direction.

The partial derivative $\frac{\partial z(r_0)}{\partial r}$ was evaluated for the undeformed mirror, and was calculated analytically from the standard sag equation which defines the mirror surface

$$s = \frac{cr^2}{1 + \sqrt{1 - (1 + k)c^2r^2}}, \quad (6.2)$$

Here c is the curvature of the surface, k is the conic constant, and $r^2 = x^2 + y^2$ is the radial coordinate. It can be taken that $\frac{\partial z(r_0)}{\partial r} = \frac{\partial s(r_0)}{\partial r}$ since s only differs from the z coordinate by a constant. For the TOLIMAN mirror, the surface profile is defined by the constants $c = -3.47021$ m and $k = -1.003145$. Since the curvature c is defined to be negative, the sag of regions away from the centre of the mirror are negative. A 2D representation of the undeformed mirror sag profile is given in Figure 6.4. After interpolation from the finite element nodes to a 1024×1024 uniform grid, Equation 6.1 was used at each point to give a uniform grid of Δs . This grid was then added onto the undeformed mirror sag surface defined by Equation 6.2 to obtain the sag profile of the deformed mirror if required for optical analysis. Here only Δs is studied as this is the most informative for looking at deformations of the mirror surface.

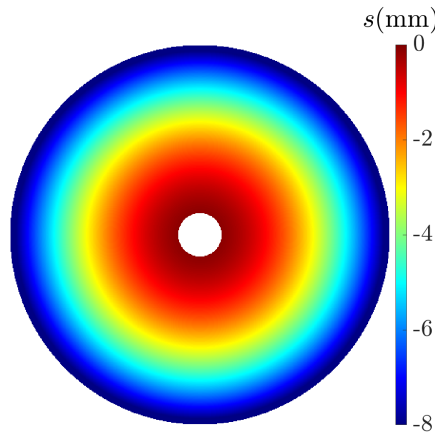


FIGURE 6.4. Sag profile of undeformed primary mirror.

6.4 Uniform Temperature Soak

The general effect of temperature change on the sag profile of the primary mirror was understood first by looking at uniform temperature changes in the entire telescope – a uniform soak temperature. This simple analysis gives a first point of reference for the order of magnitude of displacements that a given temperature change may produce. Figure 6.5 shows the sag profile for temperature changes of 0.1°C and 0.25°C greater and less than the operating temperature of 20°C .

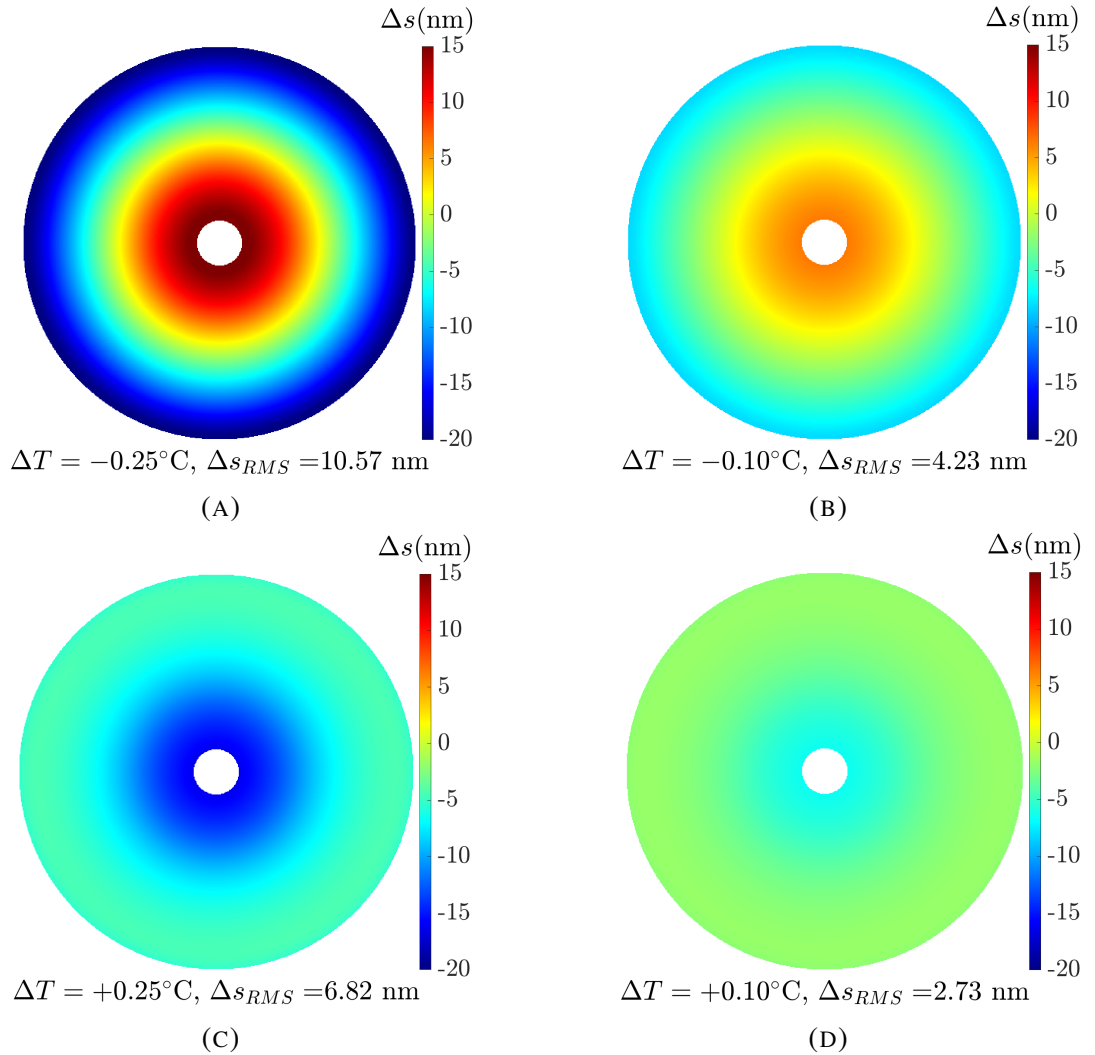


FIGURE 6.5. Δs profiles of primary mirror for uniform temperature soak for temperature changes of $\Delta T = -0.25^\circ\text{C}$ (A), $\Delta T = -0.1^\circ\text{C}$ (B), $\Delta T = +0.25^\circ\text{C}$ (C), and $\Delta T = +0.1^\circ\text{C}$ (D) from the set point temperature $T = 20^\circ\text{C}$.

As expected, all Δs distributions are symmetric about the optical axis – there is no significant asymmetry in the telescope structure or the applied temperature condition. Interestingly, soak

temperatures below 20°C result in significantly larger deformations of the primary mirror than temperatures greater. Δs_{RMS} is nearly twice as large for $\Delta T < 0$. Also of note is that for temperatures above and below the set point temperature, Δs is always negative at the outside regions of the mirror, and is negative over the entire mirror for $\Delta T < 0$. When the temperature is decreased, the entire telescope contracts, however the CTE of the primary mirror is slightly lower than that of the structure (Table B.1), as a result, stresses are induced due to the relative difference in expansion of the two materials. Figure 6.6A shows the Von-Mises stress on the telescope tube for a soak temperature of $\Delta T = -0.25^\circ\text{C}$. A band of higher stress can be seen around the mounting region of the primary and secondary mirrors due to this difference in CTE. These stresses restrict the primary mirror from contracting, and hence change the deformation profile. Similarly, when $\Delta T > 0$, the same effect occurs, but this time the tube expands more than the mirror, and since the mirror is bonded, the mirror is “pulled” in the radial direction. These stresses are very small in magnitude, less than 1 MPa, and so will not cause yielding in the structure. An important factor for the mechanical design will be to avoid stress induced by the detector mount which is assumed to be Invar 36 with a much lower CTE than the telescope.

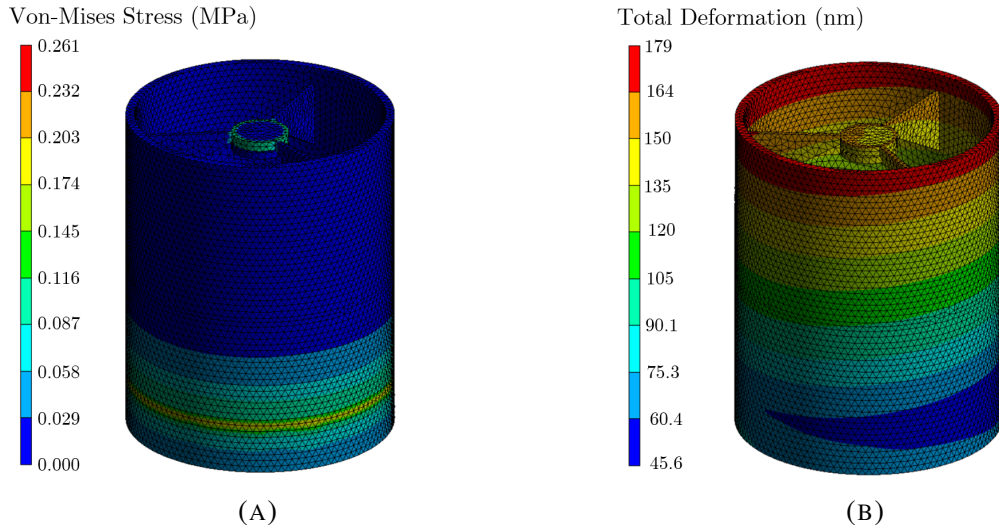


FIGURE 6.6. Von-Mises equivalent stress on the telescope structure (A) and total deformation magnitude (B) for a temperature soak of $\Delta T = -0.25^\circ\text{C}$

Figure 6.6B shows the total deformation of the entire telescope structure. The asymmetry of the deformation is a result of the choice of datum, or which point of the telescope is fixed – hence why the mean value of deformation was subtracted when calculating the mirror sag profile.

6.5 On Orbit Temperature Profiles

Here, temperature profiles from the transient orbital simulations in Section 5.4 are considered. The temperature distributions for the entire telescope at the points which produce the minimal and maximum temperatures were interpolated onto the structural model, and the Δs profile was calculated. Results for the 7° inclination cold case orbit are shown in Figure 6.7, while results for the other three orbital cases can be found in Appendix A3.

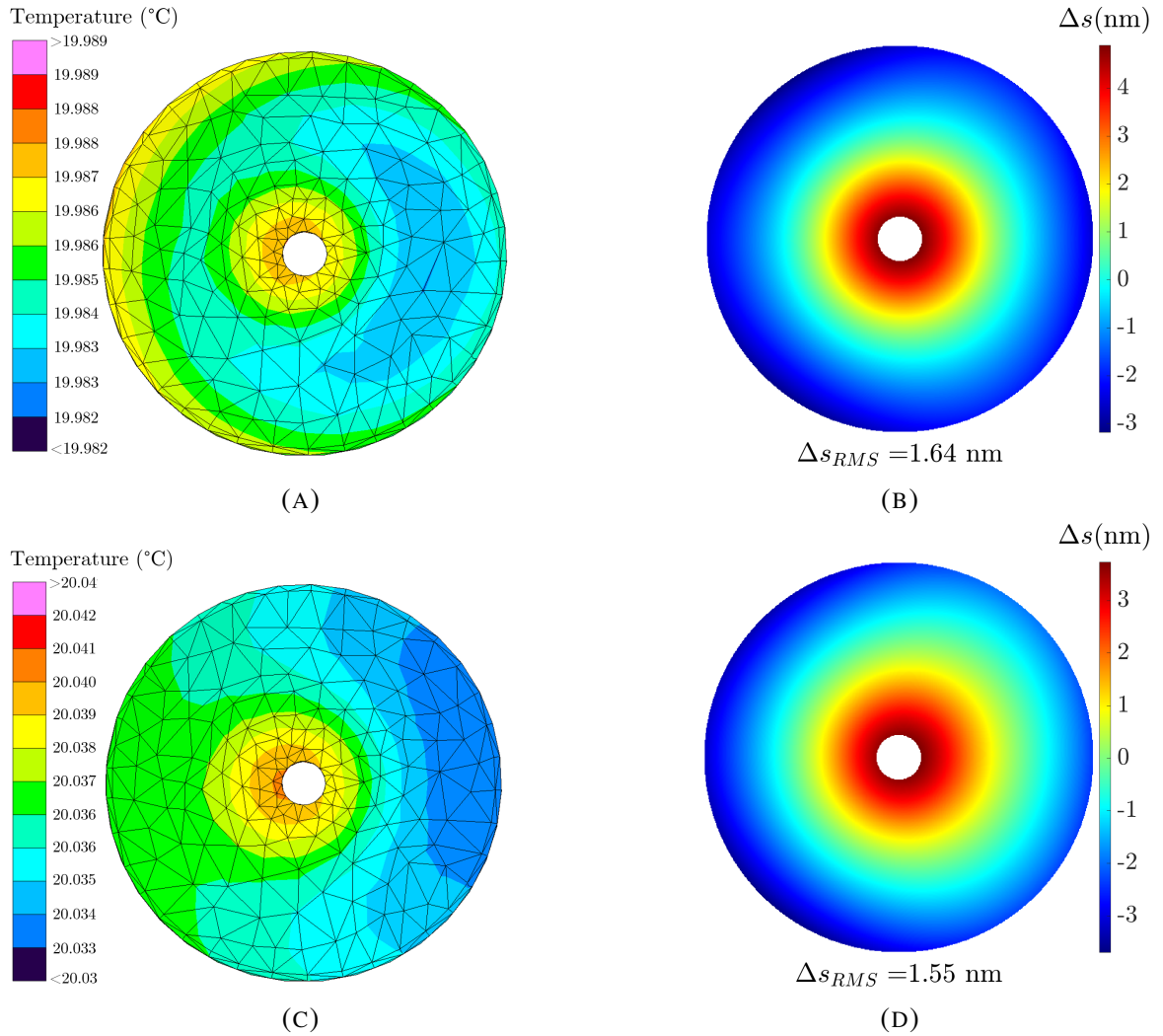


FIGURE 6.7. Temperature distributions for 7° inclination cold case orbit at times corresponding to 812 min (A), and 867 min (C) in simulation results from Figure 5.8 and their respective Δs profiles in (B) and (D).

In all orbital cases, the temperature profile shows a similar asymmetry where the right side of the mirror is colder than the left side, which arises due to heat conducting through the spacecraft from the solar panels, heating the telescope more on one side (See Figure 4.9).

The temperature gradients that result from this are tiny however, less than 0.05°C , which is in part due to the very high thermal conductivity of the SiC telescope mirror and structure. This demonstrates that thermal stability of an optical system is not only a function of the CTE, α , but also the thermal conductivity k . Glass materials with extremely low thermal expansion such as Zerodur have thermal conductivities many times lower than SiC, and will allow the formation of larger thermal gradients. The corresponding distortions may be enough to offset the benefits that come with an extremely low CTE material, especially since thermal gradients are generally more detrimental to optical performance than uniform changes in temperature. Optical performance will also be affected by temperature dependant properties other than CTE, such as the refractive index, which further increases the importance of temperature stability over purely dimensional stability considerations.

The relative uniformity of temperature distributions on the primary mirror mean that Δs profiles are similar to those found in the uniform temperature soak analysis. Differences exist however not only due to mirror temperature profiles, but also the temperature profiles of the telescope – which are not as uniform. Thermal gradients in the telescope structure cause more complicated deformation and stress patterns. The comparison between different orbits yields interesting results, with the SSO cold case orbit (Figure A.11) yielding the greatest dimensional stabilities, while the SSO hot case (Figure A.12) giving the least dimensional stability in terms of Δs_{RMS} . A similar relationship between hot and cold cases can be seen for the 7° inclination orbits. In general, hot cases result in the worst thermal conditions for optical performance due to greater heating loads from the Earth causing larger thermal gradients. In particular, the SSO orbit results in much greater temperature swings since the Earth transitions from being completely out of view for roughly one half of the orbit, to completely in view on the other side of the orbit.

Finally, of note is the difference in Δs_{RMS} between the two orbital positions considered for each case. For the 7° inclination cold case orbit, the difference between hottest and coldest primary mirror results is only 0.09 nm, but is 2.1 nm for the hot case. Similarly, for the SSO orbit, we find a difference of 1.25 nm for the cold case, and 1.59 nm for the hot case. The lower inclination orbit, for certain arrangements, will provide high stabilities at certain parts of the orbit, while in others, produce much worse results. At least for the cases studied here, the SSO orbit provides more consistent stability, largely be due to the smaller change in orbital beta angle throughout the year.

Conclusions and Future Work

To conclude, a cold bias thermal control approach was developed for the TOLIMAN space telescope. An in depth thermal analysis for two candidate orbits – a 7° inclination and an SSO orbit – was performed, which began with a detailed study of these orbits and their possible variations throughout the year. Parametric thermal models allowed optimisation routines to be used in the determination of worst case orbit design points, optimal heater placement, and in the analysis of uncertain design variables. A reduced order state space model for the telescope was derived from basic thermodynamic principles and fit to the detailed thermal model in Thermal Desktop. This provided a highly simplified, but accurate thermal model of the telescope which can be used in the design of a feedback controller. The utility of this reduced order model was established through the implementation of a PI controller for the heater system. This control system was shown to achieve temperature stability of $20 \pm 0.05^\circ\text{C}$ on the primary mirror $20 \pm 0.25^\circ\text{C}$ on the metering structure in simulation for all orbital cases considered. Between the two candidate orbits, it was found that while a higher inclination gave more viewing times of the science target, it also resulted in greater variations in orbital heating from the Earth, giving larger temperature swings with each orbit. The resulting structural stability of the proposed thermal control approach was studied, and a methodology was outlined which enables these results to be integrated with optical analysis. The achieved temperature stability was found to give RMS sag deviations of no more than 5 nm on the primary mirror in the orbital worst cases studied.

This work also outlines and provides a more complete resource in the main theory, methodologies, and technologies used in the thermal analysis for space telescope missions, as well as the integration of the results with mechanical and optical analysis. All the main concepts required to perform a complete thermal analysis were introduced in depth, including the study of orbits and their influence on thermal performance, worst case determination, the modelling of thermal heat transfer and contact conductances, and the effect of uncertainties. Considerations for using

thermal control technologies such as MLI and surface coatings, as well as their integration with thermal models was discussed in depth. Emphasis was made on incorporating thermal control in the earliest parts of the design stage, even when many components of the system are uncertain, ultimately leading to a more optimal design, especially for missions with high thermal stability requirements.

The thermal design of the TOLIMAN telescope is ongoing, and will change as the rest of the mission matures. The fabrication process for the diffractive pupil is still being developed, and an alternative approach places it on a low CTE glass plate at the front of the telescope. Whilst this approach is favourable from a fabrication point of view, the use of a low thermal conductivity element at the front of the telescope presents more challenges for the thermal and dimensional stability. Further research and analysis is required to determine if such an approach can meet the mission requirements. As the effect of orbital geometry on optical performance and pointing stability is understood, detailed trade studies can be performed to determine the optimal orbit for mission success. As more information on the mission design becomes available, thermal models for the spacecraft and telescope can be updated, and further analysis in the form of degradation studies should be performed. Thermal vacuum testing of components must be used to verify the accuracy of the thermal models introduced here, and adjustments to them must be made accordingly. The performance of the feedback controllers and passive design used for temperature stability must also be tested.

Bibliography

- [1] Centrella Kouveliotou et al. ‘Enduring quests-daring visions (NASA astrophysics in the next three decades)’. In: *arXiv preprint arXiv:1401.3741* (2014).
- [2] Engineering National Academies of Sciences and Medicine. *Pathways to Discovery in Astronomy and Astrophysics for the 2020s*. Washington, DC: The National Academies Press, 2021. ISBN: 978-0-309-46586-1. DOI: [10 . 17226 / 26141](https://doi.org/10.17226/26141). URL: [https :
// www . nap . edu / catalog / 26141 / pathways - to - discovery - in -
astronomy-and-astrophysics-for-the-2020s](https://www.nap.edu/catalog/26141/pathways-to-discovery-in-astronomy-and-astrophysics-for-the-2020s).
- [3] Peter Tuthill et al. ‘The TOLIMAN space telescope’. In: *Optical and Infrared Interferometry and Imaging VI*. Vol. 10701. International Society for Optics and Photonics. 2018, 107011J.
- [4] Olivier Guyon et al. ‘High-precision astrometry with a diffractive pupil telescope’. In: *The Astrophysical Journal Supplement Series* 200.2 (2012), p. 11.
- [5] Eduardo Bendek et al. ‘Precision astrometry mission for exoplanet detection around binary stars’. In: *Space Telescopes and Instrumentation 2018: Optical, Infrared, and Millimeter Wave*. Vol. 10698. International Society for Optics and Photonics. 2018, 106980G.
- [6] Evgenya L. Shkolnik. ‘On the verge of an astronomy CubeSat revolution’. In: *Nature Astronomy* 2.5 (May 2018), pp. 374–378. ISSN: 2397-3366.
- [7] Russell Genet et al. ‘CubeSat Astronomical Telescopes and Research in the 2020s’. In: *Bulletin of the American Astronomical Society*. Vol. 51. Sept. 2019, p. 51.
- [8] Mary Knapp and Sara Seager. ‘ASTERIA: A CubeSat Enabling High Precision Photometry in a Small Package’. In: *cosp* 42 (2018), E4–1.
- [9] David R Ardila et al. ‘The Star-Planet Activity Research CubeSat (SPARCS): A Mission to Understand the Impact of Stars in Exoplanets’. In: *arXiv preprint arXiv:1808.09954* (2018).
- [10] Rémi Duquette. ‘MOST, Micro-variability of Oscillations of STars, microsatellite structural analysis and design’. PhD thesis. 2000.

- [11] OHB Sweden. URL: <https://www.ohb-sweden.se/>.
- [12] Saber Astronautics. URL: <https://www.saberastro.com/>.
- [13] Aperture Optical Sciences (AOS). URL: <https://apertureos.com/>.
- [14] *Exoplanet and Candidate Statistics*. URL: https://exoplanetarchive.ipac.caltech.edu/docs/counts_detail.html.
- [15] M. Smith et al. ‘On-Orbit Results and Lessons Learned from the ASTERIA Space Telescope Mission’. In: 2018.
- [16] Lorraine Fesq et al. ‘Extended Mission Technology Demonstrations Using the ASTERIA Spacecraft’. In: *2019 IEEE Aerospace Conference*. 2019, pp. 1–11.
- [17] Christopher Pong. ‘On-orbit performance & operation of the attitude & pointing control subsystems on ASTERIA’. In: (2018).
- [18] David G. Koch et al. ‘KEPLER MISSION DESIGN, REALIZED PHOTOMETRIC PERFORMANCE, AND EARLY SCIENCE’. In: *The Astrophysical Journal* 713.2 (Mar. 2010), pp. L79–L86.
- [19] RM Cutri et al. ‘Explanatory supplement to the WISE all-sky data release products’. In: *Explanatory Supplement to the WISE All-Sky Data Release Products* (2012), p. 1.
- [20] George R Ricker et al. ‘Transiting exoplanet survey satellite’. In: *Journal of Astronomical Telescopes, Instruments, and Systems* 1.1 (2014), p. 014003.
- [21] Andrea Fortier et al. ‘CHEOPS: a space telescope for ultra-high precision photometry of exoplanet transits’. In: *Space Telescopes and Instrumentation 2014: Optical, Infrared, and Millimeter Wave*. Vol. 9143. International Society for Optics and Photonics. 2014, 91432J.
- [22] Evgenya Shkolnik. ‘Monitoring the High-Energy Radiation Environment of Exoplanets around Lowmass Stars with SPARCS (Star-Planet Activity Research CubeSat)’. In: *NASA APRA Proposal* (2016), 16–APRA16.
- [23] Paul A Scowen et al. ‘SPARCS payload assembly, integration, and test update’. In: *Space Telescopes and Instrumentation 2020: Ultraviolet to Gamma Ray*. Vol. 11444. International Society for Optics and Photonics. 2020, 114440A.
- [24] Naoteru Gouda. ‘Jasmine’. In: *Scholarpedia* 6.10 (2011), p. 12021.
- [25] Naoteru Gouda. ‘Infrared Space Astrometry Missions~ JASMINE Missions~’. In: *Galactic Archaeology: Near-Field Cosmology and the Formation of the Milky Way*. Vol. 458. 2012, p. 417.

- [26] Shin Utsunomiya et al. 'Structure design of the telescope for Small-JASMINE program'. In: *Space Telescopes and Instrumentation 2014: Optical, Infrared, and Millimeter Wave*. Vol. 9143. International Society for Optics and Photonics. 2014, 91430Z.
- [27] Kotaro ONA et al. 'The Advanced Super Invar Alloys with Zero Thermal Expansion for Space Telescopes'. In: *TRANSACTIONS OF THE JAPAN SOCIETY FOR AERONAUTICAL AND SPACE SCIENCES, AEROSPACE TECHNOLOGY JAPAN* 18.3 (2020), pp. 32–37.
- [28] Slavek Rucinski et al. 'Most (microvariability & oscillations of stars) Canadian astronomical micro-satellite'. In: *Advances in Space Research* 31.2 (2003), pp. 371–373.
- [29] Alexander Schwarzenberg-Czerny et al. 'The BRITE Nanosatellite Constellation Mission'. In: *38th COSPAR Scientific Assembly*. Vol. 38. Jan. 2010, p. 15.
- [30] Yunus engel. *Thermodynamics : an engineering approach*. New York: McGraw-Hill Education, 2015. ISBN: 978-0-07-339817-4.
- [31] Frank Incropera. *Fundamentals of heat and mass transfer*. Hoboken, NJ: John Wiley, 2007. ISBN: 0-471-45728-0.
- [32] Milan Yovanovich. 'Micro and macro hardness measurements, correlations, and contact models'. In: *44th AIAA aerospace sciences meeting and exhibit*. 2006, p. 979.
- [33] David Gilmore. *Spacecraft thermal control handbook*. El Segundo, Calif: Aerospace Press, 2002. ISBN: 1-884989-11-X.
- [34] SR Mirmira, EE Marotta and LS Fletcher. 'Thermal contact conductance of adhesives for microelectronic systems'. In: *Journal of thermophysics and heat transfer* 11.2 (1997), pp. 141–145.
- [35] John Howell. *Thermal radiation heat transfer*. Boca Raton, FL: CRC Press, 2021. ISBN: 9780367347079.
- [36] Lee D Feinberg et al. 'Space telescope design considerations'. In: *Optical Engineering* 51.1 (2012), p. 011006.
- [37] Michael W McElwain et al. 'Status of the James Webb Space Telescope mission'. In: *Space Telescopes and Instrumentation 2020: Optical, Infrared, and Millimeter Wave*. Vol. 11443. International Society for Optics and Photonics. 2020, 114430T.
- [38] José Meseguer. *Spacecraft thermal control*. Cambridge Philadelphia: Woodhead Pub, 2012. ISBN: 978-1-84569-996-3.

- [39] Sumanth RM. ‘Computation of Eclipse Time for Low-Earth Orbiting Small Satellites’. In: *International Journal of Aviation, Aeronautics, and Aerospace* 6.5 (2019), p. 15.
- [40] MN Ismail et al. ‘Eclipse intervals for satellites in circular orbit under the effects of Earth’s oblateness and solar radiation pressure’. In: *NRIAG Journal of Astronomy and Geophysics* 4.1 (2015), pp. 117–122.
- [41] Bely. *The design and construction of large optical telescopes*. New York: Springer, 2003. ISBN: 0-387-95512-7.
- [42] Michael W Werner et al. ‘The Spitzer space telescope mission’. In: *The Astrophysical Journal Supplement Series* 154.1 (2004), p. 1.
- [43] Jianyin Miao. *Spacecraft thermal control technologies*. Singapore: Springer, 2021. ISBN: 978-981-15-4983-0.
- [44] Robert Karam. *Satellite thermal control for systems engineers*. Reston, Va: American Institute of Aeronautics and Astronautics, 1998. ISBN: 1-56347-276-7.
- [45] MM Finckenor and D Dooling. *Multilayer insulation material guidelines*. Nasa, 1999.
- [46] Lonny Kauder. ‘Spacecraft thermal control coatings references’. In: *NASA, Goddard space flight center* (2005).
- [47] Ann F Whitaker and John Gregory. *LDEF materials results for spacecraft applications*. Vol. 3257. National Aeronautics and Space Administration, Office of Management . . ., 1994.
- [48] A Anvari, Foad Farhani and KS Niaki. ‘Comparative study on space qualified paints used for thermal control of a small satellite’. In: (2009).
- [49] Eric Sunada and Jose Rodriguez. ‘JPL advanced thermal control technology roadmap’. In: (2017).
- [50] Timothy D Panczak and Brent A Cullimore. *Parametric Thermal Analysis and Optimization Using Thermal Desktop™*. Tech. rep. SAE Technical Paper, 2000.
- [51] D Martin. *Parametric Models and Optimization for Rapid Thermal Design*. Tech. rep. SAE Technical Paper, 2004.
- [52] B Cullimore. ‘Automated determination of worst-case design scenarios’. In: *SAE transactions* (2003), pp. 215–222.
- [53] Katelyn Elizabeth Boushon. *Thermal analysis and control of small satellites in low Earth orbit*. Missouri University of Science and Technology, 2018.
- [54] Rodrigo Avila de Luis. ‘Standardized Thermal Control Solutions for PocketQubes’. In: (2019).

- [55] Kevin Dale Stout. 'Design optimization of thermal paths in spacecraft systems'. PhD thesis. Massachusetts Institute of Technology, 2013.
- [56] J Lai. *Thermal Analysis for a High Precision Pointing Helios CubeSat*. 2019.
- [57] Emanuel Escobar, Marcos Diaz and Juan Cristóbal Zagal. 'Evolutionary design of a satellite thermal control system: Real experiments for a CubeSat mission'. In: *Applied Thermal Engineering* 105 (2016), pp. 490–500.
- [58] Isidoro Martinez. 'Heat transfer and thermal radiation modelling'. In: (2012). URL: <http://imartinez.etsiae.upm.es/~isidoro/tc3/Heat%5C%20transfer%5C%20and%5C%20thermal%5C%20radiation%5C%20modelling.pdf>.
- [59] Borivoje Budimira Mikic, M Michael Yovanovich and Warren M Rohsenow. *The effect of surface roughness and waviness upon the overall thermal contact resistance*. Tech. rep. Cambridge, Mass.: MIT Dept. of Mechanical Engineering,[1966], 1966.
- [60] James A Greenwood and JB PI Williamson. 'Contact of nominally flat surfaces'. In: *Proceedings of the royal society of London. Series A. Mathematical and physical sciences* 295.1442 (1966), pp. 300–319.
- [61] BB Mikić. 'Thermal contact conductance; theoretical considerations'. In: *International Journal of Heat and Mass Transfer* 17.2 (1974), pp. 205–214.
- [62] MR Sridhar and MM Yovanovich. 'Review of elastic and plastic contact conductance models-comparison with experiment'. In: *Journal of Thermophysics and Heat Transfer* 8.4 (1994), pp. 633–640.
- [63] MM Yovanovich. 'New contact and gap conductance correlations for conforming rough surfaces'. In: *AIAA Paper* 81-1164 (1981), pp. 1–6.
- [64] S Song and M Yovanovich. 'Explicit relative contact pressure expression-Dependence upon surface roughness parameters and Vickers microhardness coefficients'. In: *25th AIAA Aerospace Sciences Meeting*. 1987, p. 152.
- [65] Adel Abdel-Halim Hegazy. 'Thermal joint conductance of conforming rough surfaces: Effect of surface micro-hardness variation'. In: (2016).
- [66] Peter Teertstra. 'Thermal conductivity and contact resistance measurements for adhesives'. In: *International Electronic Packaging Technical Conference and Exhibition*. Vol. 42770. 2007, pp. 381–388.
- [67] Robert C Campbell, Stephen E Smith and Raymond L Dietz. 'Measurements of adhesive bondline effective thermal conductivity and thermal resistance using the laser

- flash method'. In: *Fifteenth Annual IEEE Semiconductor Thermal Measurement and Management Symposium (Cat. No. 99CH36306)*. IEEE. 1999, pp. 83–97.
- [68] CG Justus et al. 'Simple thermal environment Model (STEM) user's guide'. In: (2001).
- [69] BJ Anderson, CG Justus and GW Batts. 'Guidelines for the selection of near-earth thermal environment parameters for spacecraft design'. In: *NASA Technical Memorandum 211221* (2001).
- [70] Romain Peyrou-Lauga. 'Using real Earth albedo and Earth IR flux for spacecraft thermal analysis'. In: 47th International Conference on Environmental Systems. 2017.
- [71] *BLACKFLY S BOARD-LEVEL*. BFS-U3-200S6-BD2. FLIR. 2019. URL: <https://www.eureca.de/files/pdf/optoelectronics/flir/BFS-U3-200S6-BD2-Datasheet.pdf>.
- [72] *7 Series FPGAs Data Sheet: Overview*. DS180. XILINIX. 2020. URL: https://www.xilinx.com/support/documentation/data_sheets/ds180_7Series_Overview.pdf.
- [73] *Mobile Low-Power DDR SDRAM*. MT46H64M16LF & MT46H32M32LF. rev. B. Micro. 2009. URL: https://media-www.micron.com/-/media/client/global/documents/products/data-sheet/dram/mobile-dram/low-power-dram/lpddr/60-series/t68m_1gb_mobile_lpddr_sdram.pdf?rev=cf7059175b164edfaa6169e079fd7476.
- [74] *Monochrome/Color CMOS Image Sensor*. IMX183CLK-J/CQJ-J. SONY. 2013. URL: https://www.sony-semicon.co.jp/products/common/pdf/IMX183CLK-J_CQJ-J_Flyer.pdf.
- [75] Vinod Nirale. 'Evaluation of effective thermal conductivity in PCB'. In: *IJRST* 3 (2016), pp. 174–179.
- [76] Rafael Lopes Costa and Valeri Vlassov. 'Simulation method for effective thermal conductivity determination of complex boards'. In: *2o Workshop em Engenharia e Tecnologia Espaciais, 03, 04 e 05 de Maio de*. 2011.
- [77] *TD Suite Manual*. Version 6.2, Patch 2. C&R Technologies. 2020.
- [78] *Thermal Solutions Design Guide*. MINCO. 2016. URL: <https://www.minco.com/wp-content/uploads/Minco-Thermal-Solutions-Design-Guide.pdf>.
- [79] Karl Johan Åström and Richard M Murray. *Feedback systems*. Princeton university press, 2010.

- [80] Lennart Andersson et al. ‘A manual for system identification’. In: *Laboratory Exercises in System Identification*. KF Sigma i Lund AB. Department of Automatic Control, Lund Institute of Technology, Box 118 (1998).
- [81] Juan Garrido, Francisco Vázquez and Fernando Morilla. ‘Multivariable PID control by decoupling’. In: *International Journal of Systems Science* 47.5 (2016), pp. 1054–1072.
- [82] Karl Johan Astrom, Karl Henrik Johansson and Qing-Guo Wang. ‘Design of decoupled PID controllers for MIMO systems’. In: *Proceedings of the 2001 American Control Conference*.(Cat. No. 01CH37148). Vol. 3. IEEE. 2001, pp. 2015–2020.
- [83] Dan Chen and Dale E Seborg. ‘Multiloop PI/PID controller design based on Ger-shgorin bands’. In: *Proceedings of the 2001 American Control Conference*.(Cat. No. 01CH37148). Vol. 5. IEEE. 2001, pp. 4122–4127.
- [84] H Philip Stahl et al. ‘Predictive Thermal Control (PTC) technology to enable thermally stable telescopes: year four status’. In: *Astronomical Optics: Design, Manufacture, and Test of Space and Ground Systems III*. Vol. 11820. International Society for Optics and Photonics. 2021, 118200A.
- [85] G Michels, V Genberg and K Doyle. ‘Integrating Ansys mechanical analysis with optical performance analysis using Sigfit’. In: *ANSYS Conference & 26th CADFEM Users’ Meeting, Darmstadt, Germany*. Citeseer. 2008.
- [86] O. C. Zienkiewicz. *The finite element method : its basis and fundamentals*. Oxford Boston: Elsevier Butterworth-Heinemann, 2005. ISBN: 9780080472775.
- [87] Keith B Doyle, Victor L Genberg and Gregory J Michels. *Integrated optomechanical analysis*. Vol. 58. SPIE press, 2002.
- [88] *MatWeb Material Property Data*. URL: <http://www.matweb.com>.
- [89] Scott Gibb. ‘An Introduction to Invar’. In: *College of Optical Sciences, University of Arizona, Tucson* (2006).
- [90] *Advanced Materials*. Trex Enterprises. URL: <http://www.trexenterprises.com/Pages/Products%5C%20and%5C%20Services/Advanced%5C%20Materials/mat%5C%20spec%5C%200503.pdf>.
- [91] *SUPERSiC Materials*. SUPERSiC-Si-3C (-8C). Entegris. 2019. URL: <https://www.entegris.com/content/dam/shared-product-assets/specialty-shared/datasheet-supersic-silicon-carbide-7062.pdf>.

- [92] *Li-ion Rechargeable Battery*. ABSL 8s3p 29.6V 8.4Ah. ABSL. 2020. URL: https://www.enersys.com/48d636/globalassets/documents/product-documentation/absl/amer/am-absl-8s3pmm-fl-aa_0420.pdf.
- [93] *Command and data handling*. Sirius TCM LEON3FT. AAC Clyde Space. 2019. URL: https://www.aac-clyde.space/assets/000/000/107/SIRIUS-TCM-LEON3FT_original.pdf?1565703953.
- [94] *Power Conditioning and Monitoring Units*. STARBUCK MICRO. AAC Clyde Space. 2020. URL: https://www.aac-clyde.space/assets/000/000/186/AAC_DataSheet_Starbuck-Micro_original.pdf?1613579830.
- [95] *Microsatellite Reaction Wheels*. RW3-0.06. Rocket Lab. 2020. URL: <https://www.rocketlabusa.com/assets/Uploads/60-MNMS-product-sheet.pdf>.
- [96] *Magnetometer*. NMRM-Bn25o485. Ver. 10a. NewSpace. 2020. URL: https://www.newspacesystems.com/wp-content/uploads/2020/10/NewSpace-Magnetometer_2020_10a.pdf.
- [97] *X Band Transceiver*. XLink-X. IQ spacecom. 2020. URL: https://www.iq-spacecom.com/images/downloads/XLink-X_Datasheet_042021.pdf.
- [98] *X Band Transceiver*. Compocel AL and Compocel AL FR. CEL Components. 2019. URL: <https://pdf.aeroexpo.online/pdf/cel-components-srl/compocel-al-compocel-al-fr/182895-16705.html#open27343>.
- [99] Wenjing Zhang and George Feng. ‘A Quick PCB Thermal Calculation for Power Electronic Devices with Exposed Pad Packages’. In: *PCIM Asia 2019; International Exhibition and Conference for Power Electronics, Intelligent Motion, Renewable Energy and Energy Management*. VDE. 2019, pp. 1–8.
- [100] Howard Curtis. *Orbital mechanics for engineering students*. Oxford: Butterworth-Heinemann, 2014. ISBN: 978-0-08-097747-8.
- [101] Vladimir Chobotov. *Orbital mechanics*. Reston, Va: American Institute of Aeronautics and Astronautics, 2002. ISBN: 1-56347-537-5.
- [102] David Vallado. *Fundamentals of astrodynamics and applications*. Dordrecht Boston: Kluwer Academic Publishers, 2001. ISBN: 9781881883128.

APPENDIX A

Further Results

A1 System Identification

Results for for the remaining orbital cases (7° hot case, and SSO cold and hot cases) not shown in Section 5.3.2 are given here.



FIGURE A.1. System identification results for 7° inclination Cold Case orbit. Note all orbital cases were used in the fitting of parameters. Panels (A), (C), (E), (G), and (I) show the nodal temperatures, heater inputs, and disturbance inputs for the training data. Panels (B), (D), (F), (H), and (J) show the same results for a validation case.

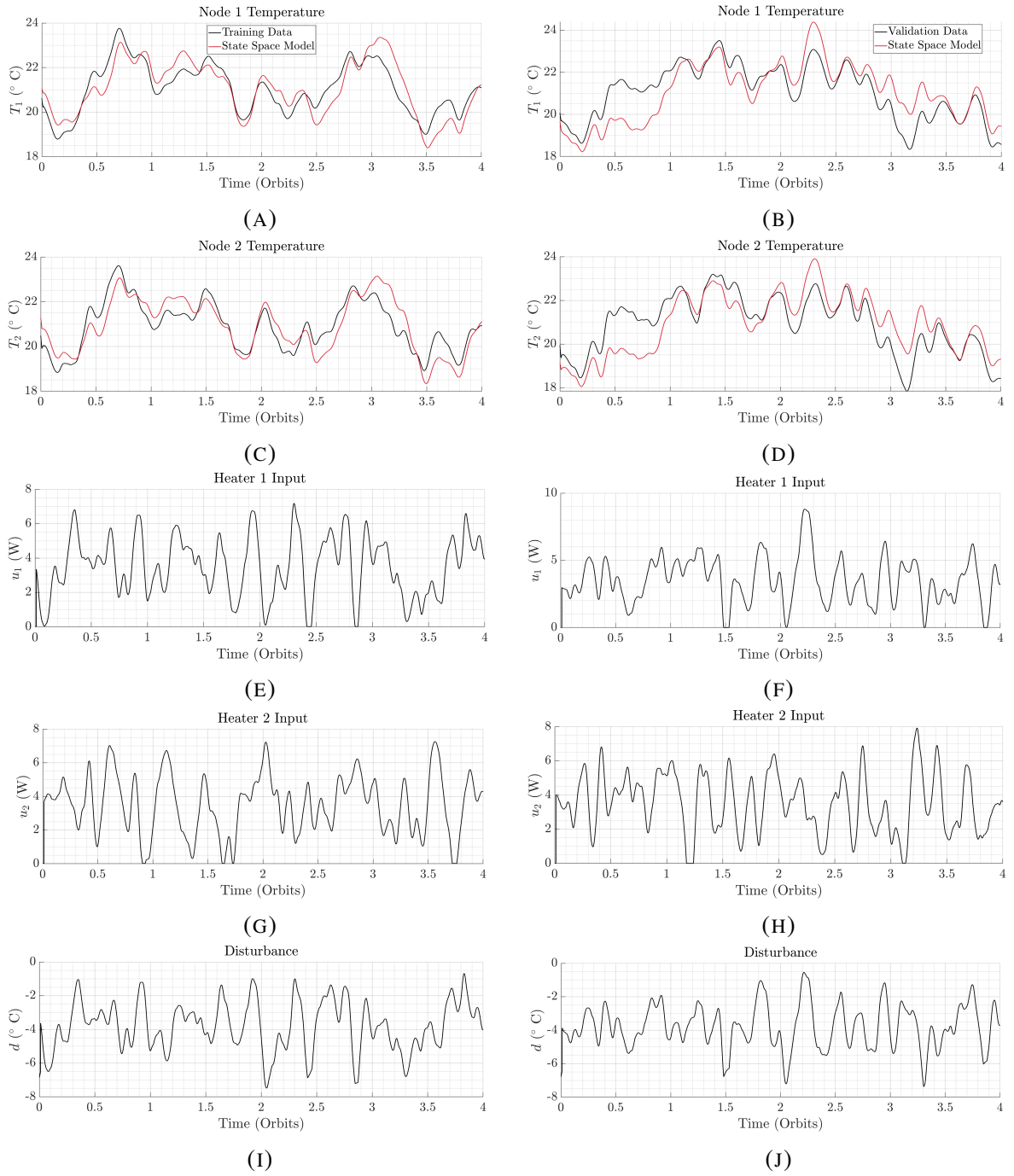


FIGURE A.2. System identification results for 7° inclination Cold Case orbit. Note all orbital cases were used in the fitting of parameters. Panels (A), (C), (E), (G), and (I) show the nodal temperatures, heater inputs, and disturbance inputs for the training data. Panels (B), (D), (F), (H), and (J) show the same results for a validation case.



FIGURE A.3. System identification results for 7° inclination Cold Case orbit. Note all orbital cases were used in the fitting of parameters. Panels (A), (C), (E), (G), and (I) show the nodal temperatures, heater inputs, and disturbance inputs for the training data. Panels (B), (D), (F), (H), and (J) show the same results for a validation case.

A2 PI Control Simulation

Results for for the remaining orbital cases (7° hot case, and SSO cold and hot cases) not shown in Section 5.4 are given here.

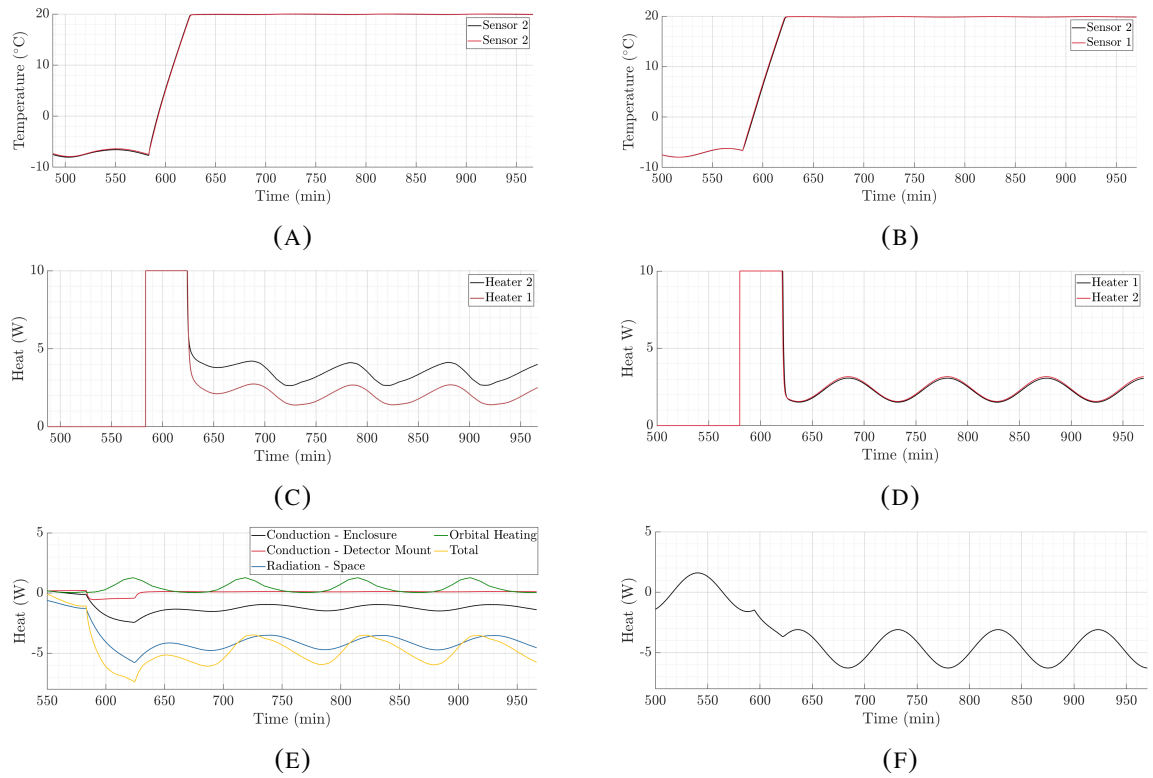


FIGURE A.4. PI controller simulation for 7° hot case orbital condition. Panels (A), (C), and (E) show the sensor temperatures, heater input, and heat transfers for the Thermal Desktop model. Panels (B), (D), and (F), show the equivalent simulation using the reduced order state space models, with (F) being the disturbance input.

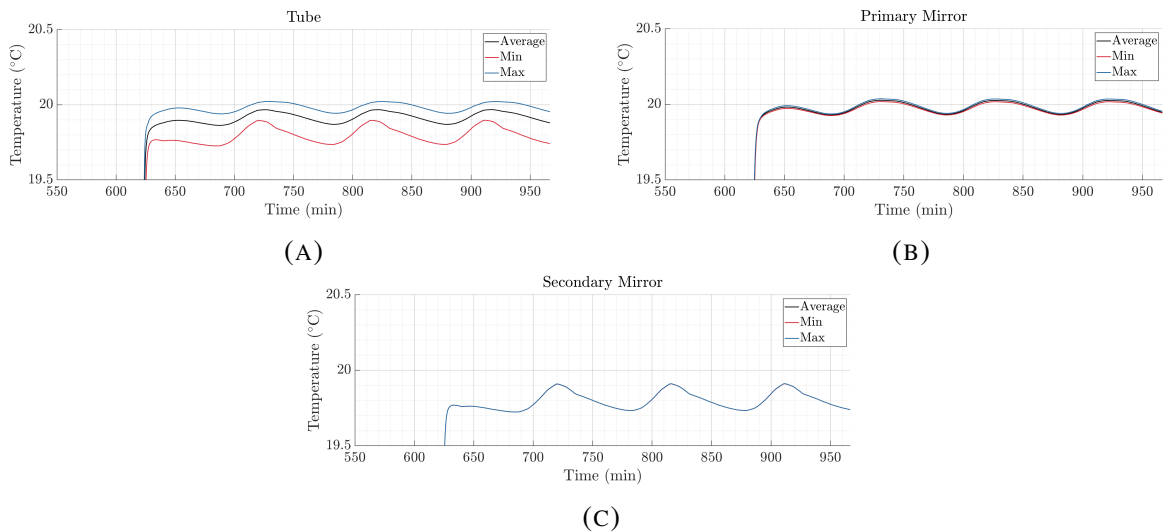


FIGURE A.5. Telescope tube (A), primary mirror (B), and secondary mirror (C) temperatures for the 7° inclination hot case orbit from Thermal Desktop model using PI controller.

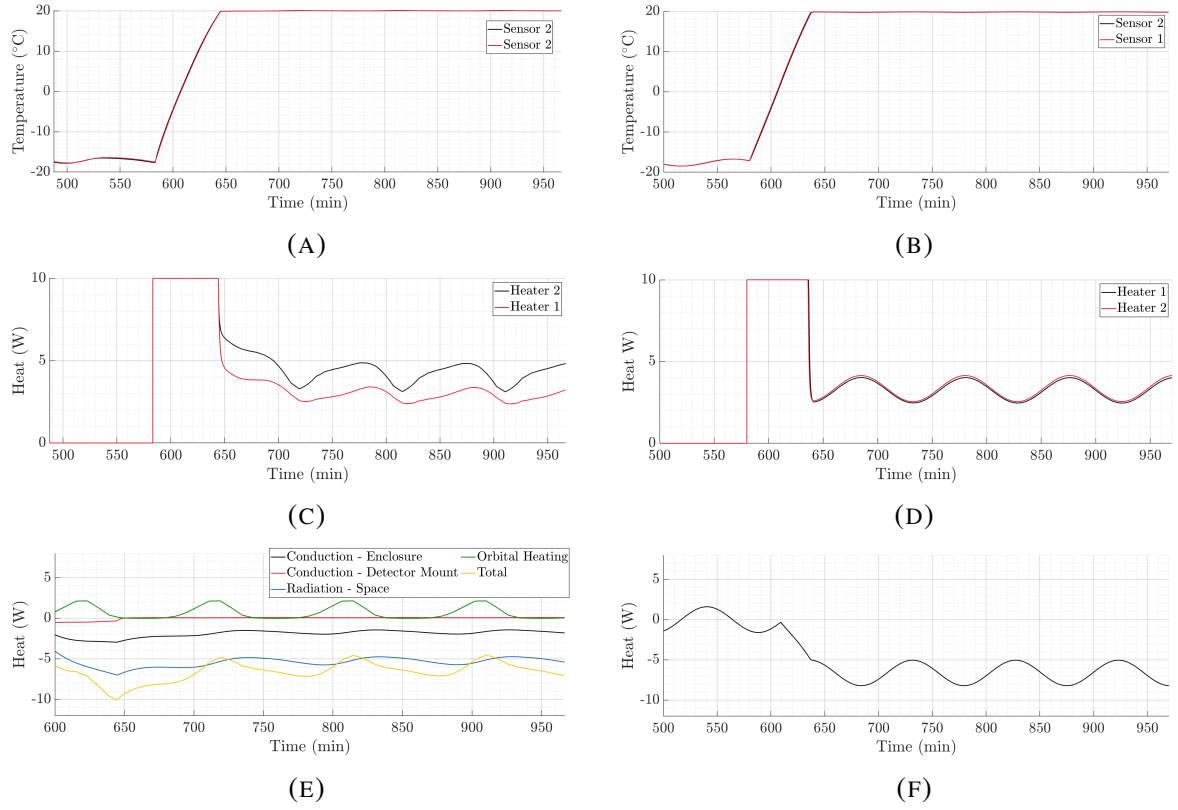


FIGURE A.6. PI controller simulation for SSO cold case orbital condition. Panels (A), (C), and (E) show the sensor temperatures, heater input, and heat transfers for the Thermal Desktop model. Panels (B), (D), and (F), show the equivalent simulation using the reduced order state space models, with (F) being the disturbance input.

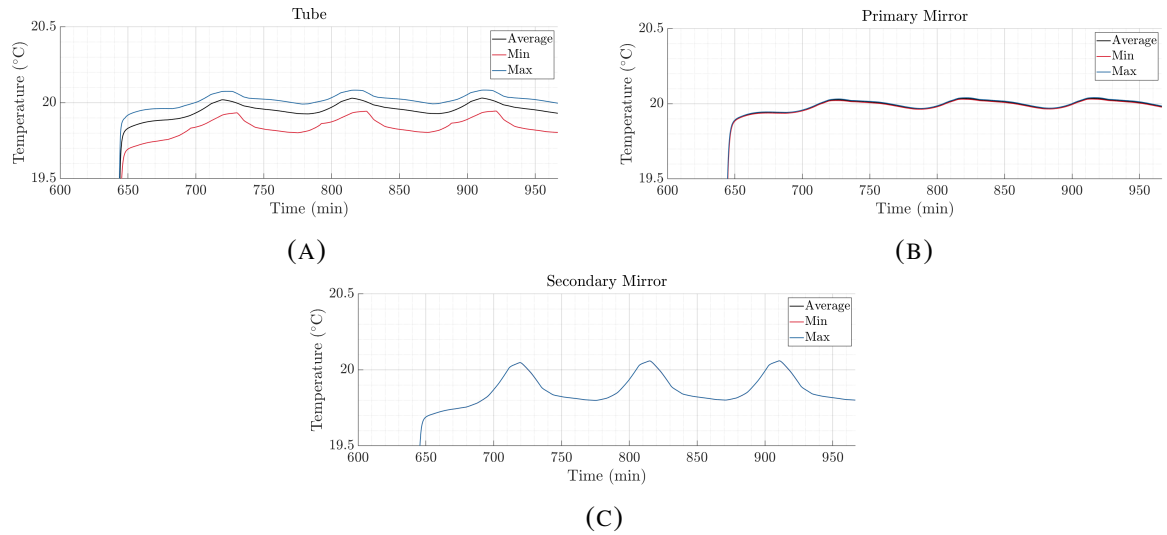


FIGURE A.7. Telescope tube (A), primary mirror (B), and secondary mirror (C) temperatures for the SSO cold case orbit from Thermal Desktop model using PI controller.

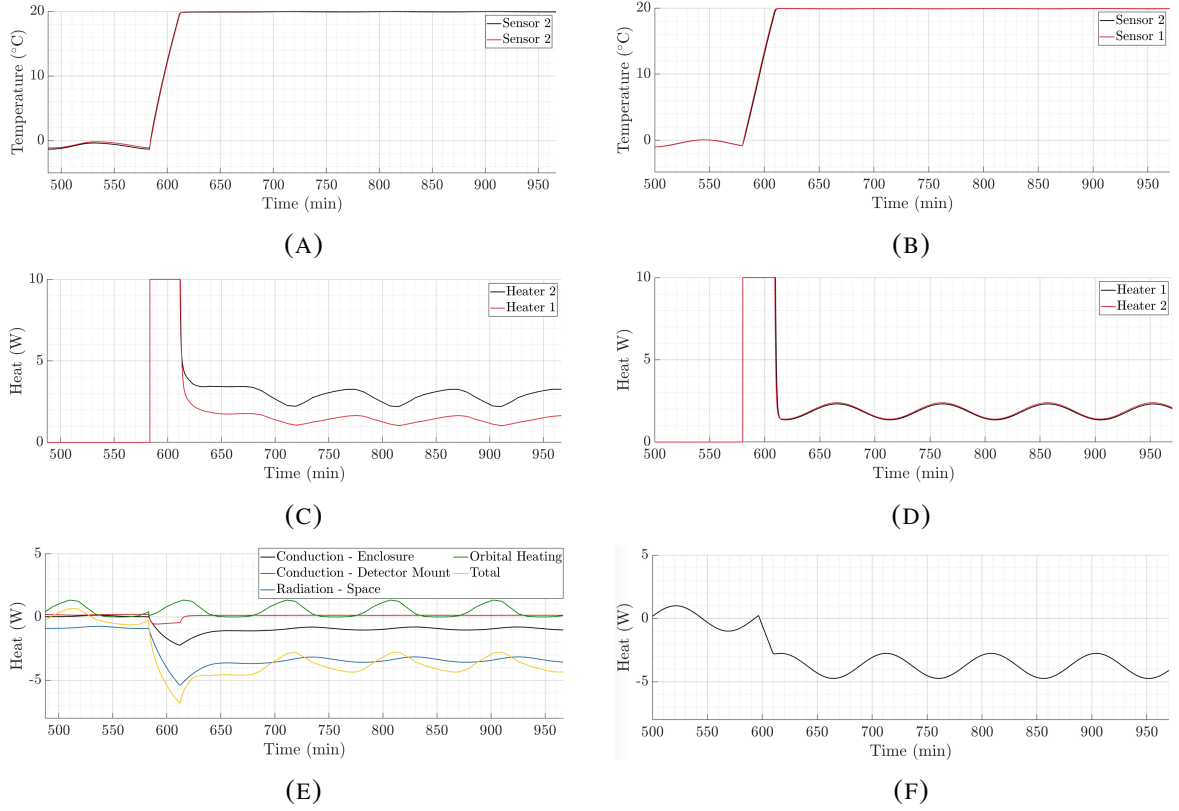


FIGURE A.8. PI controller simulation for SSO hot case orbital condition. Panels (A), (C), and (E) show the sensor temperatures, heater input, and heat transfers for the Thermal Desktop model. Panels (B), (D), and (F), show the equivalent simulation using the reduced order state space models, with (F) being the disturbance input.

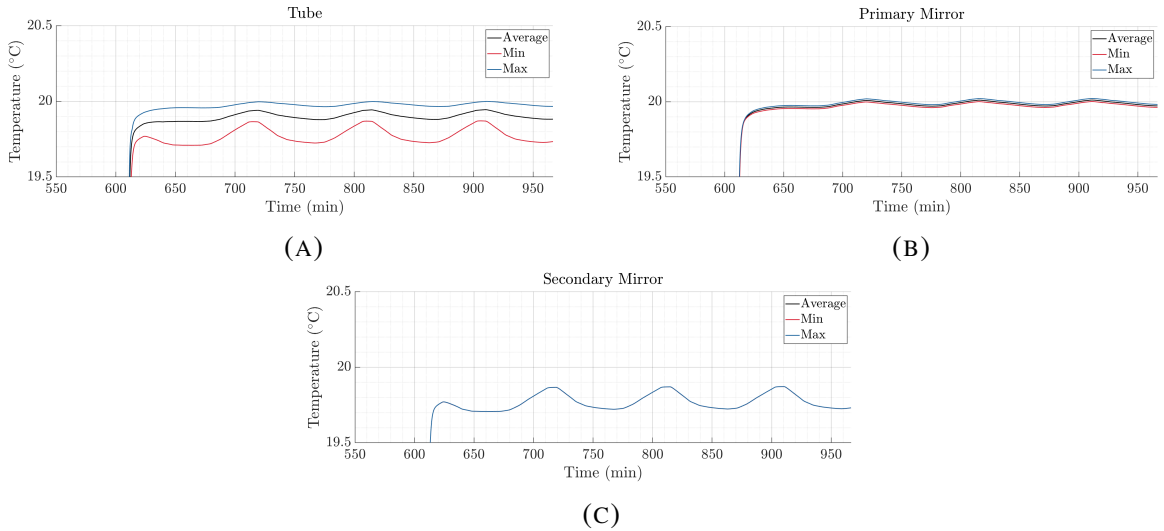


FIGURE A.9. Telescope tube (A), primary mirror (B), and secondary mirror (C) temperatures for the SSO hot case orbit from Thermal Desktop model using PI controller.

A3 On Orbit Mirror Sag Profiles

Results for the remaining orbital cases (7° hot case, and SSO cold and hot cases) not shown in Section 6.5 are given here.

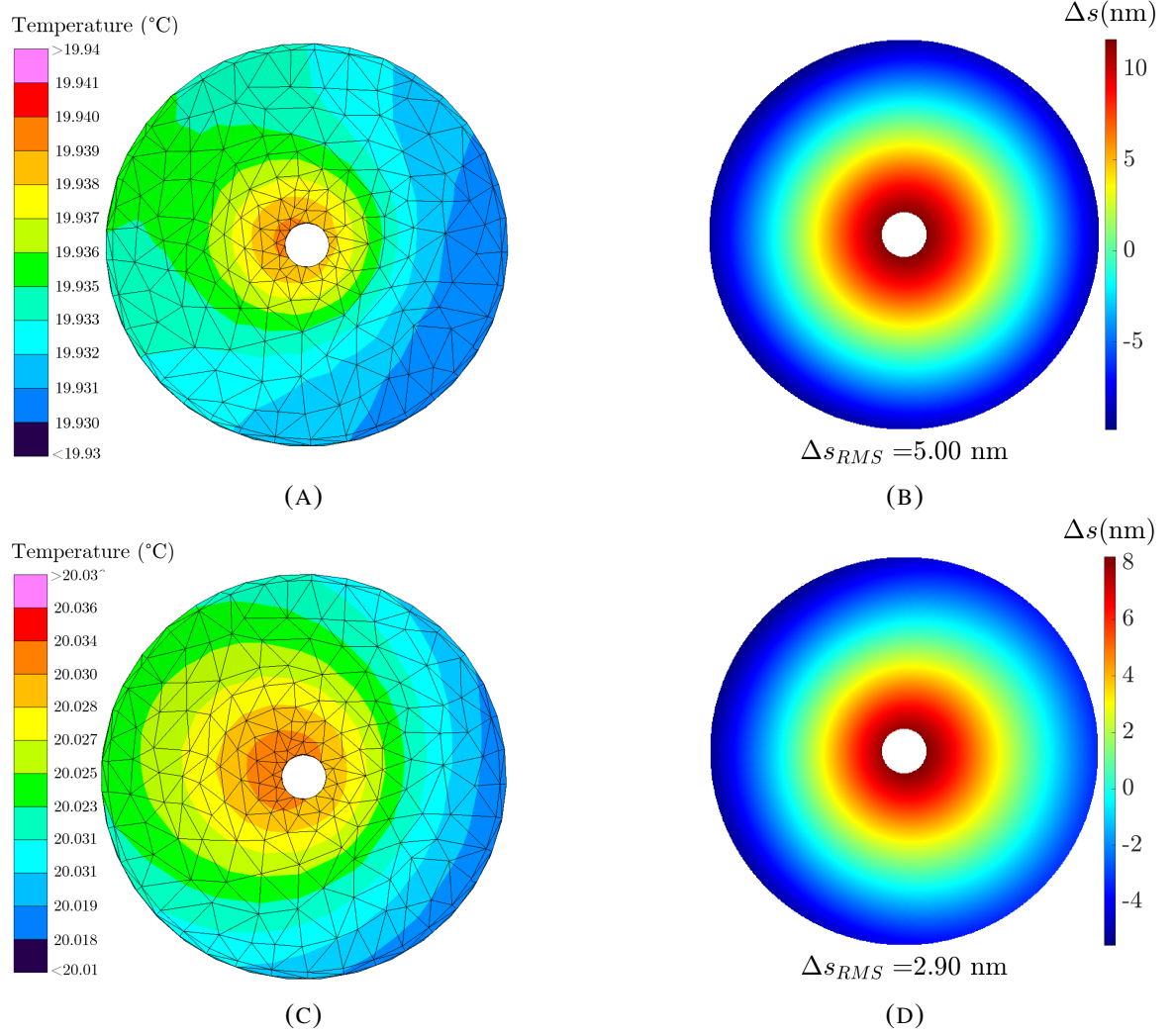


FIGURE A.10. Temperature distributions for 7° inclination hot case orbit at times corresponding to 785 min (A), and 825 min (C) in simulation results from Figure A.5 and their respective Δs profiles in (B) and (D).

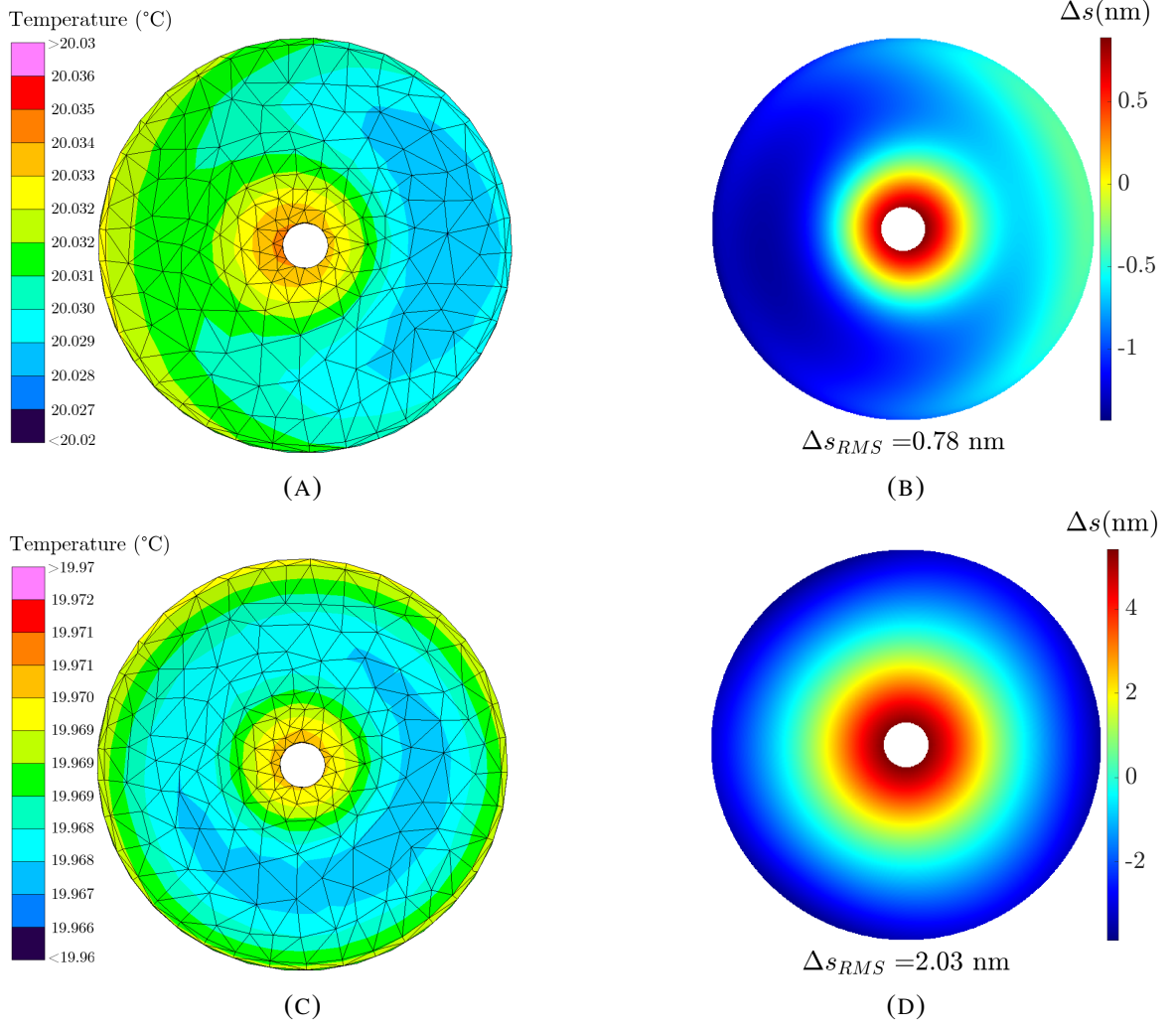


FIGURE A.11. Temperature distributions for SSO cold case orbit at times corresponding to 815 min (A), and 885 min (C) in simulation results from Figure A.7 and their respective Δs profiles in (B) and (D).

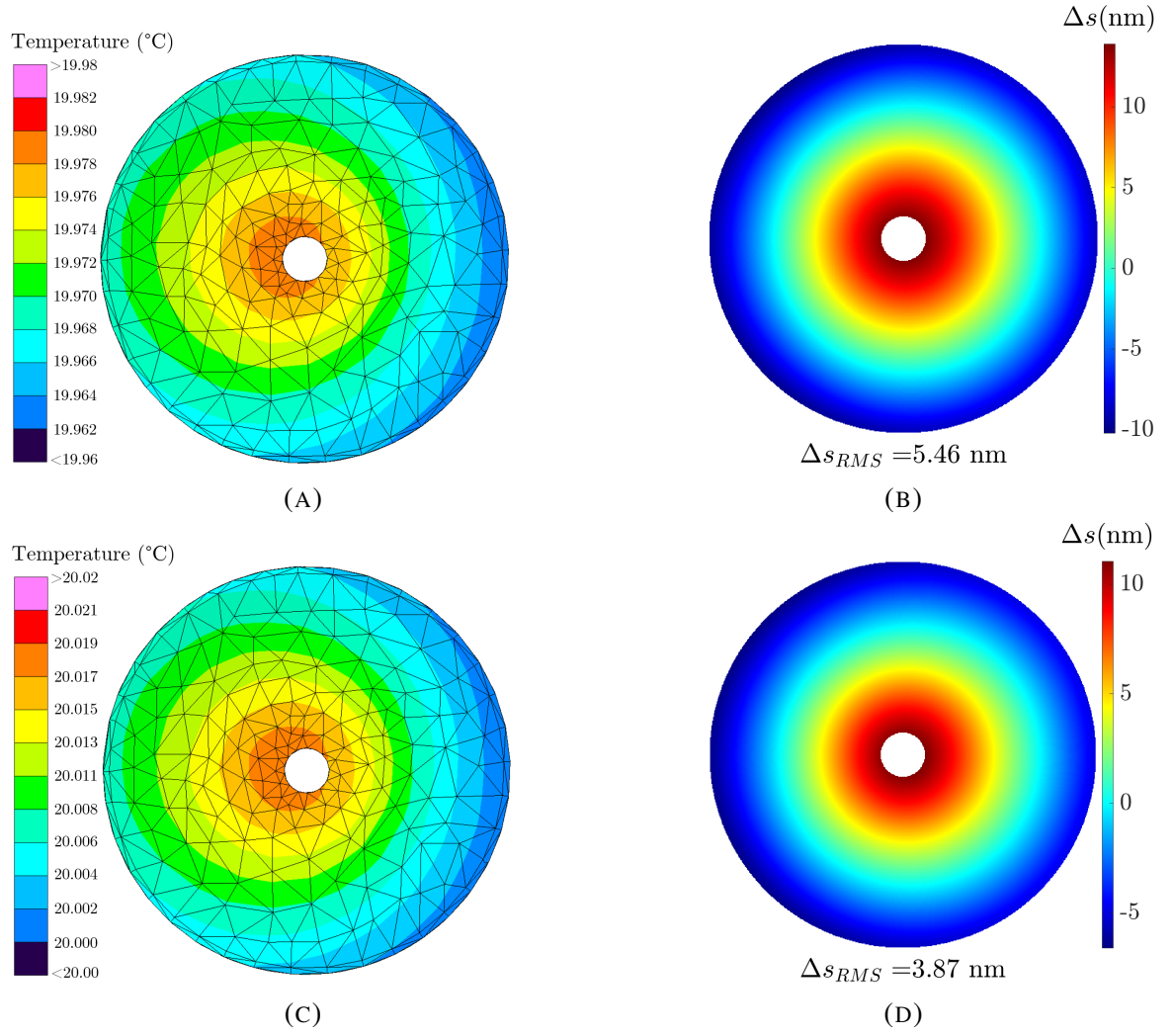


FIGURE A.12. Temperature distributions for SSO hot case orbit at times corresponding to 815 min (A), and 885 min (C) in simulation results from Figure A.9 and their respective Δs profiles in (B) and (D).

APPENDIX B

Material and Surface Optical Properties

TABLE B.1. Thermophysical properties for bulk materials used in the creation of thermal models.

Material	Density (kg/m ³)	Elastic Modulus (GPa)	Poisson Ratio	Thermal Conductivity (W/m/K)	Specific Heat Capacity (J/kg/K)	CTE ($\times 10^{-6}/\text{K}$)	Ref.
6061 Aluminium	2770	68.9	0.33	167	896	24	[88]
Brass	8530	97	0.31	110	380	-	[88]
Copper	8933	130	0.34	401	385	-	[88]
Invar 36	8050	141	0.26	10.4	515	1.0	[89]
SiC (Structure)	3200	466	0.21	220	640	3.5	[90]
SiC (Mirror)	3030	331	0.17	220	635	2.4	[91]
Silicon	2330	-	-	148	712	-	[88]
Germanium	5260	-	-	61	324	-	[88]
Epoxy	1350	-	-	1.4	300	-	[88]
Kapton	1410	-	-	0.2	1004	-	[88]

TABLE B.2. Surface optical properties for various surfaces used in thermal models.

Surface	α_S	ϵ_{IR}	Ref.
Polished Aluminium	0.15	0.05	[33]
Anodized Aluminium (Clear Anodize)	0.31	0.81	[33]
Solar Cells	0.92	0.85	[33]
5 Mil Teflon (Aluminized)	0.17	0.79	[33]
2 Mil Kapton (Aluminized)	0.45	0.73	[33]
Black Kapton	0.93	0.85	[33]
Chemglaze Z306	0.96	0.91	[33]
SiC (Mirror & Structure)	0.8	0.9	Best Estimate
Mirror Surface	0.15	0.05	[33]

APPENDIX C

Calculation of Thermal Conductances, Lumped Properties, and Effective Conductivities

C1 Spacecraft Model

C1.1 Lumped Component Properties

Table C.1 shows the lumped thermophysical optical properties for the heat generating components used in the spacecraft model. Only components which produce more than 2% of the total heat generation are included. This means components such as the magnetorquers and GPS receiver are not included as heat generating components. Their thermal mass is accounted for in the satellite structure. Since very little is known about the material and construction of the components, the heat capacity is estimated to be the same for all components to be between that of copper and aluminium. Optical properties are taken to be the same as those for anodized aluminium.

TABLE C.1. Lumped thermophysical and optical properties of electronic components used in spacecraft model.

Component	Mass (kg)	Specific Heat Capacity (J/kg/K)	α_S	ϵ_{IR}	Ref.
Battery	1.66	600	0.31	0.81	[92]
Computer	0.13	600	0.31	0.81	[93]
PDU	2.45	600	0.31	0.81	[94]
Reaction Wheels	2	600	0.31	0.81	[95]
Magnetometer	0.085	600	0.31	0.81	[96]
X-Band Transceiver	0.2	600	0.31	0.81	[97]

C1.2 Component Power Dissipation

Table C.2 shows the nominal heat generation of each of the major heat producing components, as well as the fraction of total spacecraft power which they consume. In total, the nominal heat load is estimated to be 22.3 W. The uncertainty in these values are accounted for by fixing the power fraction for each component and having the total spacecraft heat generation be a parameter in the model. It is important to note that in reality, some of these components will be switched off during various points in the mission (i.e. the power fraction is zero). This complexity is neglected in this analysis.

TABLE C.2. Nominal heat generation and component power fractions for spacecraft thermal model.

Component	Nominal Power (W)	Power Fraction	Ref.
Battery	1.5	0.07	[92]
Computer	1.3	0.06	[93]
PDU	4	0.18	[94]
Reaction Wheels	5	0.22	[95]
Magnetometer	0.5	0.02	[96]
X-Band Transceiver	10	0.45	[97]

C1.3 Contact Conductances

At this stage in the spacecraft design, no information is known on the use mounting methods for electronic components and the solar array. Hence estimates are made for the conductance coefficients of these components to the rest of the spacecraft structure. For all electronic components listed in Table C.1, a contact conductance coefficient of $1000 \text{ W/m}^2/\text{K}$ is used. This represents a lower estimate for the contact between an aluminium interface with 100 kPa contact pressure [31]. Solar panels are generally mounted in a thermally isolating manner to the rest of the spacecraft structure. Hence a conduction coefficient of $100 \text{ W/m}^2/\text{K}$ is used. This may represent a aluminium interface mounting, with a reduced effective area by ten times.

C1.4 Solar Array Honeycomb Insulation Thermal Conductivity

The layered structure used for the solar array is shown diagrammatically in Figure C.1. A honeycomb core has an aluminium face sheet adhered to either side. Solar cells are then attached using adhesive on one side. The honeycomb and face sheet thermal model is based on a design manufactured by CEL Components [98]. Whilst this is unlikely to be the actual design used on the TOLIMAN spacecraft, it provides a reasonable starting point given current information. This layered structure is modelled within Thermal Desktop as a single finite difference surface with unique nodes on each face to allow for through thickness temperature distributions.

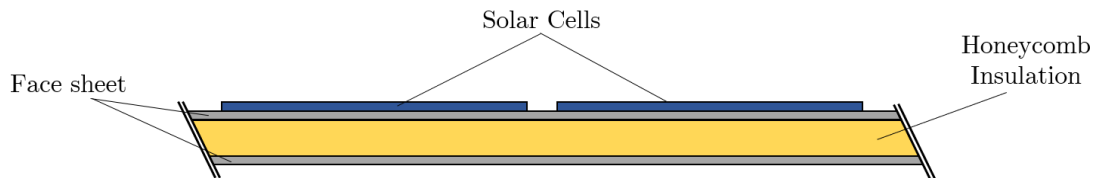


FIGURE C.1. Insulation layered structure used on spacecraft solar array. Solar cells are bonded with adhesive to one of the two face sheets of the honeycomb insulation layer.

The effective thermal conductivity of honeycomb insulation depends on the direction that is being considered, since the shape of the honeycomb structure itself is not isotropic. A schematic diagram of the core material for the honeycomb structure is shown in Figure C.2.

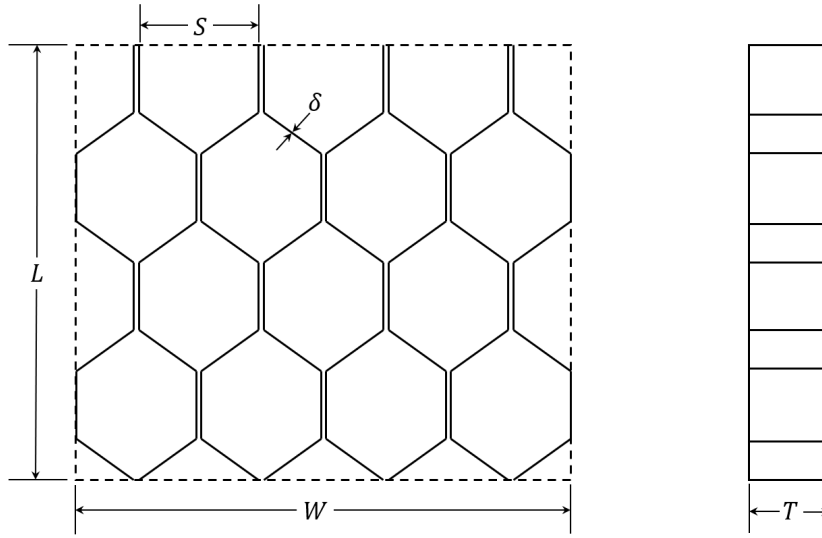


FIGURE C.2. Schematic diagram of honeycomb insulation, W , S , and T are the dimensions of the honeycomb sheet, S is the size of a single honeycomb cell, and δ is the thickness of the ribbon used to construct the honeycomb.

The method for calculating the thermal conductivity used is given by Gilmore [33]. Since the radiation heat transfer between the walls of the honeycomb ribbon is small, it is not considered. If k is the thermal conductivity of the ribbon material, then the effective thermal conductivity in the directions parallel to the L , W and T dimensions are

$$k_{L,eff} = \frac{3k\delta}{2S}, \quad (C.1)$$

$$k_{W,eff} = \frac{k\delta}{S}, \quad (C.2)$$

$$k_{T,eff} = \frac{32k\delta}{9S}. \quad (C.3)$$

Radiative heat transfer will occur between the two face sheets on either side of the honeycomb core. This is accounted for by defining an effective emissivity for the radiative transfer, ε^* . For two infinite parallel plates, the effective emissivity is given by [33]

$$\varepsilon^* = \left(\frac{1}{\varepsilon_1} + \frac{1}{\varepsilon_2} - 1 \right)^{-1}, \quad (C.4)$$

where ε_1 and ε_2 are the surface IR emissivities of each side of the face sheet.

Properties from honeycomb manufactured by CEL Components is used [98]. These have aluminium face sheets, so the emissivity is taken to be $\varepsilon_1 = \varepsilon_2 = 0.05$ (polished aluminium). Table C.3 shows the resulting effective conductivities and emissivity.

TABLE C.3. Effective thermal conductivities for honeycomb core and effective emissivity between face sheet material for [98] with $S = 0.01$ m, $\delta = 0.5$ μ m, and $k = 190$ W/m/K.

Effective Thermal Conductivity (W/m/K)	$k_{L,eff}$	5.70×10^{-4}
	$k_{W,eff}$	0.950
	$k_{T,eff}$	3.38
Effective Emissivity	ε^*	0.026

C2 Detector Model

C2.1 Lumped Component Properties

The ICs on the detector model are treated as single lumped nodes. The density (which allows the calculation of its mass from its volume determined from the CAD model of the detector) and the specific heat capacity of these components is treated as a 40/30/30 mass average of Silicon/Epoxy/Copper, of which the thermal properties can be found in Table B.1. This gives an average density of 4020 kg/m³, and specific heat capacity of 490 J/kg/K. The optical properties of these components is estimated to be $\alpha_S = \varepsilon_{IR} = 0.9$, based off their dark colour. The density of the PCB is chosen such that the total mass of the detector thermal model matches the 10 g true mass of the detector. The effective thermal conductivity of the PCB board is taken to be 8 W/m/K isotropically, based of the results by Nirale [75] and Costa and Vlassov [76].

C2.2 Component Power Dissipation

Table C.4 shows the nominal power consumption for the components on the detector, as well as the calculated power factor used in the thermal model. Important to note is that these figures were largely estimated from data sheets. In the absence of datasheets, it was estimated based on the specification that detector total nominal power consumption is approximately half its maximum power consumption, which is specified to be 3 W [71].

TABLE C.4. Nominal estimated heat generation and component power fractions for detector thermal model.

Component	Nominal Power (W)	Power Fraction	Ref.
FPGA	0.6	0.4	[72]
CMOS	0.6	0.4	[74]
SDRAM	0.15	0.1	[73]
Power Management	0.15	0.1	Best Estimate

C2.3 Contact Conductances

There are many ways of modelling heat transfer between an IC and the PCB it is mounted on. One simple approach proposed by Zhang and Feng [99] treats the connection between

the chip and board as a two resistor model, with thermal nodes representing the case and junction of the IC, and the PCB board. A further simplification is made here that the entire chip is treated as a single node. Whilst this approximation is coarse, it is fitting for the detector thermal model, especially given the available information. The contact conductance coefficients between the chips and board is taken to be $560 \text{ W/m}^2/\text{K}$ for all components. This is determined by dividing the junction-board conductance by the contact area from the FPGA datasheet [72]. Junction-board conductances are not available for other components on the board, so as a best estimate, this same figure is used for all.

The contact conductance coefficients between the spacers that separate the PCBs, as well as the contact between the PCB and standoff are taken to be $500 \text{ W/m}^2/\text{K}$, which represents a lower estimate for a copper-copper interface contact at 100 kPa contact pressure [31].

C3 Telescope Model

C3.1 Uniform Pressure Contact Conductance

Table C.5 shows the surface properties used for the calculation of bare surface contact conductance coefficients. Surface roughness properties are not available at this stage in the mission design. Generally such information is given with the manufacturing specifications of the parts. Order of magnitude estimates are then made for the RMS roughness, and typical values for the mean absolute value of the roughness slope are used from [65]. The Vickers coefficient c_1 is taken to be standard Vickers hardness that can be found in most material property references. The coefficient c_2 requires more specialised testing described by Hegazy [65]. This coefficient accounts for the dependence of hardness with depth. Since such testing results are not available, the coefficient c_2 is set to zero, hence neglecting the dependence of hardness with depth.

TABLE C.5. Estimated surface roughness properties used for calculation of bare surface contact conductances.

Material	RMS Roughness, $\sigma \text{ (m)}$	Mean Absolute Value Slope, m	Vickers Correlation Coefficient, $c_1 \text{ (GPa)}$	Vickers Correlation Coefficient, c_2	Ref.
SiC	1.0×10^{-6}	0.076	23.0	0	[90]
Aluminium	1.0×10^{-6}	0.076	1.05	0	[65]
Invar	1.0×10^{-6}	0.076	1.80	0	[65]
Ceramic	1.0×10^{-6}	0.076	19.0	0	[88]

Resulting bare surface conductances calculated using Equations 3.2 and 3.3 are shown in Table C.6. Material and surface properties from Tables B.1 and C.5 are used for these calculations.

C3.2 Adhesive Contact Conductance

The primary and secondary mirrors are bonded to the telescope structure. Bonding adhesive properties are unavailable, hence an estimate of 2.4 W/m/K for the thermal conductivity is made based on several common adhesives tested by Mirmira [34]. Bond line thickness is

TABLE C.6. Bare surface contact conductance coefficients for various interfaces used in thermal analysis for contact pressures of 30 kPa.

Contact	Bare Contact Conductance Coefficient(W/m ² /K)
SiC-SiC	53
SiC - Aluminium	870
SiC - Invar	55
SiC - Ceramic	35
Aluminium - Ceramic	11

estimated to be 100 μm , and a surface true indicator reading (TIR) is estimated as 15 μm . Using the Equation 3.5, and the data in Tables C.5 and C.6, a contact conductance coefficient of approximately 2200 W/m²/K is calculated. This is relatively high, and matches the discussion with the telescope manufacturer that the bond interface will be thermally very continuous. Hence this approximation seems reasonable.

C3.3 Bolted Joint Contact Conductance

The bolted connection contact conductance between the detector mount and primary mirror is estimated using the correlation given by Equation 3.4. Using the thermal conductivities of Invar 36 and the SiC structure given in Table B.1, with a bolt circle of three bolts, each with a torque of 1.5 N·m, a contact conductance of 360 W/m²/K is calculated. This is a relatively low conductance due to the low thermal conductivity of the Invar detector mount.

C4 Patch Heater Modeling

Patch heaters and their associated sensors are modelled using finite difference surfaces. This way, a heat capacity and thermal contact conductance to the telescope structure can be assigned. This arrangement is shown in Figure C.3. A Thermal Desktop contactor is created between the heater and the telescope structure surface, meaning the sensor region does not affect the heat transfer from the the patch heater – a valid assumption since the actual sensors will be very small. A contactor is also used between the sensor region and the telescope.

The average temperature over the sensor region is taken to be the measured temperature, this represents taking the average temperature from a line of sensors situated beneath the heater. Heater properties are modelled after polyimide thermofoil heaters by Minco [78]. The heaters are treated as 2 mm thick 40/60 mix of copper/kapton. This gives the effective density to be $\rho = 4420 \text{ kg/m}^3$, thermal conductivity of 160 W/m/K, and specific heat capacity of $c_p = 756 \text{ J/kg/K}$ [88]. The surface optical properties are taken to be that of 2 mil aluminized kapton, with $\alpha_s = 0.45$ and $\varepsilon_{IR} = 0.73$ [33]. The same properties are used for the sensing region, but no optical properties are used as they are not included in radiation calculations. Contact conductance coefficients to the spacecraft structure are taken to be 1000 W/m²/K for the heaters, and 10000 W/m²/K for the sensor regions. These are large, but not infinite values

that represent the good thermal coupling due to the epoxy used to bond the patch heaters and sensors to surfaces.

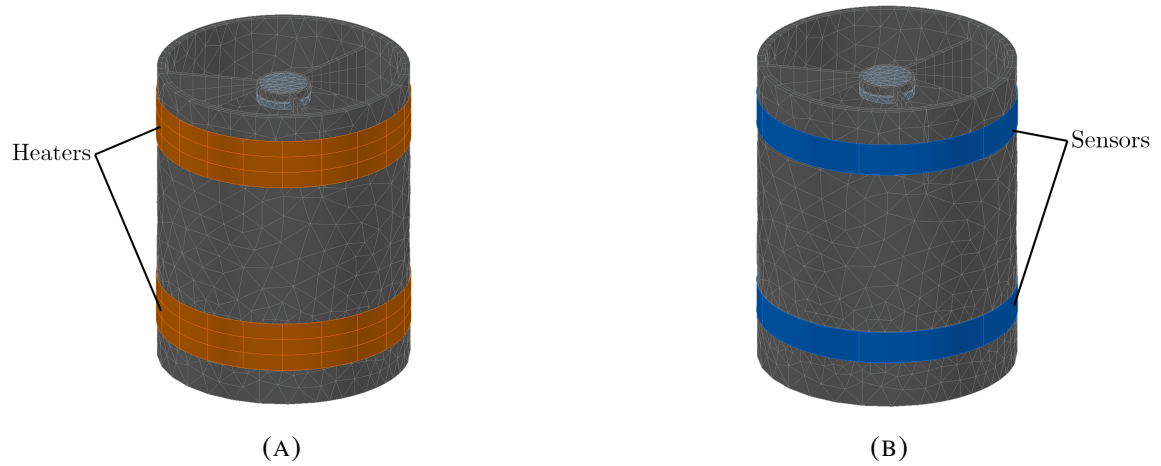


FIGURE C.3. Finite element thermal model for telescope in Thermal Desktop with finite difference surfaces used to represent patch heaters (A) and sensing regions beneath the patch heaters (B).

APPENDIX D

Mesh Convergence Study of Telescope Thermal Model

To save computational cost, the coarsest possible mesh was chosen that achieves satisfactory convergence. Mesh convergence was conducted here with a steady state analysis for the entire satellite model. This way the thermal condition used in the mesh convergence study is most similar to the true cases it will be used for in simulation. The mesh size is varied between meshes consisting of 2923 elements to 22 312 elements. For each mesh, a steady state simulation was run, and the maximum and minimum temperature of the telescope structure was recorded. The results are shown in Figure D.1. Note that the simulation was run without any heat input from the detector. For all mesh sizes, the maximum and minimum temperatures are within less than 1% of each other, which is satisfactory accuracy for this study. A mesh consisting of 6928 elements is used in the final model as this was the finest mesh which gave acceptable computation times in transient orbit simulations.

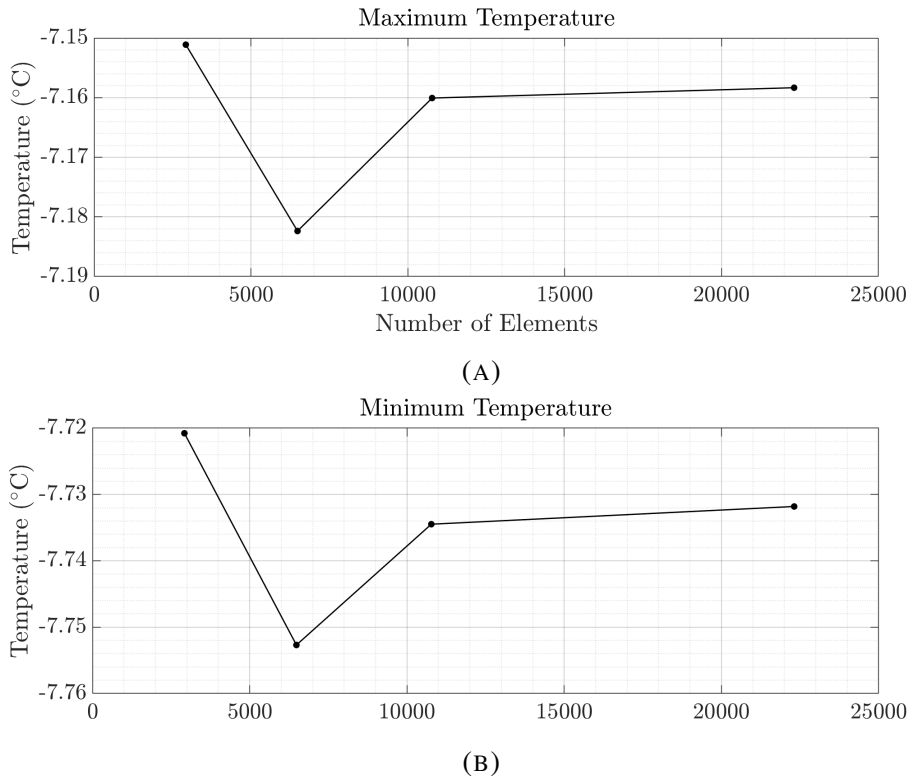


FIGURE D.1. Mesh convergence study results for thermal finite element model of telescope, maximum (A) and minimum (B) temperatures on the telescope structure are used.

APPENDIX E

Mesh Convergence Study of Telescope Structural Model

A mesh convergence study was performed to determine an appropriate mesh size, while keeping computational cost low. A structural deformation analysis was done on the telescope structural model for a range of mesh sizes. The thermal condition added was a uniform temperature soak of 19°C applied to the entire telescope structure. The maximum and minimum change in sag on the mirror surface (Δs) was recorded for mesh sizes consisting of 1546 elements to 162 890 elements. The results can be found in Figure E.1. For meshes consisting of 9217 elements and above, the model is sufficiently converged, giving less than 1% error between solutions. A final model consisting of 91 819 elements was used. Worth noting is that the finite element mesh used for structural analysis was required to be much finer than that of the thermal model (Appendix D).

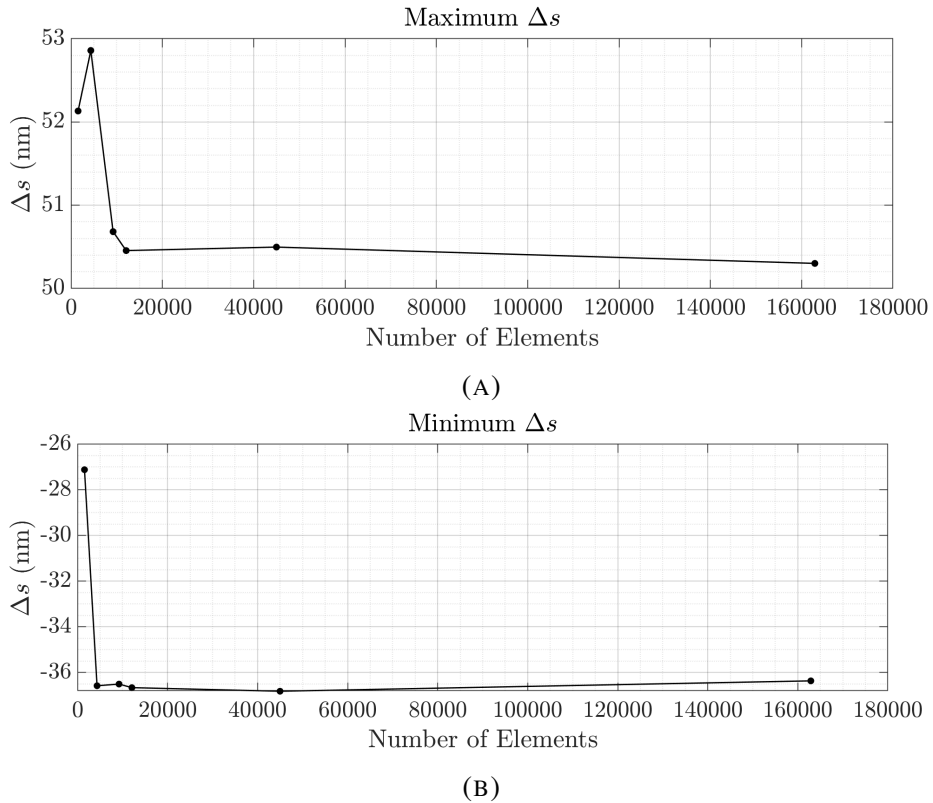


FIGURE E.1. Mesh convergence study results for structural finite element model of telescope, maximum (A) and minimum (B) Δs on the primary mirror are used.

Orbit Propagator

F1 Orbital Mechanics

F1.1 Coordinate Systems

Describing the position of objects around the Earth (such as stars) is most conveniently done using the Earth-Centered Inertial (ECI) reference frame. Here, the reference frame has its origin at the center of the Earth with the coordinate axis fixed with respect to the stars as shown in Figure F.1. The X axis is defined by the reference direction, which is the vernal equinox, and the Z axis is defined by the celestial north pole. The right ascension, α and declination, δ are angles measured from the reference direction (the vernal equinox) and the celestial equator, and are commonly used to denote the position of stars.

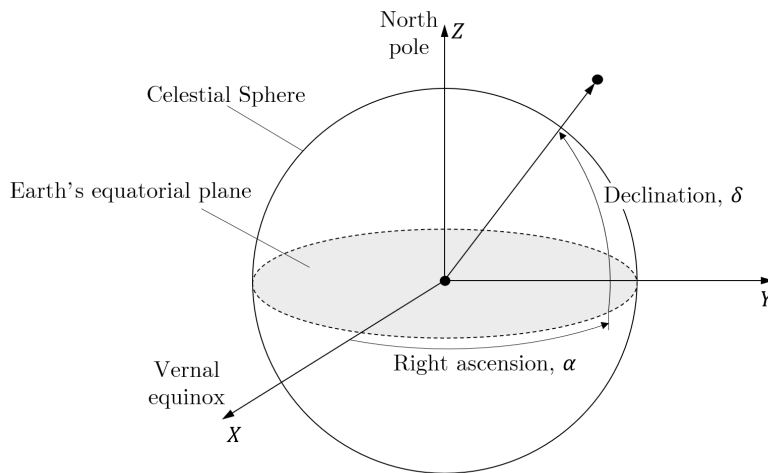


FIGURE F.1. ECI frame defined by reference directions, the vernal equinox and north pole, with right ascension α and declination δ used to define the location of a point on the celestial sphere.

The six classical orbital elements are used to define an elliptical orbit of a spacecraft around the Earth. This is also the preferred coordinate system used when calculating the dynamics of an orbit in the two body problem. The six classical orbital elements are listed below (note that

only the magnitude of vector quantities are used) [100]

h : Specific angular momentum.

i : inclination.

Ω : Right ascension of the ascending node.

e : Eccentricity.

ω : Argument of Perigee.

θ : True anomaly.

These orbital elements are illustrated in Figure F.2. Often times, the specific angular momentum h is replaced with the semi-major axis of the orbit a (which is the semi-major axis of the ellipse describing the orbit), or as is the case with a Two Line-Element (TLE), the mean motion n , and the true anomaly θ may be replaced with the mean anomaly M .

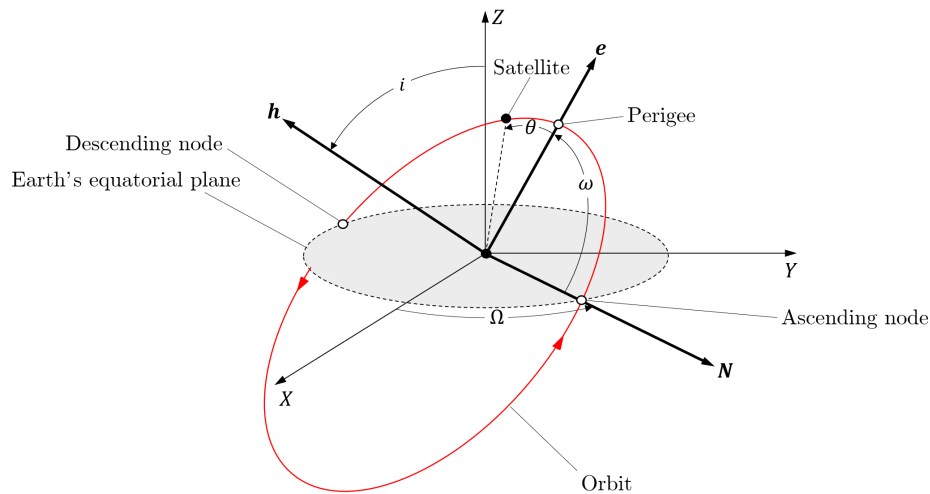


FIGURE F.2. An illustration of six classical orbital elements used to describe an orbit and the position of a satellite within it. The angular momentum vector h is perpendicular to the orbital plane, the eccentricity vector e passes through the point of closest approach (the perigee), and the node line vector N passes through the ascending node and lies within Earth's orbital plane.

F1.2 Orbits as a Function of Time

The dynamics of a satellite in orbit is dealt with here using the two body problem, that is, only the gravitational effects of the Earth on the spacecraft are considered. For a spacecraft in orbit around a perfectly homogeneous sphere, all orbital elements remain constant over time except for the mean, eccentric, and true anomaly (M , E , and θ). If we define the mean motion as the

average angular velocity of the orbit

$$n = \frac{2\pi}{T}, \quad (\text{F.1})$$

where T is the period of the orbit, then the mean anomaly is the azimuth of a fictitious body moving around the ellipse at constant angular speed n , that is $M = n(t - t_0)$, where t_0 is the initial time. The eccentric anomaly is defined by circumscribing an auxiliary circle around the orbit, and is related to the mean anomaly by Keplers equation [100]

$$M = E - e \sin E. \quad (\text{F.2})$$

Note this equation is transcendental in E , so to calculate the eccentric anomaly from the mean anomaly, it must be solved numerically. The Newton-Raphson method is used in this case. Given the eccentric anomaly, the true anomaly is found through

$$\tan \frac{E}{2} = \sqrt{\frac{1-e}{1+e}} \tan \frac{\theta}{2}. \quad (\text{F.3})$$

With this, propagating an orbit through time becomes simple. The first step is to obtain the mean motion of the orbit, then to calculate the mean anomaly as a function of time through $M = n(t - t_0)$, then calculate the eccentric anomaly through Equation F.2, then calculate the true anomaly with Equation F.3. The mean motion is determined from the period of the orbit T , as shown in Equation F.1. If the semi-major axis of the orbit is known, the period can be calculated from Kepler's law

$$T = \frac{2\pi}{\sqrt{\mu}} a^{3/2}, \quad (\text{F.4})$$

where $\mu = 3.9860 \times 10^{14} \text{ m}^3 \text{ s}^{-2}$ is the gravitational parameter for the Earth. If the angular momentum of the orbit is known, the semi-major axis can be calculated through

$$a = \frac{h^2}{\mu} \frac{1}{1 - e^2}.$$

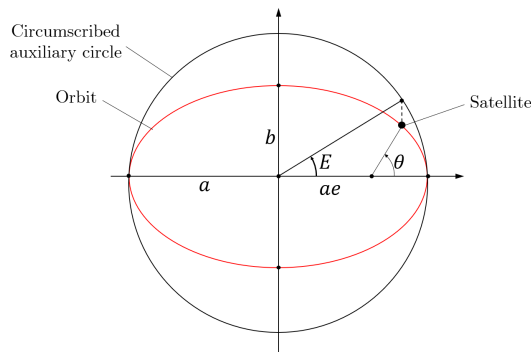


FIGURE F.3. The true anomaly θ is the angle measured from the perigee centered at the focus of the orbit, the eccentric anomaly E is the angle measured from the center of the circumscribed auxiliary circle of the intersection of the line project vertically from the satellite position onto the circle.

F1.3 Perturbations

In actuality the Earth is non-spherical, which causes anomalies in the gravitational field as a satellite moves around the Earth. The most prominent is the primary equatorial bulge due to the Earth's rotation as modelled by the J_2 zonal harmonic. There is approximately a 22 km difference in radius of the Earth between the poles and the equator. The result is a temporal drift of the right ascension of the ascending node, and argument of perigee, as well as a correction to the mean motion. The corrected mean motion $\bar{n}$ and the drift rates $\dot{\omega}$ and $\dot{\Omega}$ are given by [101]

$$\bar{n} = \sqrt{\frac{\mu}{a^3}} \left[1 + \frac{3}{2} \frac{J_2 R^2}{p^2} \left(1 - \frac{3}{2} \sin^2 i \right) (1 - e^2)^{1/2} \right] \quad (\text{F.5})$$

$$\dot{\omega} = \frac{3}{2} \frac{J_2 R^2}{p^2} \bar{n} \left(2 - \frac{5}{2} \sin^2 i \right) \quad (\text{F.6})$$

$$\dot{\Omega} = -\frac{3}{2} \frac{J_2 R^2}{p^2} \bar{n} \cos i, \quad (\text{F.7})$$

where R is the Earth's equatorial radius, $p = a(1 - e^2)$, and $J_2 = 0.00108263$. These corrections result in the orbit having a westward drift over time, as well as a rotation of the perigee.

F1.4 Solar Zenith Angle

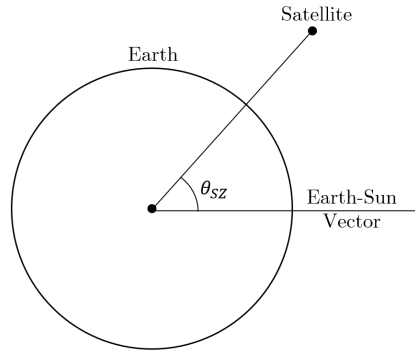


FIGURE F.4. Illustration of the solar zenith angle θ_{SZ} , the angle between the Earth-Sun vector and the Earth-satellite vector.

F1.5 Calculation of Earth Vector

The vector from the Earth to the spacecraft is simply the position vector of the spacecraft in the ECI frame. This is found by converting the classical orbital elements $(h, e, \Omega, i, \omega, \theta)$ into cartesian coordinates in the ECI frame. First the position is converted from the orbital elements

into the “perifocal frame” – a cartesian frame centered at the focus of the orbit [100]

$$\mathbf{r}_p = \frac{h^2}{\mu} \frac{1}{1 + e \cos \theta} \begin{bmatrix} \cos \theta \\ \sin \theta \\ 0 \end{bmatrix},$$

where $\mathbf{r}_p$ is the position vector in the perifocal frame. Then three successive rotation matrices are applied to give the ECI position vector

$$\mathbf{r} = R_3^T(\Omega) R_1^T(i) R_3^T(\omega) \mathbf{r}_p$$

where

$$\begin{aligned} R_3^T(\Omega) &= \begin{bmatrix} \cos(\Omega) & \sin(\Omega) & 0 \\ -\sin(\Omega) & \cos(\Omega) & 0 \\ 0 & 0 & 1 \end{bmatrix} \\ R_1^T(i) &= \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos(i) & \sin(i) \\ 0 & -\sin(i) & \cos(i) \end{bmatrix} \\ R_3^T(\omega) &= \begin{bmatrix} \cos(\omega) & \sin(\omega) & 0 \\ -\sin(\omega) & \cos(\omega) & 0 \\ 0 & 0 & 1 \end{bmatrix}. \end{aligned}$$

F1.6 Calculation of Sun Vector

For determining the position of the sun, very precise ephemerides are available from JPL for example, however, it is more convenient to use a mathematical approximation for the implementation in a computer program. The method used here is accurate to 0.01° and is valid between the years 1950 to 2050, and can be found in Vallado [102]. First the number of Julian centuries T_{UT1} from the J2000 epoch is found given the Julian Date JD_{UT1} with

$$T_{UT1} = \frac{JD_{UT1} - 2451545}{36525}.$$

This can then be used to find the mean longitude of the sun

$$\lambda_{M_\odot} = 280.4606184^\circ + 36000.77005361 \cdot T_{UT1}.$$

The solar mean anomaly is calculated through

$$M_\odot = 357.5277233^\circ + 35999.05035 \cdot T_{UT1}.$$

The ecliptic longitude can then be found as

$$\lambda_{ecliptic} = \lambda_{M_{\odot}} + 1.914\,666\,471^{\circ} \cdot \sin(M_{\odot}) + 0.001\,999\,464\,3 \cdot \sin(2M_{\odot}).$$

The solar distance can be found through

$$r_{\odot} = 1.000\,140\,612 - 0.016\,708\,617 \cdot \cos(M_{\odot}) + 0.000\,139\,589 \cdot \cos(2M_{\odot}).$$

Finally, the angle between the ecliptic and celestial equator is found with

$$\varepsilon = 23.439\,291^{\circ} - 0.013\,004\,2 \cdot T_{UT1}.$$

The sun vector is then given by

$$\mathbf{r}_{\odot} = \begin{bmatrix} r_{\odot} \cos(\lambda_{ecliptic}) \\ r_{\odot} \cos(\varepsilon) \sin(\lambda_{ecliptic}) \\ r_{\odot} \sin(\varepsilon) \sin(\lambda_{ecliptic}) \end{bmatrix} \text{ AU},$$

where AU is an astronomical unit.

F1.7 Eclipse Determination

Given the Earth vector $\mathbf{r}_{\oplus}$ and Sun vector $\mathbf{r}_{\odot}$ of the spacecraft, determining whether or not the spacecraft is in eclipse is a problem of determining if the relative position vector between them intersects the Earth. It will be assumed that the Earth is spherical. The general algorithm for determining if the relative position vector between two vectors $\mathbf{r}_1$ and $\mathbf{r}_2$ in the ECI frame intersects the Earth is given by Vallado [102] and presented in Algorithm 1. Note that the vectors $\mathbf{r}_1$ and $\mathbf{r}_2$ are in units of the average Earth radius.

Algorithm 1 Eclipse

- 1: $\tau_{min} = \frac{|\mathbf{r}_1|^2 - \mathbf{r}_1 \cdot \mathbf{r}_2}{|\mathbf{r}_1|^2 + |\mathbf{r}_2|^2 - 2\mathbf{r}_1 \cdot \mathbf{r}_2}$
 - 2: ECLIPSE = TRUE
 - 3: **if** ($\tau_{min} < 0$) or ($\tau_{min} > 1$) **then**
 - 4: ECLIPSE = FALSE
 - 5: **else if** ($(1 - \tau_{min})|\mathbf{r}_1|^2 + (\mathbf{r}_1 \cdot \mathbf{r}_2)\tau_{min} \geq 1$) **then**
 - 6: ECLIPSE = FALSE
 - 7: **end if**
-

F2 Orbit Propagator Algorithm

The algorithm implemented in MATLAB to propagate the orbit is given in Algorithm 2, and uses all the basic ideas discussed previously. After the completion of the algorithm, the program outputs the orbital elements (the ones which vary with time - θ , Ω , and ω) at each time step. The methods given in Appendix F1.5 and F1.6 to calculate the position of the Sun and Earth

vectors as a function of time throughout the orbit. Algorithm 1 can be used to determine if the spacecraft is in eclipse or not. The vector representing the optical axis of the telescope is simply the vector pointing towards Alpha Centauri (Assuming the telescope is looking at Alpha Centauri) and can be found from the right ascension and declination of Alpha Centaur. The angle ϕ between any two vectors $\mathbf{a}$ and $\mathbf{b}$ can be found using the definition of the dot product, $\cos \phi = \mathbf{a} \cdot \mathbf{b} / |\mathbf{a}| |\mathbf{b}|$.

Algorithm 2 Orbit Propagator

```

1: % Initialise orbital parameters
2:  $n, \Omega, \omega, \theta, M, E, i, a$ 
3:
4: % Calculate drift rates due to orbital perturbations
5: if (perturbations) then
6:    $\bar{n} = \sqrt{\frac{\mu}{a^3}} \left[ 1 + \frac{3}{2} \frac{J_2 R^2}{p^2} \left( 1 - \frac{3}{2} \sin^2 i \right) (1 - e^2)^{1/2} \right]$ 
7:    $\dot{\Omega} = -\frac{3}{2} \frac{J_2 R^2}{p^2} \bar{n} \cos i$ 
8:    $\dot{\omega} = \frac{3}{2} \frac{J_2 R^2}{p^2} \bar{n} \left( 2 - \frac{5}{2} \sin^2 i \right)$ 
9: else
10:   $\bar{n} = n$ 
11:   $\dot{\Omega} = 0$ 
12:   $\dot{\omega} = 0$ 
13: end if
14:
15: % Iterate time
16: for  $t = t_0, t_{End}$  do
17:    $M = M_0 + \bar{n}(t - t_0)$  % Update mean Anomaly
18:    $E = \text{eccentricAnomaly}(M, e)$  % Eccentric anomaly using Newton's Method
19:    $\theta = 2 \arctan \left( \sqrt{\frac{1+e}{1-e}} \tan \left( \frac{E}{2} \right) \right)$  % True anomaly
20:    $\Omega = \Omega + \dot{\Omega}(t - t_0)$  % RAAN due to orbital perturbation
21:    $\omega = \omega + \dot{\omega}(t - t_0)$  % Argument of perigee due to orbital perturbation
22: end for

```
