Thermal Environment for Polar Communications and Weather System in the Telesat-Tundra Orbit

by

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Abstract

This thesis examines the thermal environment and control system for the Polar Communications and Weather (PCW) satellite system as proposed by Telesat. The Telesat-PCW will use a unique orbit, based on the Tundra orbit to provide continuous meteorological and communications coverage to the Arctic region. The Telesat Tundra Orbit (TTO) was selected as it can meet the mission requirements using systems designed for Geostationary Orbits (GEO), a mature and robust field. However, the unique combination of communications and meteorological payloads, coupled with a previously unused orbit, presents an unexamined thermal design scenario. Using an existing geostationary satellite design for PCW would significantly reduce costs. This thesis describes the spacecraft thermal environment of the TTO, compares it to the thermal environment of the geostationary orbit, and numerically analyzes the suitability of a typical geostationary satellite thermal design system for PCW by using a Thermal Mathematical Model (TMM). The analysis shows that a thermal control system for a typical GEO communications satellite can sufficiently regulate PCW spacecraft temperatures in the TTO.

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Acronyms and Abbreviations

ABI	Advanced Baseline Imager	MTG	Meteosat Third Generation
ADCS	Attitude Determination and Control System	OMUX	Output MULTipleXer
BOL	Beginning Of Life	OSR	Optical Surface Reflectors
CAD	Computer Aided Design	PCW	Polar Communications and Weather
EOL	End Of Life	RLG	Ring Laser Gyro
EPC	Electronic Power Conditioner	RW	Reaction Wheel
FEA	Finite Element Analysis	Rx	Receive
FEM	Finite Element Method	SSM	Second Surface Mirrors
GEO	Geostationary Orbit	SSPA	Solid State Power Amplifier
GOES	Geostationary Orbit Environmental Satellite	STK	Satellite Tool Kit
HEC	High Efficiency Cryocooler	TAP	Three Apogee Orbit
HEO	Highly Elliptical Orbit	TCS	Thermal Control System
IMUX	Input MULTipleXer	TMM	Thermal Mathematical Model
IR	Infrared Radiation	TTC	Telemetry Tracking and Command
LCAMP	Linearized Channel AMplifier	TTO	Telesat Tundra Orbit
LEO	Low Earth Orbit	TWTA	Travelling Wave Tube Amplifier
LHP	Loop Heat Pipe	Tx	Transmit
MET	Meteorological	UHF	Ultra High Frequency
MLI	Multi-Layer Insulation	URD	User Requirements Document
MMH	Monomethyl Hydrazine	WGS	Wideband Global SATCOM
MODIS	MODerate resolution Imaging Spectrometer		

Part 1: Introduction

In response to the Government of Canada's publishing of a User Requirements Document (URD) for a Polar Communications and Weather (PCW) satellite system (1), Telesat has proposed a system of two, three-axis stabilized satellites orbiting in the Telesat Tundra Orbit (TTO), pictured in The TTO is a variation of an elliptical geosynchronous orbit with four of the six Classical Orbital Elements given in Table 1-1. The two remaining orbit elements are the Longitude of the Ascending Node and the Mean Anomaly. The Mean Anomaly varies through the course of the day, and the Longitude of the Ascending Node for TTO has not been selected yet, and is dealt with in this thesis. The high inclination combined with selecting an apogee over the North Pole provides 24 hour continuous coverage of the Arctic region. (2)

Figure 1-1 Telesat Tundra Orbit

Table 1-1 Telesat Tundra Orbit Mean Classical Elements

	Symbol	TTO
Semi Major Axis	a	33063 km
Eccentricity	e	0.3
Inclination	i	90°
Argument of Perigee	ω	270°

Both the orbit and the proposed payload mix are unique, and as such, the thermal environment and the impact on the Thermal Control System (TCS) design are not well understood. The classic Tundra orbit has been used by communications satellites previously. Sirius Satellite Radio's Radiosat 1, 2 and 3 all use the classic Tundra orbit at 63.4° inclination to provide satellite radio service to North America. (3)

This thesis will examine the TCS design considerations for a meteorological communications satellite in the TTO orbit. Specifically this thesis will analyze whether a typical geostationary communications satellite TCS is capable of providing the appropriate thermal environment, and evaluate the potential need for advanced thermal control techniques

Part 1 outlines the motivation for the PCW system. Part 2 comprises a survey of available literature on PCW related topics, and the reasoning behind Telesat's proposed orbit. Part 3 describes the spacecraft design assumptions, calculations and decisions, the thermal design analysis, and the finite element modeling. Part 4 presents the results of the simulations of the hot and cold worst cases analysis done in the NX™ Space Systems Thermal software. Conclusions and recommendations for future work are presented in Part 5.

1.1.1. The Arctic

There is an increasing interest in the Arctic region as average global temperatures rise and the Arctic ice recedes. With less difficult access to the Arctic due to ice, significant natural resources become accessible and much shorter seaways become available between Asia, Europe and North America. In 2013, for the first time, a vessel successfully navigated the Arctic Ocean during winter (4), and ocean traffic through the Northwest Passage is steadily increasing. This increased traffic will bring increased demand for secure reliable communications.

Additionally, the increased traffic will drive increased demand for accurate weather forecasting. Observing the Arctic environment is crucial to that and other applications. The effects of climate change are more pronounced in the area, making it a valuable area for climate change predictions. Effects of air pollution and volcanic ash transport are more pronounced there, and the Arctic environment has an impact on global weather systems as a whole (5). Improved knowledge of the Arctic weather systems will help improve global weather models.

1.1.2. Current Communications Coverage

Current communications is dominated by geostationary (GEO) satellites, such as Anik F2 and the Global Wideband System (WGS) (6). The GEO orbit is generally desirable for communications, as these satellites maintain a nearly constant position in the sky. This makes ground station antenna pointing a trivial problem, and provides suitable coverage over much of the Earth disk.

Where GEO satellites fall short is at the extreme Northern (and Southern) latitudes. Particularly above 70° North latitude, communications reliability decreases (7). Two things happen as latitude is increased: one, the line of sight to the satellite passes through more of the Earth's atmosphere, exacerbating weather and atmospheric interference, and two, the elevation angle of the ground antenna decreases to the point where local geographical features are increasingly in the line of site, thus blocking communication. Non-GEO systems are required to cover the entire polar region. Above 80 degrees north latitude there is no visibility to GEO as the elevation angle is less than zero.

1.1.3. Current Meteorological Coverage

For reasons similar to satellite communications, much of the Earth's meteorological (MET) coverage is carried out by GEO satellites such as Meteosat and Geostationary Operational Environmental Satellite (GOES) systems. These systems provide continuous weather monitoring, but face the same limitations as do GEO communications satellites: coverage of latitudes north of 70° is limited or impossible.

There are polar orbiting satellites providing a measure of MET coverage of the Arctic, such as the A-Train constellation (8). However, according to Kroupnik (9), the system's refresh rates are limited, data latency prevents its use in real-time weather reporting and a coverage gap remains between these polar satellites and the GEO systems.

One requirement of the PCW system is that observations of the same region take place no less than every 20 minutes. At an altitude of 690 km, the A-train Constellation satellites only revisit the Arctic roughly every 100 minutes. Many satellites would be needed in

this type of orbit to provide continuous 24 hour coverage, making a cost prohibitive alternative.

1.1.4. Polar Communications and Weather System

In 2011 the Government of Canada released a document outlining requirements for PCW (1). PCW is intended to address communications and meteorological service gaps and limitations of systems operating in geostationary and polar orbits. The main requirements were for 24 hour coverage of the Arctic for meteorological measurements and high frequency (X- and Ka-Band) communications. More specifically, the meteorological requirements amounted to continuous coverage of the polar region north of 60° with a local zenith angle no less than 70° , as shown in Figure 1-2.

Figure 1-2 PCW Viewing Criteria

1.1.5. TTO Solution

Telesat has proposed a system to provide these services in conjunction with meeting the PCW requirements. Bigras et. al. (2) described a system using two satellites in a modified Tundra orbit which has a radiation environment that is less severe than the other PCW alternatives. As such, the Tundra satellites could be designed for a GEO-like design life of 15-18 years, drastically reducing the cost of a long term solution.

Similarly, the requested payloads suggest that a commercially available GEO satellite bus could be used instead of designing a specific unique platform. This thesis will evaluate the feasibility of using a GEO satellite bus design in the TTO orbit as it pertains to the thermal environment.

1.2. Aims and Objectives

The main goal of this thesis is to assess the suitability of GEO-like TCS designs to the TTO orbit. In doing so, the following goals are explored.

1. Identify worst case hot and worst case cold conditions.
2. Characterize eclipse durations and thermal effect.
3. Characterize impact of the TTO orbit on thermal design.
4. Analyze the suitability of typical GEO TCS to the Tundra orbit PCW system.

NX™ Space Systems Thermal, a finite element analysis (FEA) software from Siemens, is specifically designed for spacecraft thermal environment analysis, and will be used to perform the analysis presented in Part 4. The Space Systems Thermal package is based on I-deas TMG Finite Element Modeling, from MAYA, a Canadian company. The thermal modeling package is integrated with NX™ Computer Aided Design (CAD).

Part 2: Literature Survey

Part 2 is a summary of a survey of the applicable literature. The PCW system proposed by Telesat is unique due to the use of a previously undefined orbit, and due to the combination of high power communications with high resolution multispectral imaging.

As such, very little information on PCW or the Telesat system exists in the literature. The survey concentrates on the literature regarding the potential PCW orbits that were considered, on the Advanced Baseline Imager (ABI) as the potential meteorological (MET) payload, and on analogous missions such as Sirius Radio which used a critically inclined Tundra orbit, and GOES-N which is a weather mission in GEO that uses a precursor device to the ABI.

2.1. Candidate Orbits

Three main candidate orbits for achieving the coverage defined by PCW have been identified. All three are highly inclined and are listed below with their orbit period.

1. Molniya – 12 hour
2. Three Apogee (TAP) – 16 hour
3. Telesat Tundra Orbit (TTO) – 24 hour

Figure 2-1 shows the candidate orbit ground traces, modeled using Satellite Tool Kit (STK) from Analytical Graphics Inc.

Figure 2-1 PCW Candidate Orbit Ground Traces

2.1.1. Molniya Orbit

Kidder and Vonder Haar (10) described the potential use of Molniya orbits for meteorological (MET) observation. The Molniya is a Highly Elliptical Orbit (HEO) that has been extensively used by the Soviet Union/Russia as a means of maximizing coverage of the northern hemisphere. Mean orbit elements are shown in Table 2-1.

Table 2-1 Molniya Orbit Mean Elements ¹

	Symbol	Molniya
Period	T	12 hour
Eccentricity	e	0.75
Inclination	i	63.4°
Argument of Perigee	ω	270°
Apogee Altitude		39819 km
Perigee Altitude		530 km

¹ The elements, Period, Eccentricity, Inclination and Argument of Perigee are four of the 6 required to define a satellite's position in space. The other two could be time since perigee passage, and longitude of the ascending node which define the satellite's instantaneous position along the orbit and the orbit's plane orientation in space. Since we are describing the orbits themselves here, the instantaneous position is irrelevant. The effect of the orientation of the orbit plane in space will have an effect on the thermal environment, particularly the eclipse seasons and will be discussed in a later chapter.

The eccentricity of 0.70-0.75, combined with an argument of perigee of 270° causes a satellite to spend the majority of its time north of the equator. Such a satellite moves slowly near the apogee and more rapidly through perigee. Thus the satellite will appear to dwell around the apogee and allow for minimal ground antenna tracking motion.

The 12 hour period results in two apogees separated by 180° , as seen in Figure 2-1 PCW Candidate Orbit Ground Traces. Two satellites, separated by 6 hours, would be needed to meet PCW requirements which results in the four apogees seen in Figure 2-1.

The inclination of 63.4° is designed to minimize the precession of the argument of perigee due to orbit perturbations. The rate of drift of perigee is given by Eq. (2.1).

$$\frac{\partial \omega}{\partial t} = \frac{3}{2} J_2 n \left(\frac{R}{a(1-e^2)} \right)^2 \left(2 - \frac{5}{2} \sin^2 i \right) \quad (2.1)$$

Where J_2 is the dominating term describing the Earth's oblateness, n is the mean motion, a is the semi major axis, e is the eccentricity and i is the inclination. At $i = 63.4^\circ$, the term in the second bracket on the right hand side of the equation vanishes. This gives rise to a 'frozen' argument of perigee, which minimizes the need for maneuvers to correct for perigee drift, and therefore tend to keep the apogee of the orbit over the North. This value of i is referred to as the *critical inclination*.

The main drawbacks to the Molniya orbit are that the low perigee causes the orbit to pass through the high energy proton region of the Van Allen Belts and induces drag effects due to the (albeit thin) atmosphere. The effect of drag needs to be compensated using thrusters. The radiation environment of the considered orbits is described in Section 2.2.

2.1.2. TAP Orbit

The TAP orbit is an alternate HEO, described by Trishchenko and Garand in (5), (11) and (12). The TAP orbit was proposed based on optimization of Arctic coverage, the resolution capability, the radiation environment, and the orbit stability, with the target being a system capable of achieving the coverage goal with two satellites. The results of the optimization study are found in (12). The optimization parameters can be summarized as follows:

1. 100% coverage of the region north of 60°N.
2. Apogee altitude no greater than 45,000 km.
3. Semi-latus rectum² at equator greater than 15000 km to reduce exposure to the high energy proton ($E > 10$ MeV) region.
4. Stable apogee and easy to maintain.

The analysis began by interpreting the coverage requirement as being that the satellite should spend greater than 50% of its time above 45°N. The conclusion they arrived at was that $e > 0.5$ is desirable, though they noted that by increasing the inclination from the critical value to 70°N would allow less eccentric orbits to achieve the desired coverage at the cost of increased fuel to counter the increase in perigee drift. The next parameter optimized was the spatial resolution. They showed that 24 hour orbits could not achieve the $e > 0.5$ constraint, thus limiting orbit selection to periods of less than 18 hours.

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² The semi-latus rectum, L , is half the length of a chord perpendicular to the semi-major axis, and passing through the “occupied” focus of a conic section. Mathematically it is defined as $L=a(1-e^2)$. In this case it refers specifically to the altitude of the satellite at the ascending and descending nodes. It is an important parameter because the Van Allen Belts are at a maximum intensity along the equator, so the semi latus rectum bounds the worst case radiation dose.

The constraint on semi-latus rectum was derived by targeting a region where the proton flux for energies $E > 10$ MeV is less than $10 \text{ cm}^{-3}\text{s}^{-1}$. At this level the proton environment was calculated to be 10^4 times smaller than for a Molniya orbit. The optimal orbit according to (12) is a 16 hour orbit which would repeat every 1.5 days. Each satellite would traverse three apogee locations before repeating the ground trace shown in Figure 2-1.

The TAP orbit was shown to be capable of providing the desired coverage at the critical inclination, mitigating the propellant needed for perigee correction maneuvers. The mean orbit parameters are found in Table 2-2.

Table 2-2 TAP Orbit Mean Parameters

	Symbol	TAP
Period	T	16 hour
Eccentricity	e	0.55
Inclination	i	63.4°
Argument of Perigee	ω	270°
Apogee Altitude		43496 km
Perigee Altitude		8102 km

2.1.3. Telesat Tundra Orbit

Bigras et al. proposed a modified Tundra orbit in (2). A traditional Tundra orbit is an elliptical, inclined orbit that is less eccentric than TAP or Molniya. The Tundra orbit is a 24 hour orbit with eccentricity between 0.25-0.30, inclined at the critical inclination of 63.4°. It was examined as a candidate for PCW, but to achieve the continuous Arctic coverage requirement it was found to require three satellites. Sirius Satellite Radio used a Tundra orbit for its first three satellites, as described by Riza in (13), as it provided a better look angle over the northern hemisphere than a satellite at GEO orbit could.

According to Bigras et al. the polar coverage could be achieved by a two satellite system if the satellites were in a Tundra orbit with an inclination between 80-90°. This modified Tundra orbit, the TTO, would enable two satellites to achieve the coverage, while still avoiding exposure of the satellite the harshest areas of the Van Allen belts. The reduced exposure would allow for the use of components designed for GEO use, and enable a 15 year design life of the PCW system, a significant cost savings when compared to 7-8 year lifetimes of TAP and Molniya.

The TTO orbit parameters are shown in Table 2-3, and are discussed in greater detail in Section 3.2.3.

Table 2-3 TTO Orbit Mean Elements

	Symbol	TTO
Period	T	24 hour
Eccentricity	e	0.3
Inclination	i	90°
Argument of Perigee	ω	270°
Apogee Altitude		48442 km
Perigee Altitude		23144 km

Since the inclination does not cause Eq. (2.1) to approach zero, the TTO will experience drift in the argument of perigee. Bigras et. al. (2) shows that at this inclination, the argument of perigee can vary up to 8.3°/year, and that correcting that drift is within the fuel budget of a typical GEO satellite.

Sources of other perturbations due to the moon, oblateness of the Earth, and even the other planets, will cause other orbital elements to vary with time. Since the other orbit parameters would also require periodic adjustment, regular stationkeeping-like

maneuvers will be needed that can correct for the perigee drift along with changes in inclination, eccentricity and semi-major axis.

2.1.4. Geostationary Orbit

A brief description of the GEO orbit is useful for comparison to the above.

Table 2-4 Geostationary Orbit Mean Elements

	Symbol	GEO
Period	T	24 hour
Eccentricity	i	0
Inclination	e	0°
Argument of Perigee	ω	n/a
Apogee Altitude		35863 km
Perigee Altitude		35863 km

As a satellite in GEO orbits Earth about the equator and at a period that matches Earth's rotation, it maintains the same position in the sky relative to an observer on the ground. This allows for ground and spacecraft equipment to require minimal adjustment pointing over time.

2.2. Orbit Comparisons

Figure 2-1 and Figure 2-2, below, show some of the differences in the orbits. Figure 2-1 shows the ground traces. The TAP orbit (in white) gets its name from the repetition of the three apogees shown. Figure 2-2 depicts the three orbits in the same plane to demonstrate the differences in semi major axis and eccentricity. The red line depicts GEO as seen edge-on from the orbit and is included to show that, at the Equator crossing, the Tundra passes between the Earth and GEO.

Figure 2-2 PCW Candidate Orbits Comparison

As seen in Figure 2-3 and Figure 2-4, the radiation environment of the Tundra orbit is more similar to that of a GEO than either of the other candidate orbits. Both Molniya and TAP orbits cross through the high energy protons environment that Tundra and GEO do not. High energy protons can cause single event upsets, displacement damage and contribute a significant portion of the spacecraft total radiation dose. While TAP was designed to miss the highest density regions of both the trapped electrons and protons, it's important to note also that a satellite in the TAP will pass through that region 3 times per day. A Molniya satellite will pass through four times per day and the Tundra only

twice daily. The TAP provides a more benign environment than Molniya, but not as benign as either GEO or Tundra.

The Tundra orbit is even more benign a radiation environment than GEO because a satellite will only pass through the region, whereas a satellite in GEO remains at the edge of the Van Allen trapped electron belt for its entire operational life.

Figure 2-3 Trapped Electron Environment of PCW Candidate Orbits (AE-8) (Axes in Earth radii)

Figure 2-4 Trapped Proton Environments of PCW Candidate Orbits (AP-8) (Axes in Earth radii)

2.3. Sirius Satellite System

Sirius Satellite Radio made use of a Tundra orbit with their Radiosat 1, 2, and 3 systems. The satellites provide broadcast radio services to North America from a Tundra orbit at the critical inclination. Barker and Stoen describe the challenges associated with the orbit in (3). Similarly to the system investigated by this thesis, the Sirius satellites were based on a commercial GEO satellite design; those spacecraft were built by Space Systems Loral using the LS1300 platform and designed for 15 years of on-orbit service. Radiosat 1 launched in June of 2000 and is still in service, after nearly 14 years.

The following environmental design concerns were identified (3):

1. *Inclined orbit created fundamentally different Sun and moon angle geometry from the geostationary orbit.*
2. *Varying orbit rate throughout the orbit.*
3. *Varying apparent size and view angle of Earth throughout the orbit.*

These will also be challenges to the TTO design. Specifically this thesis addresses the thermal implications of item 1.

The fundamental difference in the Sun-orbit geometry between GEO and Tundra orbits arises from the difference in inclination of the orbits. The Beta angle (β) is the angle between the satellite orbit plane and the Sun vector as shown in Figure 2-5. For GEO satellites, the beta angle is almost exclusively defined by the Sun's declination angle, which varies from -23.5° (South) at the December Solstice to 23.5° at the June Solstice.

Figure 2-5 Beta Angle Definition

Barker and Stoen's calculated beta angles for the Sirius orbit showed that it ranges between $\pm 85^\circ$. (3)

In a GEO satellite, the solar arrays are rotated about one of the spacecraft axes (pitch axis), which is normal to the orbit plane. This rotation accounts for the diurnal rate of rotation of the spacecraft and keeps the arrays pointed towards the Sun, but does not

compensate for the seasonal declination. Thus GEO solar arrays must be sized to accept a $\cos(23.5^\circ)$ loss in power at the solstices.

Simple pitch axis rotation of the solar arrays on a spacecraft in Tundra orbit is not sufficient to keep the arrays facing the Sun at all times. The Sirius satellites require a more complex steering method to ensure that the payload is pointed towards the Earth and that the solar arrays and be pointed normal to the Sun to maximize power generation. The method rotates the spacecraft body about the nadir, or yaw axis in such a way that between the yaw steering and solar array rotation, the solar panels can be rotated to minimize the angle of incidence of the Sun on the arrays.

The yaw rates required by the steering law are maximized at low β angles. Spacecraft hardware limits the rates at which the yaw can be steered, leading to periods where it is not possible to achieve the rates required by yaw steering. Conveniently though, this occurs when the β angle is small and for Sirius, this led to two operating conditions: yaw steering when $|\beta| > 14^\circ$ and normal pointing when $|\beta| < 14^\circ$.

2.4. Applicable Meteorological Spacecraft Designs

2.4.1. GOES-N

GOES-N is a pair of active MET satellites operating in GEO. Built by Boeing, GOES-N design is based on the BSS601 platform, which is generally a commercial GEO communications satellite bus. GOES-N is described by Landecker and Gale (14). GOES-N TCS is shown in Figure 2-10 and Figure 2-11.

Heat is dissipated to space by two main radiator panels (bus and payload) oriented to the north and south. Radiators are covered with OSR and MLI to achieve desired emittance properties. All other exterior surfaces are covered with MLI.

Fixed conductance heat pipes are embedded in the honeycomb panels to spread heat over the radiator panels and transfer heat to the panels from interior equipment. Where necessary to improve the heat spreading for high dissipation units, e.g. Solid State Power Amplifiers (SSPA), these units are bonded to thermal doublers then to the radiator panels.

To facilitate internal radiation heat transfer between equipment, all surfaces are painted black. The batteries and the MET payload are thermally isolated from the rest of the bus and payload modules.

The main meteorological devices, the Imager and the Sounder, require a tightly controlled thermal environment to function optimally. An optical port shield minimizes direct solar heating on the scanning aperture, but there are still times when this method is insufficient, thus there is a louver-controlled radiator to eliminate excess heat from solar radiation.

Figure 2-6 GOES-N Thermal Control Systems Deployed Configuration (14)

The louver is an active thermal control device that opens when the actuators are heated. The opening increases the area of the radiating surfaces that is exposed to space allowing them to dissipate more heat. When the actuators cool, the louvers close and less heat is dissipated.

2.4.1. GOES-R

GOES-R is the planned replacement of GOES-N. Scheduled for launch in 2016, GOES R will consist of two GEO satellites at 75W and 137W to cover North America. The spacecraft is being built by Lockheed Martin, based on their A2100 satellite typically used for GEO communications services.

Figure 2-7 GOES-N Thermal Control Systems Exploded View (14)

2.4.2. Meteorological Payloads – Advanced Baseline Imager

The Advanced Baseline Imager (ABI) is a meteorological payload capable of meeting much of the PCW requirements. The ABI is being developed for use on GOES-R as described above. Though the details of the device operation are scant due to proprietary reasons, some basic information is described by Krimchansky et al. in (15).

The ABI is a multi-spectral radiometer, shown in Figure 2-8 Advanced Baseline Imager , built by Exelis.

Figure 2-8 Advanced Baseline Imager (16)

The ABI has the following physical specifications:

Table 2-5 ABI Physical Specifications

Mass	285	kg
Volume	3.279	m ³
Power	620	W

The advanced focal plane of the ABI imager requires the use of active cooling to maintain its specified performance. The specific thermal requirements were not found in the literature, however there are papers covering the development and testing of the ABI Cryocooler, the device that would keep the focal plane detectors to the very low temperatures required for accurate operations.

Colbert et. al. describe the ABI Cryocooler performance in (17). The ABI cooler system is a High Efficiency Cryocooler (HEC) pulse tube cooler with a remote cold head. The device is capable of:

- 1.9-2.3 W of cooling at 53 K and 162 W input power
- 5.1-8.0 W of cooling at 183 K and 170 W input power

This suggests that the ABI focal plane component requires a thermal environment of 53-183 K for proper operations. The cryocooler has survival and operating temperature ranges shown in the table below.

Table 2-6 ABI Cryocooler Temperature Ranges

	Low [K]	High [K]
Survival	243	333
Operating	258	328

These are slightly more stringent than the values that typical spacecraft electronics require (233-348 K for survival and 253-333 K for operation). Clearly the cold head must be able to survive the 53K environment at which it was load tested, therefore there must be thermal isolation of the cooler's cold head from the rest of the device. At the operating temperatures above, the rest of the device could be maintained at typical spacecraft electronics temperatures.

Table 2-7 Advanced Baseline Imager Spectral Bands (16)

TABLE 1. Summary of the wavelengths, resolution, and sample use and heritage instrument(s) of the ABI bands. The minimum and maximum wavelength range represent the full width at half maximum (FWHM or 50%) points. [The Instantaneous Geometric Field Of View (IGFOV).]					
Future GOES imager (ABI) band	Wavelength range (μm)	Central wavelength (μm)	Nominal subsatellite IGFOV (km)	Sample use	Heritage instrument(s)
1	0.45–0.49	0.47	1	Daytime aerosol over land, coastal water mapping	MODIS
2	0.59–0.69	0.64	0.5	Daytime clouds fog, insolation, winds	Current GOES imager/sounder
3	0.846–0.885	0.865	1	Daytime vegetation/burn scar and aerosol over water, winds	VIIRS, spectrally modified AVHRR
4	1.371–1.386	1.378	2	Daytime cirrus cloud	VIIRS, MODIS
5	1.58–1.64	1.61	1	Daytime cloud-top phase and particle size, snow	VIIRS, spectrally modified AVHRR
6	2.225–2.275	2.25	2	Daytime land/cloud properties, particle size, vegetation, snow	VIIRS, similar to MODIS
7	3.80–4.00	3.90	2	Surface and cloud, fog at night, fire, winds	Current GOES imager
8	5.77–6.6	6.19	2	High-level atmospheric water vapor, winds, rainfall	Current GOES imager
9	6.75–7.15	6.95	2	Midlevel atmospheric water vapor, winds, rainfall	Current GOES sounder
10	7.24–7.44	7.34	2	Lower-level water vapor, winds, and SO ₂	Spectrally modified current GOES sounder
11	8.3–8.7	8.5	2	Total water for stability, cloud phase, dust, SO ₂ rainfall	MAS
12	9.42–9.8	9.61	2	Total ozone, turbulence, and winds	Spectrally modified current sounder
13	10.1–10.6	10.35	2	Surface and cloud	MAS
14	10.8–11.6	11.2	2	Imagery, SST, clouds, rainfall	Current GOES sounder
15	11.8–12.8	12.3	2	Total water, ash, and SST	Current GOES sounder
16	13.0–13.6	13.3	2	Air temperature, cloud heights and amounts	Current GOES sounder/GOES-12+ Imager

The above table gives the instantaneous geometric field of view (spatial resolutions), for the instrument at GEO. Assuming the exact same device characteristics, the spatial resolution of ABI in TTO at apogee would be larger by a factor of 1.35. For example, in GOES Band 2, the GEO-ABI can resolve features 0.5 km and larger, while the TTO-ABI

would only resolve 0.68 km and greater. It's noted that this is the worst case. The resolution would improve as the altitude decreases from apogee. At perigee, TTO-ABI resolution would be maximized at 0.32 km.

1.1.1. Meteorological Payloads – MODIS Imager

Parallels to ABI can be drawn from a predecessor design, the MODerate resolution Imaging Spectrometer (MODIS) device. As seen in Table 2-7 in the previous section, MODIS is considered one of the heritage instruments for ABI. The thermal control of the MODIS instrument is described by Bortfeldt in (18). The instrument has the following general physical properties:

Table 2-8 MODIS Physical Specifications

Mass	235	kg
Volume	2.20	m ³
Power	275	W

Key thermal design features and requirements identified include:

- Wide range of internal operating temperatures (85 K for the focal plane to 393 K for infrared source)
- Thermal isolation from spacecraft $Q < 18$ W.
- White paint on space viewing radiator surfaces of electronics units to dissipate heat
- Reflective aluminum on the space viewing door
- Black paint around other apertures
- MLI covering most other external surfaces

The mainframe structural material is beryllium for the favorable conductivity (k), specific heat capacity (C_p) and particularly for the low coefficient of thermal expansion (CTE). The results of performance predicting models showed an internal operating temperature of 291 K, consistent with the ABI cryocooler operating range.

Part 3: Thermal Design

This chapter describes the development of the model to be used for consideration. The intent is to develop a working thermal model for a typical GEO satellite, including the meteorological payload, and apply that design to the modified Tundra orbit.

This section will explain the thermal environment, the spacecraft and especially thermal design decisions, identify the bounding worst case hot and cold scenarios, and present basic spacecraft thermal balance calculations.

3.1. Spacecraft Reference

It is helpful to start with definition of the panels and axes that will be referred to in the remainder of this thesis. Every attempt is made to match the Tundra nomenclature to the same used for GEO satellites. The notable challenge is in the use of the compass points North, South, East, and West to name the panels. While suitable for GEO satellites that always maintain the same relationship with respect to the Earth's pole, North will only be accurate for a satellite in Tundra for a portion of the time.

Nonetheless the GEO nomenclature for faces and axes will be maintained for the Tundra orbit. They are shown in Figure 3-1, the West face is not shown, but it is logically opposite the East.

Figure 3-1 PCW Spacecraft Nomenclature

3.2. External Thermal Environment

The space environment is dominated by extremely cold temperatures (about 3 K) and by significant heat sources whose effect on spacecraft temperatures depend on location and orientation. While the spacecraft also will be subjected to manufacture, integration, test and launch thermal environments, this thesis focuses on the on-orbit portion of the mission for design and modeling.

3.2.1. Boundary Conditions and Sources of Heat in Space

There are three main sources of heat in Earth orbit: heat generated by the spacecraft, heat from the Sun, and heat from the Earth. Albedo, the reflection of solar heat off of the surface of the Earth, is also significant for some orbits.

3.2.1.1. Spacecraft Generated

The most significant source of heat for a communications satellite is the payload, and specifically the power amplification stages. With typical Traveling Wave Tube Amplifier³ (TWTA) efficiency of 50-60%, a nameplate 100 W TWTA will produce roughly the same amount of heat.

Spacecraft electronics and other components will generate heat as well, although few to the same order of magnitude as the power amplifiers. Other significant components include:

- Output Multiplexers (used to combine multiple channels for feed to a single transmission antenna. Since these signals are already amplified, there is a significant amount of power handled by these devices. Losses will be translated as heat and some OMUX devices experience more than 0.5 dB loss, which is 10% of the input signal power.
- Batteries are needed to provide power when solar arrays are not illuminated. This produces significant heat in short discharge cycles and cooling during charge cycles.

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³ TWTAs are a staple power amplifier for microwave frequency signals both in space and on Earth. The signal travels along a helical structure inside a vacuum. The signal power is amplified through interaction with an electron beam that travels axially along the helix.

- Thrusters generally produce significant heat through combustion of propellants to provide thrust.

The components assumed for this thesis are presented later in this chapter.

3.2.1.2. Solar Flux

The Sun radiates as a black body at 5800 K and produces a heat flux of 1367 W/m^2 (19) (also known as the solar constant) at Earth's average distance from the Sun. Because Earth's orbit is elliptical, the solar flux varies throughout the year. The solar maximum and minimum are:

- Maximum: December solstice, $Q_{\text{Sun}} = 1414 \text{ W/m}^2$ (19)
- Minimum: June solstice, $Q_{\text{Sun}} = 1322 \text{ W/m}^2$ (19)

The radiation spectrum of the Sun is a much shorter wavelength than a spacecraft might radiate, which allows for the use of special thermal finishes which will reflect solar heat, while still radiating spacecraft heat to space.

3.2.1.3. Earth Infrared Radiation

Earth is a source of heat against the cold of space. The heat flux can be approximated by treating Earth as a radiating blackbody at $\sim 255 \text{ K}$. Earth Infrared Radiation (IR) poses an additional challenge to spacecraft thermal designers as compared to solar in that the spectral wavelength of the Earth IR is effectively the same that the spacecraft would radiate at. As such, selective surfaces as described in the previous section do not exist to reject Earth IR while still emitting spacecraft heat.

The value of Earth IR can be calculated or looked up in tables based on orbit parameters. Values range from 218-275 W/m² at Earth's surface. The magnitude of Earth IR flux decreases with increasing altitude. Using Eq. (3.1), with the TTO Perigee altitude of 23144 km, the worst case 275 W/m² represents:

$$\begin{aligned}
 Q_{IR} &= \left(275 \frac{W}{m^2}\right) \left(\frac{R_{earth}}{R_{perigee}}\right)^2 & (3.1) \\
 &= \left(275 \frac{W}{m^2}\right) \left(\frac{6378}{6378 + 23144}\right)^2 \\
 &= 12.8 \frac{W}{m^2}
 \end{aligned}$$

This is a negligible amount compared to incident Sunlight.

3.2.1.4. Albedo

The amount of albedo experienced by a spacecraft is a complex function of the time of year, the relative positions of the Sun, Earth and spacecraft, the spacecraft altitude, and of the surface and atmospheric properties of the Earth at the location off which the Sunlight reflects. Gilmore (19) provides tables to look up representative values of albedo as a percentage of solar flux. The percentage ranges from 18% to 57%. Following a similar calculation as for Earth IR in the previous section, the worst case albedo experienced by a satellite in TTO might be as high as 37.6 W/m², at perigee. Again, this is negligible compared to solar and spacecraft generated.

3.2.1.5. Space Environment

Aside from the heat sources mentioned above, the space environment is a cold vacuum. Without a significant source of convective cooling, the only means for a spacecraft to

dissipate heat is direct radiation to space. The temperature of space is generally considered to be 3-4 K. (19)

3.2.2. Thermal Environment at GEO

Satellites in GEO typically require one face be aimed toward Earth. This nadir pointing keeps antennas and reflectors pointed according to their mission. Over the course of a single orbit (one day) the East, West, Earth and Anti-Earth faces all have a similar progression of orientation with respect to the Sun. The major thermal impacts occur during the solstices, and equinoxes. At the solstices either the North or South face is constantly illuminated. During equinoxes when neither the North nor South face is illuminated (significantly) and the spacecraft enters into daily eclipse cycles (of up to 70 minutes duration).

Figure 3-2 depicts GEO, centered about Earth. As the spacecraft orbits from local noon, it needs to rotate in the pitch axis to maintain accurate pointing. Throughout the course of a single orbit, each of the Earth, Anti-Earth, East and West faces are Sunlit for a portion of the time. Some portion of the time (e.g. local noon for the Anti-Earth face) each of those four faces is in direct Sunlight and will experience the full solar flux. For this reason, the radiators on a GEO spacecraft are typically the North and South faces.

3.2.2.1. Solstice

The solstices are the two events during Earth's orbit when the Sun is at maximum and minimum declination angles. During the December solstice the Sun is 23.5 degrees in the South, thus radiating on the South spacecraft panel. The opposite occurs in the North

during the June solstice. Since the Earth is closer to the Sun on the December solstice, this constitutes the hot worst case season for GEO spacecraft.

Figure 3-2 GEO Thermal Environment and Pointing

3.2.2.2. Equinox

The two annual equinoxes occur when the Sun lies at the intersection of the ecliptic and Earth's equatorial plane. During the equinoxes, the GEO orbit then roughly coincides with the ecliptic plane, thus at spacecraft local midnight, the Sun is on the opposite side of the Earth, causing an eclipse. The sudden removal of solar radiation causes a dramatic change in thermal environment. The longest eclipse during this season is roughly 70 minutes. The season lasts approximately 45 days about the equinox.

3.2.2.3. Earth IR and Albedo

Though both have been shown to be very small values at the Tundra perigee, the sensitive nature of the MET payload detector warrants a comparison of these values. As the ABI is designed for operation at GEO, it is assumed that if the Earth IR and Albedo values are comparable between Tundra and GEO then it is safe to assume that the ABI internal design will be adequate. At GEO, the maximum values of Earth IR and Albedo can be shown to be 6.2 W/m^2 and 18 W/m^2 respectively. These are nearly half of the TTO calculated maximum values of 12.8 W/m^2 and 37.6 W/m^2 occurring at perigee. The ABI is intended to look directly at the Earth, thus both of these sources would enter directly into the ABI's aperture. For this reason and because the cryocooler is only capable of between 1.9 W and 8.0 W of cooling, these differences could be significant in the ABI thermal performance at perigee. Analysis of the problem of the effect of albedo and Earth IR on the ABI thermal performance requires details of the ABI design that are not available.

To summarize, Earth IR and Albedo are negligible to the design at the spacecraft level, and at the detailed level, only the internal ABI thermal environment might be affected, but not enough is known about that design to model it accurately.

3.2.3. Thermal Environment at TTO

Where the GEO orbit is in more or less the same orientation with respect to the Sun throughout a year, a 90 degree inclined orbit is in very different orientations with respect to the Sun, depending on the time of year and the orientation of the orbit plane in space. It creates a significant challenge to the mission designers to maintain proper payload

pointing and still have enough control to keep the solar arrays pointed towards the Sun to provide adequate power margin.

In the following analysis it is assumed that the spacecraft has sufficient control authority to maintain perfect Sun-nadir steering. This involves two main constraints:

1. The East face always faces the Sun
2. The Nadir face always faces the Earth

Margins will be added into the design that will account for the real world possibility that the constraints cannot be met as ideally as they can by software.

The variation of the TTO orbit orientations gives rise to two pointing modes and depending on orbit design, these “pointing seasons” could occur during any of the traditional seasons, which has significant impact on the thermal environment that PCW might experience.

Four principal plane orientations in space considered are shown in Figure 3-3 through Figure 3-6. As the figures show, the orbit plane geometry of the TTO orbit varies, depending on the value of the Right Ascension of the Ascending Node (RAAN).

Figure 3-3 Tundra Orbit Orientation, RAAN = 0°

Figure 3-4 Tundra Orbit Orientation, RAAN = 45°

Figure 3-5 Tundra Orbit Orientation, RAAN = 90°

Figure 3-6 Tundra Orbit Orientation, RAAN = 135°

3.2.3.1. Sun-Nadir Steering Season

For the majority of the time in orbit, PCW in TTO will require a complex steering law in order to maintain the payload's orientation to the Earth and keep the solar arrays positioned such that the array drives can keep the arrays towards the Sun. Sun-Nadir steering is used on the Sirius satellites for the classic Tundra orbit and is difficult to conceptualize. Figure 3-7 describes the steering law as it applies to a non-zero beta angle orbit as shown in Figure 3-4. Figure 3-8 shows the attitude angles required by the Sun nadir steering law.

Figure 3-7 Sun-Nadir Steering Beta Angle Non Zero

Figure 3-8 Sun-Nadir Steering RPY for Beta=45°

Figure 3-8 shows the attitude angles through a single orbit when the Beta angle is 45°. The Pitch angle increases through the orbit at rates that match the changing orbit velocity due to the eccentricity. The Roll and Yaw angles oscillate to achieve Sun-nadir steering.

3.2.3.2. Eclipse Season

In the cases where the Sun lies in the orbit plane (e.g. during March and September, when $RAAN = 0^\circ$), the orbit is analogous to the environment during GEO equinox, in that the spacecraft solar array drives alone can keep the solar arrays aligned properly by counteracting the pitch rotation of the spacecraft. The Earth, Anti-Earth, East and West faces all experience periodic Sunlight and shadowing through the course of an orbit, though the eccentricity of the orbit will cause the amount of time each face is in the Sun to be different. The North and South faces are almost completely free of Sunlight.

Most importantly, however is that the orbit passes through the Earth's shadow causing periods of eclipse.

From Figure 3-9, we have a very similar situation to GEO, just with a 90° inclination. Figure 3-10 shows no change in the satellite yaw rate over a single orbit. The pitch rate varies in magnitude given the orbit's eccentricity, but the direction is constant. The variation in roll angle shown in the plot is due to the fact that the orbit plane would be tilted from how it appears in the figure.

Figure 3-9 Sun-Nadir Steering Beta Angle Zero

The thermal implications of the TTO eclipse season are:

- Sunlight is distributed over all four non-radiating faces in a diurnal cycle.

- When satellite is near opposition, the ABI focus will be pointed in the general direction of the Sun.
- The spacecraft will enter Earth's shadow somewhere along this orbit depending on the season. Eclipses in the TTO are dealt with in the following section.

Figure 3-10 Roll Pitch Yaw Plot for TTO Eclipse Season

3.2.3.3. Eclipse

STK was used to model theoretical PCW missions for the four RAAN scenarios from above. The missions were assumed to start at January 1, 2017, and analyzed over a 15 year life. The following maximum eclipses were found for each of the RAAN cases for a two satellite system.

These eclipse times include the spacecraft time in both umbra and penumbra. It is important to note that most of the longest eclipses take place when the Sun is in the South

Thus at the time of eclipse, the spacecraft will be over the North, and these eclipses will occur when the spacecraft needs to be operational.

If the communications payload is operational when the satellite is north of the equator and non-operational in the South, then from Table 3-1, we see that, though the spacecraft is experiencing the longest eclipse in the December case, the communications payload

Table 3-1 TTO Longest Eclipse Comparisons

Satellite	Duration [s]	Duration [min]	Start Date	Start Time
RAAN = 0°				
PCW1	5387.88	89.8	9/23/2031	10:20:00
PCW2	5396.26	89.9	9/21/2025	19:37:55
RAAN = 45°				
PCW1	5737.45	95.6	11/10/2031	12:10:02
PCW2	5728.27	95.5	11/10/2031	00:09:08
RAAN = 90°				
PCW1	5818.20	97.0	12/22/2031	12:56:33
PCW2	5814.07	96.9	12/22/2031	00:56:33
RAAN = 135°				
PCW1	5681.27	94.7	02/01/2031	11:47:40
PCW2	5688.15	94.8	02/01/2031	23:46:42

is still fully operational. The heat generated by the communications equipment is greater than the amount of solar incident on the spacecraft. Similarly, during the March/September eclipse (Figure 3-11) the communications would be operational for at least a portion of the eclipse. Whereas June eclipses (Figure 3-13) occur while the spacecraft is in a portion of the orbit where communications may be in limited use, or not used at all.

Figure 3-11 March/September Eclipse (RAAN = 0°)

Figure 3-12 December Eclipse (RAAN = 90°)

Figure 3-13 June Eclipse (RAAN = 90°)

3.3. Typical GEO Thermal Design

The dominant feature of a GEO communications satellite TCS design are the radiating panels on the North and South faces. These panels are the mounting sites for the communications equipment that generate high heat loads, such as travelling wave tube amplifiers and signal multiplexers. The radiators are created by covering the surface of the structural panels with coating that is highly reflective in the solar spectrum, and highly emissive in the IR spectrum, such as Optical Solar Reflectors (OSR). With high IR emissivity, and low solar absorptivity, OSR rejects the majority of the solar flux while still radiating to space.

Radiator panels are usually constructed with embedded heat pipes to isothermalize the panel, and to provide the mounted electronics with a direct, high conductivity path to the radiating surface. Figure 3-14 shows a schematic of a typical GEO satellite radiator

panel, where: Q_{int} is the heat generated internally by the device, A is the radiating area of the panel, α is the absorptivity, ϵ is the emissivity, σ is the Stefan-Boltzmann constant, and T is the panel temperature.

Figure 3-14 Radiator Panel Schematic (19)

The four other faces experience a diurnal variation in solar incidence, and thus experience significant periods of full Sunlight each day. These faces are insulated from the Sun and space with Multi-Layer Insulation (MLI), to minimize the flux through these faces. The MLI surface coating is variable depending on design, generally either black or gold Kapton to maximize IR emissivity at the surface.

Internally, most components require operating temperatures that span about room temperature ranges. To minimize temperature gradients within the spacecraft cavity, surfaces are painted black to maximize radiative heat transfer between components.

3.4. Thermal Requirements and Constraints

This section describes the assumed subsystems and components for PCW, as they pertain to the thermal design. From these assumptions, a set of thermal constraints and requirements is extracted.

3.4.1. Spacecraft Subsystems Assumptions

This section describes the assumed subsystems and components for PCW. The Basic design (structure) will be based on the Space Systems Loral 1300 series platform shown in Figure 3-15.

3.4.1.1. Spacecraft Mass Assumptions and Allocations

Table 3-2 shows the average subsystem mass budgets for GEO satellites, without fuel. PCW should be expected to be comparable to these values.

Table 3-2 Average Communications Satellite Mass Budget (7)

	Average
Payload	27%
Structure	21%
Thermal	4%
Power	32%
TTC	5%
ADCS	7%
Propulsion	4%
Total	100%

For example, if PCW spacecraft dry mass is assumed to be 2000 kg, then the thermal systems should total about 80 kg.

1300 Satellite: Modular Design for Efficient Adaptation

Figure 3-15 SSL LS1300 Spacecraft Bus Diagram (20)

3.4.1.2. Communications Payload

The Communications payload shall be based on (21) (1) (22) and is assumed to consist of:

- 10 UHF transponders (12:10⁴ SSPA at 90W each, 292 MHz – 380 MHz)
- 18 Ka-Band Channels (20:18 TWTA at 100W each, 19.7GHz – 31.0 GHz)

⁴ The nomenclature shown is an industry standard for redundancy. In this use it can be read “12 for 10” meaning a total of 12 units for 10 active units required. The difference in the two numbers is the number of spare units onboard.

- 6 X-Band Channels (8:6 at 100W each, 7.25 GHz – 8.4 GHz)

Among many smaller components, a complete transponder in the Ka and X bands (single channel) consists of the receive antenna, receiver (Rx), low noise attenuator (LNA) power amplification (TWTA), power conditioning (EPC), attenuation (LCAMP), signal separation (IMUX), signal recombination (OMUX), transmit antenna, and all the waveguide and cabling required to make the connections. The EPC and TWTA are based on those manufactured by L-3 Communications (23), and the remaining components based on the product line from Thales (24). As will be shown in Section 3.7.5, the TWTAs will need to have integrated radiation cooling fins.

It is assumed that all the communications antennas are deployable, as they need to fit into the launch vehicle fairing. As specific antenna design is beyond the scope of this thesis, two frequency independent circular reflectors will be used to characterize the thermal environment exterior to the spacecraft. They will be 2.5 m diameter, shaped graphite, based on a design by Vanguard (25) with emissivity of 0.92 and absorptivity of 0.96. Operating and survival temperatures are summarized for all equipment in Table 3-12.

Figure 3-16 UHF Antennas on Skynet 5d Spacecraft (26)

Table 3-3 Communications Payload Equipment Specifications

Name	Power [W]	Units	Active	Unit Mass [kg]	Total Mass [kg]	Total Power [W]	Unit Diss. [W]	Total Diss. [W]
UHF								
SSPA	150.0	12	10	1.2	14.4	1500.0	60.0	600.0
LNA	12.6	2	2	3.2	6.4	21.0	12.6	21.0
OMUX	0.0	6	4	1.6	1.6	0.0	115.9	231.8
Switches	2.0	12	1	0.3	3.9	2.0	5.0	50.3
Rx	21.0	12	-	2.0	10.0	63.0	21.0	63.0
Ka								
TWTA	181.8	20	18	1.5	29.9	3272.7	81.8	1472.7
LCAMP	3.2	20	18	0.3	5.8	57.6	53.2	57.6
LNA	12.6	5	3	3.2	17.1	42.0	12.6	42.0
EPC	21.8	10	9	1.8	17.5	196.4	21.8	196.4
OMUX	0.0	6	4	7.2	7.2	0.0	231.8	463.5
Rx	21.0	5	3	2.0	10.0	63.0	21.0	63.0
Switches	2.0	24	1	0.3	7.8	2.0	5.6	100.7
X								
TWTA	181.8	10	8	1.0	9.7	1454.6	81.8	654.6
LCAMP	3.3	10	8	0.2	1.9	26.4	3.3	26.4
LNA	12.6	3	2	3.2	9.6	25.2	12.6	25.2
EPC	21.8	5	4	1.8	8.8	87.3	21.8	87.3
OMUX	0.0	3	2	0.0	0.0	0.0	165.7	206.0
Rx	21.0	5	3	2.0	10.0	63.0	21.0	63.0
Switches	2.0	12	1	0.3	3.9	2.0	5.6	44.8
Harness	-	-	-	-	50.0	-	-	-
X/Ka 1	-	2	2	-	20.0	-	-	-
UHF Rx/Tx	-	2	2	-	30.0	-	-	-
X/Ka Ant. drives	-	2	2	15.0	30.0	-	-	-
Totals					344.9	6872.1		4418.8

The UHF antennas would be helical antennas, similar to those pictured in Figure 3-16. It is assumed that two such rigid antennas, of 4 m length, will be required, one for transmit and one for receive.

3.4.1.3. Meteorological Payload

As per Garand and Morris (1), the MET payload instrument should be compatible with either the GOES-R or the Meteosat Third Generation (MTG) Geosynchronous satellite systems. The analysis in this thesis will assume that PCW will carry the Advanced Baseline Imager (ABI) from the GOES-R System. Due to thermal restrictions on the IR detectors, the ABI should be as thermally isolated as possible from the rest of the spacecraft. Table 2.6 showing ABI specifications is repeated below for convenience.

Table 3-4 ABI Physical Specifications

Mass	285	kg
Volume	3.279	m ³
Power	620	W

3.4.1.4. Structure

The Proton rocket launched the Sirius satellites into the classic Tundra orbit, and could be suitable for the TTO as well. The Proton Breeze M PLF-BR-15255 fairing has 9.145 m of clearance, before the nose cone tapers off (27). The fairing diameter is 4.6 m, so the spacecraft bus must be smaller still to allow room for external appendages, i.e. stowed solar arrays and stowed reflectors. The bus will be assumed to be 2.5 m x 2.5 m x 3.5 m. The ABI instrument would be installed on top of that, bringing the total height of the bus and ABI to about 5.5 m. UHF antennas would then extend to a height of 7.5 m.

Structural panels will be 4 cm thick Aluminum honeycomb, with heat pipes embedded in the two radiator panels for better thermal distribution. The central cylinder shown in Figure 3-15 is the main support for the internal electronics deck and the propellant tanks, and is assumed to be the same Aluminum honeycomb. Assuming the panels without heat pipes or MLI have a density of 624 kg/m^3 , then the structural mass is shown in Table 3-5.

Table 3-5 Structure Exterior Surface Area and Mass

	Area (m ²)	Mass [kg]
North Panel	8.89	79.65
South Panel	8.89	79.65
Earth Panel	6.45	57.81
Anti Earth Panel	6.45	57.81
East Sub Panel	6.25	56.00
West Sub Panel	6.25	56.00
EastBatteryPanel	2.75	24.64
WestBatteryPanel	2.75	24.64
Cylinder	n/a	13.55
Mid Panel Deck	n/a	56.00
Structure Total [kg]		386.92

3.4.1.5. Propulsion

A standard chemical bipropellant propulsion system consisting of fuel and oxidizer, the tanks required to hold the propellants, thrusters and the lines, valves and actuators required for propellant transport, will be assumed for PCW. The power requirements are low, and the technology very robust. This would consist of 12 thrusters to provide redundant capability for translational thrust in three directions, as well rotational torque about each of the three principal axes.

Astrium's 22 N thruster (28) has flown on several missions, and has the following relevant specifications:

- Mass: 0.65 kg
- Propellant: Monomethyl Hydrazine (MMH)
- Oxidizer: N_2O_4
- Platinum alloy nozzle operational up to 1500 °C
- Isp: 290 s

The bipropellant would be stored in separate but nearly identical tanks attached to the central cylinder. Propellant tank specifications for typical GEO satellites:

- Mass: 16 – 29 kg
- Volume: 235 – 516 L
- Diameter: 0.600 – 1.070 m
- Height: 680 – 1150 mm
- Material: Ti6Al4V, Titanium alloy

The fuel and oxidizer they carry have more stringent thermal requirements than the tanks. MMH freezes at -52 °C and boils at 87 °C. N_2O_4 is more restrictive, being a liquid between -11 °C and 21 °C. This range is similar, albeit slightly cooler than, the operating ranges for electronics given above. A similar MMH/ N_2O_4 system is used on GOES-R.

Table 3-6 Propulsion System Equipment Specifications

	# Units	Unit Mass [kg]	Total Mass [kg]
22 N Thruster	12	6.5	78.0
Fuel Tank	1	29.0	29.0
Oxidizer Tank	1	29.0	29.0
Lines	-	10.0	10.0
Propulsion Total			146.0

The thruster dissipation is not included in this table, or in the thermal budget. Thrusters will be located on the exterior of the spacecraft, and would thus radiate directly to space.

It is assumed that they are sufficiently insulated from the spacecraft, and of such short duration of use as to have small effect on spacecraft body temperatures.

3.4.1.6. Attitude Determination and Control

The MET payload requires precise position and orientation knowledge, requiring star trackers. Star tracker capability more than adequately covers the payload and TTC pointing knowledge requirements. The sensors selected for this design are Goodrich HD-1003 Star Tracker, (29).

Attitude rate sensors will be needed to feed into the Sun-nadir steering algorithms. Honeywell's DG1320 Ring Laser Gyros (RLG) (30) will provide rate measurements. Actuators will be the HR0160 Reaction Wheels (RW) from Honeywell (31) in addition to the above mentioned thrusters. The relevant ADCS equipment specifications are given in Table 3-7.

Table 3-7 ADCS Equipment Specifications

	Units	Active	Unit Mass [kg]	Total Mass [kg]	Power [W]	Total Power [W]
Star Trackers	4	2	3.9	15.4	10.0	20.0
RW ⁵	4	4	12.0	48	195.0	780.0
RLG	4	4	0.5	1.8	1.6	6.4
Electronics	2	1	50	100.0	100.0	100.0
ADCS Total				65.2		906.4

---

⁵ Power given is at maximum demand. The actual demand will be transient and variable throughout the orbit. The thermal and power subsystems should consider the worst case of full demand across all 4 RW.

3.4.1.7. Power

The main source of spacecraft power is from solar arrays. Stowed alongside the spacecraft during launch and deployed once on orbit, the solar arrays of spacecraft are critical systems.

During eclipses, solar power is not available. However, the spacecraft will still need to operate during these times. Battery power will be required to span these times that solar power is unavailable.

Given the above design specifications the total power required for the spacecraft is 8708 W. Proper solar array sizing must take into account EOL conditions, possible failures, and any other likely degradation of generating capability, such as micrometeoroid deposit on the solar cell surfaces and radiation damage. This loss of capability will be accounted for in solar array sizing.

Lithium ion batteries are the emerging state of the art for satellite batteries, with a greater power to mass ratio than the previous generations. The power provided remains fairly stable over life, losing less than 10% capability over 10 years (32). Lithium ion batteries will be assumed for PCW.

A power budget is presented in Section 3.5. The major power subsystem equipment specifications are developed in the following sections.

3.4.1.7.1. Battery Sizing

Batteries need to provide power throughout the deepest eclipse, and at EOL, after potential loss of capability. From Table 3-1, we know that the deepest eclipse is 97 min, during which a total of 14550 W-h will need to be provided to the spacecraft.

The Eagle Picher 10197 Lithium Ion Battery (33) can deliver 3300 W-h at BOL. Four such batteries would provide 13200 W-h, not quite enough to cover the required power. Assuming 10% degradation over life, and an 80% maximum depth of discharge, either six such batteries, or batteries capable of 4041 W-h are required to meet this demand.

Larger batteries could be used, however four batteries allow for a better distribution of mass within the satellite, and reduces risk in the event of a battery failure.

The batteries have the given specific energy of 94.6 W-h/kg. Assuming the manufacturer can create a 4950 W-h battery with the same specific energy, this would result in an individual battery mass of 52.3 kg. The operating temperature range is -5 °C to +35 °C and survival range is -15 °C to +40 °C.

3.4.1.7.2. Solar Array Sizing

In the worst case, the solar array must provide enough power for full payload and bus operation, and battery recharging during the summer months when incident solar is at its lowest. A 10% margin will be added to account for circuit failures.

Total power requirement for bus and payload as estimated is 8708 W. Additional capacity is required to recharge the batteries following eclipses.

It is assumed that the batteries should be fully recharged within 16 hours of the end of discharge. This gives close to 6.5 hours of margin before the start of the next eclipse.

The additional power, P , required by the arrays is calculated using Eq. (3.2).

$$P = \frac{14550 \text{ W} \cdot h}{16 \text{ h}} \quad (3.2)$$

$$= 909.4 \text{ W}$$

The total power required is then:

$$\begin{aligned} P_{total} &= (909 \text{ W} + 8708 \text{ W}) \cdot 110\% \\ &= 10579 \text{ W} \end{aligned} \tag{3.3}$$

The following further assumptions are made in sizing the arrays:

- Power design point is 11000 W to encompass 10579 W required.
- Solar Cells: Triple Junction GaAs cells at 29.5% BOL efficiency in converting solar to electrical power.
  - Though Solar cell efficiency is a function of temperature, this calculation is intended to find the maximum solar array area, thus the efficiency is taken at the reference temperature of 20 °C.
- Degradation and debris deposit causes 25% loss in capability to EOL.
- Solar minimum 1322 W/m².
- Solar panel width will be just slightly less width than the spacecraft body: 2.4 m.
- 2 Solar array wings with the same number of panels per wing.
- There will be limitations on how accurately the arrays can be pointed. In GEO the arrays can be pointed off of the Sun by as much as 23.5°. Though Sun-Nadir steering is supposed to keep this angle near zero, a GEO-like margin for off-pointing is factored into the design.
- The entire face of a panel is covered in solar cells, i.e. spacing between individual cells is negligible.

With these assumptions we can calculate the area required to generate the spacecraft power.

$$A = \frac{11000 \text{ W}}{\left(1322 \frac{\text{W}}{\text{m}^2}\right) (0.75)(0.295)(\cos (23.5^\circ))} \quad (3.4)$$

$$= 43.7 \text{ m}^2$$

For 6 panels, each one could be 3.03 m long, which is again smaller than the panel against which the array will need to be stowed.

Assuming a specific power of the solar array of 28.7 W/kg (19), then the total mass of the arrays is 383.2 kg.

Together with the battery mass and adding 10% for harnessing, the total power subsystem mass budget is 651.7 kg, as shown in Table 3-8.

Table 3-8 Power Subsystem Mass

	Units	Unit Mass [kg]	Total Mass [kg]
Batteries	4	52.3	209.3
Solar Arrays	6	68.9	383.2
Structure, Mechanisms and Harnessing	-	59.2	59.2
Power System Total			651.7

3.5. Mass and Power Budget

The mass budget is given in Table 3-9. The right-most column shows the percentage difference between the PCW design and the average values provided earlier in Table 3-2. There is a fairly strong correlation with the industry averages given once a 10% margin for error has been added in. The payload is considerably higher than average as may be expected with the unique combination of payloads.

The power budget in Table 3-10 also shows positive margins for both battery and solar array power.

Table 3-9 Mass Budget

	Avg. GEO	Mass [kg]	+10% Margin	% of total	Delta
Payload	27%	625.9	688.5	32%	4.8%
Structure	21%	386.9	425.6	20%	-1.4%
Thermal	4%	46.4	51.0	2%	-1.2%
Power	32%	651.7	716.9	33%	1.6%
TTC	5%	23.8	26.2	1%	-3.6%
ADCS	7%	165.3	181.8	8%	1.6%
Propulsion	4%	146.00	160.6	8%	3.7%
Total	100%	1945.9	2140.5	100%	0.3%

Table 3-10 Power Budget

Solar Array	
Area	43.67 m ²
Efficiency (EOL)	0.22
Solar Min.	1322 W/ m ²
Capability	12702 W
With 10% loss	11432 W
EOL Margin – June Solstice	27.0%
EOL Margin – Recharge Ops	10.1%

Battery (4)	
Eclipse Power Required	14550 W
Rated Power	4950 W
Less 10% degradation	4455 W
DOD Limit	80%
Max DOD, BOL	71.1%
Max DOD, EOL	79.0%

3.6. Thermal Systems

Thermal control components come in two main categories, active and passive. Passive techniques are preferred, due to lower power and mass, on average. GEO satellites use the following main thermal control components:

- *Radiators*: A surface with high emissivity that is exposed to space and allows heat to be radiated. A variety of surface finishes, are available including highly reflective, Optical Surface Reflectors (OSR) or Second Surface Mirrors (SSM) with high emissivity ($\epsilon = 0.80$) and low absorptivity ($\alpha = 0.08$). OSR are applied to radiating panels to reflect solar flux and still radiate to space.
- *Insulation*: Multi-layer insulation (MLI) is used on internal and external components to isolate environments. Different surface finishes (e.g. black or gold Kapton) will further affect the flux through the insulation. All MLI on the model will be assumed to have an effective conductivity, $k_{\text{eff}} = 5 \times 10^{-5} \text{ W/m}\cdot\text{K}$. Surface properties of the various surface finishes used in this design are in Table 3-11. Using a constant effective conductivity is a simplification, as the real conductivity is a function of the temperature difference between the two sides of the MLI. Large gradients will lead to larger effective conductivity and thus greater flux across the MLI.
- *Heat Pipes*: A working fluid cycles through evaporation and condensation cycles within a wicked pipe that allow high heat loads to be transported. Heat pipes are used to transport heat from high dissipation components to radiators, and are embedded within radiating panels to isothermalize the panel to improve radiation capability.
- *Heaters*: Generally, spacecraft heaters are conductive wires that can be wrapped and arranged in a variety of configurations to suit the particular needs. The wires carry a current and dissipate heat as a result. Heaters can be controlled by autonomous software or by explicit command as required.

- *Thermal Contacts:* A variety of interface materials are available, either to resist or facilitate heat transfer through the interface. For Communications equipment mounted directly to radiator panels a material with high conductivity ($k = 2\text{-}5 \text{ W/m}\cdot\text{K}$) will be used to overcome thermal contact resistances.

Table 3-11 Surface Finish Properties

	α_{BOL}	α_{EOL}	ϵ_{BOL}	ϵ_{EOL}
OSR, 8-mil quartz	0.05	0.22	0.80	0.80
Z-306 Black Paint	0.95	0.92	0.87	0.87
Z-93 White Paint	0.17	0.55	0.92	0.92
Black Kapton	0.92	0.89	0.88	0.88
Gold Kapton	0.22	0.27	0.81	0.81

The surface properties shown above are typical values and considered to be independent of direction of incident flux. The absorptivity values given reflect performance in typical solar wavelengths, while the emissivities are in IR wavelengths. Both values are subject to variation in wavelength, but are assumed constant for the model.

3.6.1. System Level Thermal Requirements

Electrical equipment requires a benign thermal environment. Each component in a spacecraft is generally described by providing two ranges of temperatures: Operating and Survival. Operating refers to the allowable temperature range of a component during operations. The Survival range defines the non-operational limits, and the TCS must be designed to control the thermal environment to within those ranges at all times. It is typical to design to an even more stringent margin as a safeguard against anomalous operating conditions, giving rise to the target range shown in Table 3-12.

Survival limits dominate in the non-operational cases, namely during launch and orbit-raising and in the event of loss of pointing control or payload shut down contingencies. The survival cases for GEO communications satellites have been well exercised in history with dozens of similar designs launching each year. This thesis concentrates on the novel conditions experienced in the TTO orbit.

Typical operating temperatures are given in Table 3-12 below. A 10 degree margin is added to develop a target temperature. The TCS design should maintain all components to within the temperatures defined by the Target columns. Since it is most cost effective to develop and test equipment in ambient room temperature conditions, it is not surprising that all components operate in the 10-20 °C range.

From the Table 3-12, if the spacecraft operates between 0-20 °C, all target temperatures can be met.

3.7. Worst Case Definitions

Common thermal design practice is to examine the TCS performance in the extreme hot and extreme cold cases. It is a safe (and indeed proven) design assumption that if a TCS design can achieve desired temperatures in the extreme cases that it will also be adequate in the less stringent cases in between.

While the period, inclination, and eccentricity have been defined, a specific orbit plane orientation (characterized by the RAAN for this thesis) for PCW hasn't been decided. So this analysis will assume that, given the TTO period, inclination, and eccentricity, the design will have to account for the worst possible thermal cases of RAAN selection.

Other factors that affect the worst cases are operations and component degradation.

Table 3-12 PCW Component Operating Temperatures (in °C)

	Survival Low	Op. Low	Target Low	Target High	Op. High	Survival High
ADCS						
Star Trackers	-30	-20	-10	45	55	65
RW		-30	-20	60	70	
RLG	-54	-54	0	40	83	94

Electronics (CDH)						
Electronics	-	-10	0	40	50	-

Communications						
SSPA (UHF)	-54	-40	-30	60	70	105
Receivers	-	-20	-10	55	65	-
TWTA (X/Ka)	-	0	10	65	75	-
LCAMP	-	-15	-5	55	65	-
LNA	-	-15	-5	55	65	-
EPC	-	-15	-5	55	65	-
Switches	-	-54	-44	74	84	-
Ka/X Reflectors	-160	-150	-150	85	85	95
UHF Helices	-120	-100	-90	90	100	120

MET - ABI						
IR Detectors	-220	-210	-210	-200	-200	-190
Cryocooler	-30	-15	-5	35	45	60
Electronics	-30	-15	-5	45	55	60

Power						
Batteries	-15	-5	0	25	35	40
PCU	-	-20	-10	45	55	-
Solar Arrays	-200	-105	-95	100	110	130

Propulsion						
MMH Tank	-52	-52	-52	87	87	87
N ₂ O ₄ Tank	-11	-11	-11	21	21	21

Thermal						
OSR Tile	-	-95	-85	50	60	-
Heater	-	-35	-25	50	60	-
MLI	-	-160	-150	240	250	-

Structure						
Honeycomb Panels	-	-45	-35	55	65	-

3.7.1. Operating Modes

The primary PCW mission is the North Polar Region as identified previously. When the Arctic is not in the satellite's view it is likely that the communications payload will need to be switched to a standby mode to avoid interference with other satellite communications missions. The MET payload would not interfere with any other mission, and could in fact supplement current GEO MET satellites by providing independent data for correlation and calibration. Furthermore, the period of time the PCW satellites spend over the Antarctic could provide useful weather data over that area. For this reason, it will be assumed that the MET payload will be in operation over the full 24 hour orbit. Communications in Antarctica are limited, and PCW could provide additional capability there, though a full payload is not likely to be utilized by the limited potential users there.

Table 3-13 PCW Operating Modes

Payload	Arctic in view	No polar regions in view	Antarctic in view
Comms	Operational	Not-Operational	Semi-Operational
Met	Operational	Operational	Operational
Bus Electronics	Operational	Operational	Operational

To bound the hot and cold worst cases, the hot case shall consider full operations at all times throughout the orbit, while the cold cases will consider no communications operations from reaching the descending node through perigee. Full communications operations will resume when the spacecraft reaches the ascending node.

3.7.2. Lifetime Effects on Thermal Control

A spacecraft is exposed to a number of factors that lead to degradation of surface properties of thermal control coatings over life such as micrometeoroids and radiation. The effect of these is that the emissivity of surface coatings is generally degraded and absorptivity increased, described in (19).

Thermal designs must meet requirements at spacecraft EOL. Spacecraft surface absorptivity values will increase to end of life, due to degradation from charged particle and ultraviolet radiation, from contaminant deposit from outgassing materials and possible damage from collisions with micrometeoroids and space debris. Design for EOL leads to BOL absorptivity values that will be lower than required at that time. Emissivity values tend to change very little over spacecraft life. As a result of the increased absorptivity at EOL driving the design, a spacecraft will run then ‘colder’ at BOL. This is typically not a major design constraint, as heaters can be sized and placed to compensate for the additional heat rejection at BOL.

3.7.3. Worst Case Cold

The coldest case with respect to solar flux occurs when the spacecraft is furthest from the Sun. The minimum solar flux is always near the June solstice for any Earth orbiting spacecraft.

Operationally, the cold case occurs when the communications payloads are powered down, i.e. when the Arctic is not in view of the spacecraft. We have assumed that the communications payload will be in full operation when the Arctic is in view (north of the nodes) and in some limited operations when it is not.

It was also previously discussed that BOL conditions are typically ‘colder’ than EOL. Summarizing the above, the worst case cold conditions are then:

- BOL
- June Solstice
- Comms Payload Powered Down (from Table 3-13)

3.7.4. Worst Case Hot

The worst hot case is of somewhat greater significance to the thermal design than the cold case. If a spacecraft design is too cold, heaters can be added or increased in power to compensate with minimal design impact, whereas if the design does not meet hot case limits, a redesign of radiator surfaces may be required. A significant increase in radiator area could have significant impact on the structure, mass budget and ultimately the fit in the payload fairing.

The maximum solar flux occurs near the December solstice. The maximum operating load occurs when all the payloads are on and operational. EOL conditions on surface finishes maximize absorptivity.

Thus the hot worst case has the following conditions

- EOL
- December Solstice
- All Payloads operating

3.7.5. Initial Thermal Balance

The first step is to make a rough estimate of the thermal environment and gain a starting point estimate for the spacecraft temperature. Since the heat emitted must be equal to the heat received and generated by all sources, the thermal balance for the Tundra case is

$$\dot{Q}_{emit} = \dot{Q}_{solar} + \dot{Q}_{gen} \quad (3.5)$$

$$\dot{Q}_{emit} = \dot{Q}_{South} + \dot{Q}_{North} + \dot{Q}_{ABI,radiator}$$

$$\dot{Q}_{gen} = \dot{Q}_{gen,bus} + \dot{Q}_{gen,ABI}$$

The following assumptions are made:

- The temperature of deep space is 0 K. A value of 3-4 K can be used but because it is used in radiation calculations where the temperature is raised to a power of 4, the effect of 3-4 K compared to spacecraft temperatures in the 273-300 K range is very small.
- Heat flux through MLI is effectively zero.
- Only the radiator panels radiate to space and both panels are at the same temperature.
- In typical operation, there would be no incident Sunlight on the radiators. However, to account for tolerance in the Sun-nadir steering capability, a maximum solar radiation angle on the panels of 30° is assumed.
- Earth Albedo and IR are negligible. (See Section 3.2.1.3 and 3.2.1.4 for a justification.)
- ABI is thermally isolated from spacecraft.

Isolating the ABI from the satellite leads to two independent balance calculations, one for the bus and one for the ABI.

Starting with the ABI instrument, if the radiator panel has an area of 3.28 m² and the same radiator thermal parameters as the spacecraft the ABI balance is then:

$$650 \text{ W} = A_{ABI \text{ panel}} * \epsilon_{ABI \text{ panel}} * \sigma * T_{ABI \text{ panel}}^4 \quad (3.6)$$

$$T_{ABI \text{ panel}}^4 = 6503.28 * 0.80 * 5.67 \times 10^{-8}$$

$$T_{panel} = 256.9 \text{ K}$$

Calculating the balance will determine a surface temperature for the radiator. The total dissipation calculated is 5875 W for bus and payload. Including a 10% margin for error gives 6463 W total dissipation required by the radiator panels without including the TWTA radiators. With emissivity equal to 0.8 for OSR, and radiator panel area of 8.89 m² we get:

$$6463 \text{ W} = 2 * A_{panel} * \epsilon_{panel} * \sigma * T_{panel}^4 \quad (3.7)$$

$$T_{panel}^4 = \frac{6463}{2 * 8.89 \text{ m}^2 * 0.80 * 5.67 \times 10^{-8} \text{ W} \cdot \text{m}^{-2} \cdot \text{K}^{-4}}$$

$$T_{panel} = 299 \text{ K}$$

This is somewhat higher than the target spacecraft operating temperatures in Table 3-12, hence the need for TWTAs that radiate directly to space.

Accounting for the radiating fins will reduce this. The radiator fins are modeled as 8 small plates extending from a 25 mm diameter core. The plates and core are 150 mm long and extend to an outer diameter of 150 mm. The total area is approximated by adding up the surface area of the core, the face area of the 8 plates and the area of the back plate.

$$A = (0.15 * \pi * 0.025^2) + (8 * 0.075 * 0.15) + (\pi * 0.15^2) \quad (3.8)$$

$$A = 0.1612 \text{ m}^2$$

The TWTA radiator fins will protrude from the radiator panels towards the East and West. Those on the East side will be illuminated by the Sun, almost end on. Taking the bluff area of the end of the TWTA fins as the illuminated area ($\pi * (0.075\text{m})^2 = 0.0177\text{m}^2$) and using the EOL absorptivity and emissivity values for Z93 White Paint ($\alpha = 0.55$; $\varepsilon=0.92$) from Table 3-11 we can perform a thermal balance on the TWTA fins.

Further assuming that the radiator fins will handle 50% of each TWTA's dissipation (40.91 W) we get:

$$40.91 \text{ W} + (0.071\text{m}^2 * 0.55 * 1414 \frac{\text{W}}{\text{m}^2}) = A_{fins} * \varepsilon_{fins} * \sigma * T_{fins}^4 \quad (3.9)$$

$$T_{fins}^4 = \frac{40.91 \text{ W} + 13.7 \text{ W}}{0.1612\text{m}^2 * 0.92 * 5.67 \times 10^{-8} \text{ W} \cdot \text{m}^{-2} \cdot \text{K}^{-4}}$$

$$T_{fins} = 283.7 \text{ K}$$

This is an acceptable value that is within the TWTA target range.

TWTA radiator fins account for 50% of total TWTA dissipation of 4419 W, so that heat dissipated by the panels is reduced to only 3355 W. That, plus the bus dissipation of 1456 W, gives 4811 W that needs to be dissipated by the two radiator panels. Repeating Eq. (3.7) with this lower value gives:

$$4811 \text{ W} = 2 * A_{panel} * \varepsilon_{panel} * \sigma * T_{panel}^4 \quad (3.10)$$

$$T_{panel}^4 = \frac{4811 \text{ W}}{2 * 8.89 \text{ m}^2 * 0.80 * 5.67 \times 10^{-8} \text{ W} \cdot \text{m}^{-2} \cdot \text{K}^{-4}}$$

$$T_{panel} = 284.4 \text{ K}$$

This temperature is slightly less than the general operating range of equipment which is necessary for heat to transfer from the interior to the surface. It is also about the same as that found for the radiator fins, which will preclude radiative transfer between those surfaces.

3.8. Thermal Budget

One way to characterize the thermal system is the development of a thermal budget. Starting with the assumption that the panels will radiate at a baseline temperature, we can identify thermal margins for each radiating surface. The four main radiating surfaces are the North, South and ABI panels, and the direct radiation TWTA fins.

The Thermal Budget is shown in Table 3-14, below. Positive margins are shown for all radiating surfaces, with substantial margin on the ABI radiator. Though the nominal mission is designed such that there would be no solar incident on the radiator panels, a 5° uncertainty was allowed for on the North panel. Since the ABI radiator faces south and it is required to maintain a very strict environment, it is assumed that the operational decision would be to allow the pointing to drift such that the North panel is illuminated before the South. Therefore, no solar term is used on the South or ABI radiators.

Table 3-14 Thermal Budget

	Radiators			
	North	South	ABI	Fins
Area [m ²]	8.89	8.89	3.28	0.16
T _{ref} [°C]	15.0	15.0	0.0	25.0
Capability [W]	2782.6	2782.6	828.6	62.2
Dissipation				
Comms [W]	1659.5	1695.6	0.0	40.9
MET [W]	0.0	0.0	650.0	0.0
Bus [W]	728.2	728.2	0.0	0.0

Solar [W]	240.3	0.0	0.0	13.7
Total [W]	2628.1	2423.8	650.0	54.6
EOL Margin [W]	154.5	358.7	178.6	7.6
EOL Margin [%]	6%	15%	27%	14%

Reference temperatures were selected based on the thermal balances performed above, and by inspection of the equipment operating temperatures.

3.9. Thermal Mathematical Model

The spacecraft model was built using Siemens NX. The package includes NX Space Systems Thermal, a thermal finite element model (FEM) solver that includes capabilities to define and solve thermal models with orbits, solar heating, and radiation to space.

The software can perform both steady state and transient solutions, using Newton's iteration method. The governing equation for steady state analyses is:

$$[K]\{u\} + [R]\{u\}^4 = \{P\} + \{N\} \quad (3.11)$$

Where:

$[K]$ is the conduction matrix

$\{u\}$ is the vector of nodal temperatures (in an absolute scale)

$[R]$ is the radiation exchange matrix

$\{P\}$ is the vector of applied heat loads that are independent of temperature

$\{N\}$ is the vector of nonlinear heat loads that are temperature dependent

For transient analysis the governing equation includes a heat capacity term as below.

$$[B]\{\dot{u}\} + [K]\{u\} + [R]\{u\}^4 = \{P\} + \{N\} \quad (3.12)$$

Where

$[B]$ is the heat capacity matrix

$$\{\dot{u}\} = \{du/dt\}$$

The software has a robust modeling architecture where the FEM can be designed to match the CAD modeling through use of assembly FEM files. An assembly FEM is built out of individual component sub-FEM, in the same manner as assembly components are built from subcomponents.

This approach was taken here, resulting in the following hierarchy of FEM and CAD:

- PCW Spacecraft
  - Bus Core
    - Central Cylinder
    - Ox Tank
    - Fuel Tank
    - Mid-Deck
  - North Panel
    - TWTA
    - Panel
    - North West Battery
    - North East Battery
  - South Panel
    - Panel
    - South West Battery
    - South East Battery
  - East Panel
  - West Panel
  - Earth Panel
  - ABI
    - Base
    - Radiator
    - Aperture Shroud
  - North Solar Array
    - Yoke

- Panels x 3
- South Solar Array
  - Yoke
  - Panels x 3
- Reflector x 2

The complete CAD model is shown in Figure 3-17.

Figure 3-17 PCW CAD Model

Many assumptions are made in order to simplify the thermal model and speed solution times. Wherever possible the FEM and CAD were simplified to represent the most important thermal features of the design.

3.9.1. Bus Core

The bus core consists of a structural cylinder, fuel tanks and equipment deck. The components modeled in the core are not meshed for analysis. Instead, they are included in the spacecraft lumped mass, which is the only node in this sub assembly. The bus lumped mass approximates the total spacecraft mass plus fuel, less the reflectors, exterior panels, batteries, solar arrays, and the single TWTA that is modeled.

The lumped mass is modeled as a point node and was defined as 2000 kg of Aluminum with two main thermal connections: the first is a near infinite ($20000 \text{ W/m}^2\cdot\text{K}$) conductance to the radiators, and an MLI-like ($5 \times 10^{-5} \text{ W/m}^2\cdot\text{K}$) conductance to the non radiating exterior surfaces. The low conductivity to the non-radiator exterior surfaces models the insulating effect of the MLI, and the high conductivity to the radiators captures the effects of heat pipe and radiative transport to those panels.

3.9.2. Radiator Panels

The three radiator panels (North, South and ABI) were modeled as solid polygons, rather than honeycomb structures. The intricacy of the honeycomb would lead to a high number of nodes, which would greatly increase the time to a solution. Embedded heat pipes are similarly not modeled, but rather the panels are given customized orthotropic conductivity values based on the work of Joshi et.al. (34). They found an overall effective thermal conductivity of panel with embedded heat pipes to be $33.77\text{-}34.42 \text{ W/m}\cdot\text{K}$ parallel to the panel face, and $4.201 \text{ W/m}\cdot\text{K}$ in the normal direction.

In the model for this thesis the values:

- $k_{x,y} = 34.0 \text{ W/m}\cdot\text{K}$
- $k_z = 4.2 \text{ W/m}\cdot\text{K}$

are used, where x and y are in the panel plane, and z is normal to the panel. The panel was modeled using a tetrahedral mesh with 0.75 m sized elements.

NX allows for the division of faces into sub faces, which is an effective way to model the payload mounted to these panels. Instead of creating multiple models of each component, the mounting side of the radiator panels was subdivided to represent component

footprints as shown in Figure 3-18 and Figure 3-19. The high thermal dissipation units: TWTAs, SSPAs, OMUX, and Batteries mounted to the North and South radiators are modeled this way.

All other units are included in the spacecraft lump mass for modeling analysis.

Figure 3-18 North Radiator Panel

Figure 3-19 South Radiator Panel

These panels were connected via modeled thermal connections to the adjoining panels and to the spacecraft lumped mass. The panel non-radiator surface conductance was assumed to be low ($5 \times 10^{-5} \text{ W/m}^2 \cdot \text{K}$), while the conductance to the lumped mass was assumed to be high ($20000 \text{ W/m}^2 \cdot \text{K}$).

3.9.3. Ka Band TWTA

One detailed TWTA model was included to verify the radiating fin calculations from Section 3.7.5. It is modeled based on the TWTA produced by L-3 Communications (23). The model is shown in Figure 3-20.

The surfaces were meshed with a 2D mesh, with the following characteristics:

- Z93 White Paint on fins
- Z306 Black paint on TWTA body

- Mounting interface to radiator panel conductance = $200 \text{ W/m}^2\cdot\text{K}$

Figure 3-20 TWTA Model

3.9.4. Batteries

The batteries are modeled with a two dimensional (2D) shell that is thermally connected to a lumped mass of 52.3 kg of Aluminum. The batteries are assumed to be covered in MLI to isolate them from the rest of the spacecraft, due to their more restrictive thermal environment. The interface between the batteries and the mounting surface on the radiator is assumed to have a conductance of $200 \text{ W/m}^2\cdot\text{K}$ to facilitate conduction to the panel.

3.9.5. East West, Earth, and Anti-Earth Panels

These panels were all modeled using a 2D surface mesh representing a Black Kapton finish over MLI blanketing, and a three dimensional (3D) mesh applied to the panel with characteristics of an Aluminum honeycomb with no embedded heat pipes.

The insulating effect of the MLI was implemented by limiting the conductance between these surfaces and the spacecraft lumped mass to $5 \times 10^{-5} \text{ W/K}$.

3.9.6. ABI

The ABI was modeled using two sub-components: the base and the radiator. The radiator was modeled similarly to the North and South radiators, where the panel itself is a 3D simplification of honeycomb aluminum with embedded heat pipes, and the radiating surface modeled with a 2D surface coat of OSR.

The interior face of the radiator was subdivided as shown in Figure 3-21, where the ABI base was thermally connected to the green region. A 2D mesh with the properties of gold Kapton was applied to the remaining area which is exposed to space, as well as to the spacecraft Earth panel.

Figure 3-21 ABI Radiator Model

The base consists of a rectangular cube 1.00 m by 1.50 m by 1.25 m, with five faces covered in goldized Teflon, the radiator interface panel, and a lumped mass that is

infinitely connected to the surface faces. The lumped mass is modeled as 285 kg of Beryllium.

The thermal conductance between the lumped mass and MLI was $5 \times 10^{-5} \text{ W/m}^2 \cdot \text{K}$, the conductance to the radiator panel was $20000 \text{ W/m}^2 \cdot \text{K}$.

The spacecraft-ABI interface is assumed to be only through radiation between the two facing panels, namely the ABI Baseplate panel and the Earth panel. It is assumed that at the mounting points, sufficient thermal resistance can be designed, so as to render conductive transfer through the mounting points negligible.

3.9.7. Solar Arrays

The arrays are a simple 3D model, similar to the North and South radiator panels, except that the surface coat models solar cells on the Sun side, and black paint on the back side. The thermal conductivity of the modeled honeycomb is less than that of the radiator panels as it is assumed that embedded heat pipes are unnecessary here, as the distribution of Sun will be uniform across each panel.

The intent of the Sun-nadir steering and solar array steering is to keep the solar arrays pointed normal to the Sun to maximize power. Assuming successful steering, the relative position of the solar arrays with respect to the Sun should not change appreciably over life. If the mission goes according to design, the only time the solar cells would not be directly illuminated would be during eclipse.

For this reason, the solar arrays are solved separately and include only transient analyses involving eclipses in June and December. This relieves the solver from computing array

positions and temperatures along with the extensive bus nodes while still providing the bounding cases for the solar arrays.

3.9.8. Communications Reflectors

Two reflectors were modeled as simple disks with 2.5 m diameter and 0.025 m thickness. Though such a complex payload would certainly require more reflectors, it is beyond the scope of this thesis to determine the number and characteristics, so this assumption is made to provide an analysis of the environment a typical reflector might experience in the TTO Orbit. It is assumed that 0.5 db, or roughly 12% of the RF power striking the reflector is lost. This loss is dissipated as heat in the reflector.

A 2D finite mesh is applied to the surfaces only and they are modeled to radiate as a black body. Though the reflectors would more likely be mesh, the models are solid, both absorbing and radiating more heat than a meshed material would. It is assumed that this will have negligible impact on the surface temperatures of the reflectors themselves.

The surface meshes are thermally connected to a 0D lumped mass of 20 kg graphite, based on a similar sized design by Vanguard. (25)

3.9.9. UHF Helix Antennas

The UHF helices are modeled as solid conical parts, 4 m tall, and 0.5 m diameter at the base. The back splash is a cup of 1 m outer diameter. Only the surfaces are meshed to simulate a surface covering. The surface is thermally connected to a lumped mass of 30 kg of Aluminum. For the receive antenna, the losses will be negligible as very low power is received at TTO orbit altitudes. The transmit antenna however will experience significant losses, as it is carrying the recently amplified signal. Losses due to resistance

in the antenna are assumed to be dissipated as heat and to be 0.5 dB, or 12% of the power.

3.9.10. Modeled Loads

There are a number of thermal loads in the model to represent the major heat sources in the spacecraft, shown in Table 3-15.

Table 3-15 NX Model Thermal Loads

Load	Value	Location	Active Mode
Spacecraft Electronics Load	1456.4 W	To spacecraft lumped mass 0D node	Always
ABI Load	650.0 W	ABI lumped mass	Always
Battery Discharge Load (4)	4 • 225.0 W	One each applied to the battery lumped masses while discharging	During Eclipse
Orbital Heating	Varies case to case (1322-1414W/m ²)	All spacecraft external surfaces could be illuminated.	Non-Eclipse
Ka TWTA Unit Load	81.8 W	Ka TWTA unit lumped mass	Comms Active
Ka TWTA Footprint Load	17 • 40.9 W	Ka TWTA footprints on North Panel	Comms Active
KA OMUX Load	4 • 115.9 W	Ka OMUX footprints on North Panel	Comms Active
X TWTA Footprint Load	6 • 40.9 W	X TWTA footprints on South Panel	Comms Active
X OMUX Load	2 • 103 W	X OMUX footprints on South Panel	Comms Active
UHF SSPA Footprint Load	10 • 60 W	UHF SSPA footprints on South Panel	Comms Active
UHF OMUX Load	4 • 58.9 W	UHF OMUX footprints on South Panel	Comms Active
Ka/X Reflector Load	2 • 77.5 W	Ka/X Reflectors	Comms Active
UHF Tx Helix Load	64 W	UHF Tx Helix	Comms Active

Part 4: Results and Discussion

The results of the TTO thermal model were compared to results of the same model run in conditions that a GEO satellite would experience.

Four TTO scenarios were used for the steady state and transient solutions:

1. TTO cold case: BOL, RAAN = 0° , June solstice, limited operations in South
2. TTO hot case: EOL, RAAN = 0° , December solstice, all payloads operations
3. TTO cold eclipse: BOL, RAAN = 90° , June solstice, limited operations in South
4. TTO hot eclipse: EOL, RAAN = 90° , December solstice, all payloads operations

For GEO, a satellite is operational throughout the 24 hours of an orbit and was modeled as such. The GEO solutions used for comparisons are:

1. GEO cold case: BOL, June solstice, all payloads operational.
2. GEO hot case: EOL, December solstice, all payloads operational.
3. GEO eclipse: EOL, March equinox, all payloads operational.

Further analyses include mesh sensitivity, orbit propagation sensitivity and solar array transient.

4.1. Steady State Results

Steady state solutions were developed for the worst cases, without any orbit propagation. The results for TTO worst cases are compared to the target temperatures and to similar results for GEO worst cases, in order to characterize the different thermal environment.

The temperatures of interest are:

- Spacecraft lump mass: SC Lump
- ABI lump mass: ABI Lump
- TWTA lump mass: TWTA Lump
- The Ka/X Reflectors, East and West
- The UHF Helices: Transmit (Tx) to the North East and the Receive (Rx) to the South West
- Batteries: NE, NW, SE, SW
- Exterior surface averages: all panels and radiators on spacecraft and ABI

4.1.1. Steady State Cold

In both the GEO and TTO cold case solutions, the spacecraft is assumed to be on the far side of the Earth from the Sun. Results are presented in Table 4-1 and again graphically in Figure 4-1 and Figure 4-2. Temperatures that fall outside the target range are shown in red.

A number of the temperatures in the TTO case fall below the cold operating limits, most notably the spacecraft, TWTA, and battery lump masses. In contrast the GEO solution shows all temperatures within limits except for the anti-Earth panel.

The main difference between these cases is that placing the payload into a reduced operations mode leads to a significant decrease in spacecraft temperatures for the TTO as compared to GEO. As previously described, adding compensation heaters to maintain minimum temperatures is straight forward solution, at the cost of increased power requirement. Should the final PCW mission require partial operations in over the non-Arctic portion of the orbit, less compensation will be required.

Table 4-1 Steady State Worst Case Cold Results

	Target Low [°C]	GEO [°C]	TTO [°C]
SC Lump	0	12.2	-32.3
ABI Lump	-5	19.7	8.9
TWTA Lump	10	24.7	-31.7
East Reflector	-150	37.2	51.7
West Reflector	-150	52.4	30.3
Tx Helix	-90	-54.5	-59.9
Rx Helix	-90	-44.3	-50.2
SW Battery	0	4.9	-36.1
NW Battery	0	6.7	-35.7
SE Battery	0	4.8	-36.1
NE Battery	0	6.8	-35.6
North Panel Average	-35	13.5	-35.6
South Panel Average	-35	12.2	-33.5
ABI Radiator Average	-35	11.8	-0.8
ABI External MLI Avg.	-150	-27.1	-32.3
ABI Baseplate Avg.	-150	97.5	83.3
Earth Panel Average	-150	131.2	116.2
Anti Earth Average	-150	-250.0	-251.1
East Panel Average	-150	-85.2	49.0
West Panel Average	-150	-75.2	-89.4

Figure 4-1 TTO Cold Case Steady State Results (South, East and Anti-Earth)

Figure 4-2 TTO Cold Case Steady State Results (North, West and Earth)

4.1.2. Steady State Hot

The results for the steady state hot case are presented in Table 4-2 and graphically in Figure 4-3 and Figure 4-4 TTO Hot Case Steady State Results (North, West and Earth). Both the GEO and TTO results are within the high temperature limits, with the small exception of the GEO spacecraft lump. The GEO spacecraft lump exceeds the target temperature by a slight margin, but remains well within the equipment Operating range.

In all the TCS performs better in the TTO orbit due to the Sun nadir steering that avoids incident Sunlight on the radiator panels. From Figure 3-2, the GEO case shown here has an illuminated North radiator panel.

Table 4-2 Steady State Worst Case Hot Results

	Target High [°C]	GEO [°C]	TTO [°C]
Spacecraft Lump	21	22.1	8.6
ABI Lump	35	30.1	6.2
TWTA Lump	65	34.0	25.2
East Reflector	85	58.0	6.1
West Reflector	85	58.5	-164.4
West Helix	90	-71.9	-57.9
East Helix	90	-196.9	-45
SW Bat	25	17.1	1.7
NW Bat	25	14.8	2.5
SE Bat	25	17.0	1.7
NE Bat	25	15.0	2.6
North Panel Average	50	21.3	9.6
South Panel Average	50	22.9	8.8
ABI Radiator Average	50	20.9	-3.3
ABI External MLI Avg.	240	-111.2	-54.4
ABI Baseplate Avg.	240	-103.4	-46.8
Earth Panel Average	240	-126.3	-83.3
Anti-Earth Panel Avg.	240	124.9	-250.1
East Panel Average	240	-75.1	107.6
West Panel Average	240	-72.7	-179.8

Even in the steady state, the TCS design in TTO shows some temperature margin, as compared to the reference temperatures presented in Table 3-14. The spacecraft and ABI radiator panel temperatures are all below that reference temperature. The TWTA lump is slightly higher than the reference temperature, but well within operating limits.

The temperature of the radiator fins (which are isothermal with the TWTA mass) are very close to the reference temperature from Table 3-14, which confirms that as a valid assumption.

4.1.1. Steady State Eclipse

Though the eclipse durations are not long enough for the spacecraft to reach a steady thermal state before the end of the eclipse, an analysis of the eclipse steady state is valuable in that it shows the extreme temperatures that might be reached under such conditions.

The cold case for TTO uses a reduced communications load and no battery discharge load while the hot case uses the full communications load and the battery discharge load. The results are shown in Table 4-3.

Not surprisingly there are a number of temperatures that exceed limits, both low and high. The spacecraft lump mass temperature on the other hand remains well within the Target range of the electronics and equipment.

Figure 4-3 TTO Hot Case Steady State Results (South, East and Anti Earth)

Figure 4-4 TTO Hot Case Steady State Results (North, West and Earth)

Another interesting result is that the ABI temperature is marginally *less* in the hot case. This shows that the spacecraft and ABI are adequately thermally isolated from each other, at least to the limit of the ABI radiator's capability to dissipate the excess flux.

Table 4-3 TTO Eclipse Worst Case Steady State Results

	Target Low [°C]	Cold [°C]	Hot [°C]	Target High [°C]
Spacecraft Lump	0	-14.8	18.9	21
ABI Lump	-5	1.3	1.2	35
TWTA Lump	10	-24.0	24.6	65
East Reflector	-150	-180.9	-179.8	85
West Reflector	-150	-181.2	-181.2	85
West Helix	-90	-164.3	-159.0	90
East Helix	-90	-177.7	-135.4	90
SW Bat	0	20.7	51.0	25
NW Bat	0	17.5	48.2	25
SE Bat	0	21.0	51.3	25
NE Bat	0	17.0	47.7	25
North Panel Average	-35	-19.3	18.3	50
South Panel Average	-35	-16.2	18.8	50
ABI Radiator Average	-35	-8.3	-8.4	50
ABI External MLI Avg.	-150	-155.0	-157.2	240
ABI Baseplate Avg.	-150	-136.0	-138.1	240
Earth Panel Average	-150	-152.7	-159.8	240
Anti-Earth Panel Avg.	-150	-250.6	-249.9	240
East Panel Average	-150	-206.9	-205.0	240
West Panel Average	-150	-209.0	-211.4	240

4.2. Transient Results

Transient results were produced using the same model as for the steady state solutions, with the orbit propagated through 5 days to achieve a stable diurnal periodicity in temperatures. June and December cases are run both with and without eclipse. Similar

transient results for a GEO in June, December are compared to the no-eclipse cases, while a GEO eclipse (March) is used to compare to both of the TTO eclipse cases.

All TTO transient cases start at the descending node. A steady state analysis is performed prior to the transient analysis to develop initial conditions. An analysis of the effect of initial conditions on the final solution appears in Section 4.4.

4.2.1. TTO Cold Case Transient Results

The cold transient case is June with no eclipse, and limited payload operation. The results of this simulation are presented here and are compared to the GEO cold case.

Figure 4-5 shows the lumped mass results for the TTO cold case. Spacecraft temperature dips below some equipment target temperatures at the lowest, and is above target temperatures at the hottest. The wide swing in spacecraft and TWTA temperature is due to the reduction in payload operations over perigee.

Figure 4-5 TTO Cold Case Transient Lump Mass Results

The results for the reflectors and helix antennas are shown in Figure 4-6 and Figure 4-7. As expected, the East components are substantially warmer than the West. For the East reflector, the dominant thermal effect is the orientation with respect to the Sun, the large hump in the plot peaks when the reflector is orthogonal to the Sun at apogee. The smaller peak is perigee.

The West reflector is shadowed for the entire orbit, and falls below the Survival temperature range during the orbit. Compensation will be required to maintain operating limits.

Figure 4-6 TTO Cold Case Transient X/Ka Reflector Results

The UHF Helices are exposed to the Sun more or less constantly through the orbit. The peak occurs around the perigee, when the maximum helix area is exposed to the Northern Sun. Throughout the orbit, the temperatures remain within Target range.

Figure 4-7 TTO Cold Case Transient UHF Helices Results

The battery temperatures in Figure 4-8 mirror the spacecraft lump temperature fluctuations and are below the Target and Operating temperatures, though still above Survival.

Figure 4-8 TTO Cold Case Transient Battery Results

4.2.2. GEO Cold Case Transient Results

The main difference in results between GEO and TTO Cold cases is due to the limited payload operation in the TTO cold case. The GEO cold case spacecraft lump temperature is stable at around 14° C, significantly higher than the warmest achieved by the TTO. Figure 4-9 also shows similar results for the TWTA and ABI lump masses, namely that the GEO temperatures are higher due to full payload operation throughout the orbit.

The TWTA results again show the complex interaction of the radiator fins, spacecraft body and Sun geometry.

Figure 4-9 GEO Cold Case Transient Lump Mass Results

Figure 4-10 shows the results for the Ka/X Reflectors. As expected for a GEO spacecraft, the East and West reflector thermal results are very similar, and within Target range throughout the orbit. The shadow periods are 12 hours apart, as each reflector is shadowed by the spacecraft body. The slight variation in the timing of the peaks is due to

the different positions at which each reflector reaches the maximum exposure to the Sun, and the spacecraft orientation.

Figure 4-10 GEO Cold Case Transient Ka/X Reflector Results

Figure 4-11 shows that the UHF helices remain in Target temperature range throughout the GEO cold case. The difference between Tx and Rx is due to the amplified signal being carried by the Tx antenna and the geometry. The Rx Helix is on the South West of the spacecraft and since in June the Sun is in the North, there is a period in the orbit where the Rx Helix is partially shadowed by the spacecraft body, resulting in the dip in temperature in the plot.

Figure 4-11 GEO Cold Case Transient UHF Helices Results

The battery results are shown Figure 4-12. With the Sun on the North of the spacecraft we see that the NE and NW batteries run at slightly higher average temperatures than their counterparts on the South panel.

Figure 4-12 GEO Cold Case Transient Battery Results

There are spikes seen in the batteries as the sun alternately illuminates the East and West faces. The magnitude of the spike on the West panel is much greater than the East. Even still, all temperatures are safely within the Target range.

4.2.3. TTO Hot Case Transient Results

The transient hot case results for TTO are shown in Figure 4-13 through Figure 4-16. The lump mass results show spacecraft, ABI and TWTA, all within the Target range. Spacecraft and ABI are fairly stable over a daily basis, while the TWTA experiences fluctuations based on the geometry between the radiator fins and the Sun.

With constant operation of the payload, the apogee and perigee can be clearly seen in the TWTA plot. As the radiator axis extends from the East panel, the fin area exposed to the Sun is maximized at those two locations on orbit. There is no shadowing of the East panel, and therefore the radiator fins, which further explains why there is little variation between the highest and lowest temperatures.

Figure 4-13 TTO Hot Case Transient Lump Mass Results

As with the cold case, the west reflector is again in shadow throughout the orbit, and as such, remains below the Target range throughout. It is also below the Survival Range for much of the orbit. The west reflector heater will be a mission critical system in this case. The East reflector cycle differs from the cycle seen in the TTO Cold Case in Section 4.2.1 in that the higher of the two peaks now occurs at perigee in December. Since the perigee passage is much quicker than the apogee passage, this spike is more pronounced than the smooth ‘hump’ earlier seen in Figure 4-6.

Figure 4-14 TTO Hot Case Transient Ka/X Reflector Results

The UHF helices remain within Target temperatures throughout the orbit with substantial margin. The smooth diurnal variation in both demonstrate that the illumination geometry of the helices changes very gradually throughout the orbit, which is expected given that the East face is constantly oriented towards the Sun.

Figure 4-15 TTO Hot Case Transient UHF Helices Results

The battery results shown in Figure 4-16 remain within Target temperatures, and are closer to the lower limits than the high. There is little significant difference between North and South temperatures as neither of those panels experience solar illumination through this orbit.

The most significant aspect of the TTO Hot Case result is that no components exceed the Target high temperature limits. Thus, the radiator areas should be sufficient to meet mission thermal requirements.

Figure 4-16 TTO Hot Case Transient Battery Results

4.2.4. Comparison to GEO Hot Case

The GEO Hot Case results illustrate how the TTO Sun-nadir steering provides a more manageable thermal environment. The ABI temperature is very high compared to other results. Due to the Sun exposure on the South facing ABI Radiator panel, this would be mitigated by inverting the spacecraft so that the South panel becomes the North and vice versa. The TTO has no thermal reason for such a maneuver.

Similarly the spacecraft lump temperature is slightly higher than the Target high of 21 °C, though it should be noted that that value is driven by the N₂O₄ tank which could be further insulated from the spacecraft.

The TWTA is also higher for the GEO Hot Case than for TTO Hot Case, though still well within its Target Range.

Figure 4-17 GEO Hot Case Transient Lump Mass Results

The results for the Ka/X reflectors shown in Figure 4-18 show a much wider range of temperatures than the TTO Hot Case. However, temperatures remain within the Target range at all times, whereas the TTO Case shows low temperatures on the West reflector.

Figure 4-19 shows the results of the UHF Helices. The Rx Helix on the South remains Sunlit through the entire orbit, while the Tx is shadowed by the spacecraft body for a time. The temperatures are about 10 °C warmer than the GEO Cold Case, but the Tx still drops below the Target temperature low limit when it goes into this shadow.

The TTO Hot case is a more stable environment for the helices, as the worst temperature swing is 5-6 °C, compared to 60-70 °C in this case.

Figure 4-18 GEO Hot Case Transient Ka/X Reflector Results

Figure 4-19 GEO Hot Case Transient UHF Helices Results

The GEO Hot Case battery temperatures are higher than those found for the TTO Hot Case. As seen in Figure 4-20, the battery temperatures approach the Target limit of 25

°C, but remain below it. Opposite to what was seen in the GEO Cold Case results, the South battery temperatures are generally slightly higher than the North battery temperatures in this case, as expected. However the larger spikes in the West batteries suggest an additional mechanism at work for the West batteries. The spikes, though 3 °C greater than the East, are still within Target temperatures, so the TCS design is still within requirements.

Figure 4-20 GEO Hot Case Transient Battery Results

4.3. Eclipse Transient Results

The worst case eclipses were modeled. The longest eclipse case for TTO was for December when RAAN = 90°, which is the Hot Eclipse Case. The corresponding eclipse season in June is the Cold Eclipse Case. Both are compared to the GEO equinoctial eclipse.

4.3.1. TTO Cold Eclipse Transient Results

The June eclipse accounts for limited payload operations over the South Pole. The deepest eclipse in June is 58 minutes long, and occurs shortly after the payload would be put into such limited operations.

Figure 4-21 shows a spacecraft lump mass temperature profile that is very similar to that of the non-eclipse cold case. This suggests that the spacecraft shadowing effect is very nearly matched by the thermal dissipation from the battery discharge, resulting in a near zero net result on the spacecraft. At the very least the payload operations dominates the eclipse conditions of being in Earth shadow and batteries discharging.

The effect of the change in orbit orientation from $RAAN = 0^\circ$ in the cold case to $RAAN = 90^\circ$ in this case is evident from the TWTAs results. The non-eclipse case (Figure 4-5) shows a smoothed curve for the TWTAs temperatures, this case shows a choppier result. In the non-eclipse case, the East panel (and thus the TWTAs radiator fins) faces the Sun through the entire orbit. For the eclipse case, the Sun is in the orbit plane and the spacecraft body is rotating about the pitch axis. Put another way, the Sun appears to rotate about the radiator fins, and the steps show the effect of variation in fin area exposed to the Sun as the spacecraft progresses in the orbit.

The steep drop in TWTAs temperature is due to the limiting of payload operations beginning at the descending node. This effect is soon followed by a small recovery due to heat dissipated by the battery discharge, which heats the panel that both the batteries and TWTAs are attached to. At the lowest point the TWTAs is below the Target and Operating Ranges, so compensation heaters will be needed to maintain the required temperature.

Figure 4-21 TTO Cold Eclipse Transient Lump Mass Results

The results for the Ka/X Reflectors are shown in Figure 4-22. The lowest temperature reached by the East reflector is -158.3 °C, which is lower than the Target range, but still within the Survival range. The West reflector is within the Target range throughout.

The difference in the depth and duration of the dips in temperature between the reflectors is due to the orientation of the spacecraft with respect to the Sun. The East face points in the direction of the velocity vector, and the Sun is opposite from the Earth when the spacecraft is at the descending node in this orbit. The East reflector is in Sunlight through perigee while the West reflector is shadowed by the spacecraft body, while the opposite is true through apogee. Because the spacecraft moves more slowly through apogee, the East reflector has more time in shadow to cool. The West reflector also cools through apogee as the bluff area of the reflector exposed to Sun reduces to when it is edge-on.

The eclipse effect is seen in the rapid drop and rise that appears in the earliest part of the plot and repeat each day. The eclipse is soon followed by temperature drops of varying magnitude due to the reflector-Sun geometry through perigee.

Figure 4-22 TTO Cold Eclipse Transient Ka/X Reflector Results

The results of the TTO Cold eclipse for the UHF Helices are shown in Figure 4-23. Both helices remain within the Target temperature range during the eclipse and the rest of the orbit. The Rx helix does reach -89.6 °C in the extreme, which is only slightly above the Target limit of -90 °C.

The Rx helix also reaches a slightly higher maximum temperature than the Tx helix due to similar spacecraft geometry as described above for the Ka/X reflectors. Through apogee, the Rx antenna (which is on the South West of the spacecraft) is more illuminated by the Sun than the Tx helix in the North East. The Sun is a more dominant thermal effect than the high power transmission.

Figure 4-23 TTO Cold Eclipse Transient UHF Helices Results

The battery results shown in Figure 4-24 show low temperatures reached on all four batteries. The peak temperatures achieved during eclipse are comparable to the high temperatures reached by the West batteries through apogee. The significant discharge load on the batteries has such a small effect on the battery temperatures because the payload is in limited operations mode for the duration of the eclipse season. The removal of the payload dissipation from the panel leaves the panel, and consequently the batteries, somewhat cooler than will be seen in the following eclipse cases.

The high temperatures are within the Target limits and are similar in magnitude to those achieved in the GEO non-eclipse cold transient case that has no battery discharge. The low temperatures are also similar to the TTO cold case.

An interesting result from this plot is the difference in temperature profiles between the batteries to the West compared to the batteries to the East. The West batteries experience

a temperature rise before all the batteries experience the sharp rise associated with the battery discharge. This rise does coincide with the part of the orbit when the West panel is Sunlit, and despite the low conductance of the MLI. The magnitude of the rise is unexpected, but still well with the Target temperature limit.

Figure 4-24 TTO Cold Eclipse Transient Battery Results

4.3.2. TTO Hot Eclipse Transient

The TTO December eclipse for RAAN = 90° lasts close to 100 minutes. The results for this simulation are shown in Figure 4-25. The TWTA results are expectedly higher than the TWTA results for the TTO Cold Eclipse case as the payload remains in full operation throughout the orbit. Temperatures remain within Target limits for TWTA, spacecraft and ABI with substantial margin on each. Similarly to the TTO Non-eclipse case, the limit of concern from the plots is the lower limit as both SC and ABI are closer to their lower Target limits than the high.

The TWTA presents the interesting results again, as the progression through this orbit can be seen in the profile. The eclipse is the third small peak that appears just before the large rise that lasts for the majority of the orbit. The low temperature part of the TWTA plot is shows period when the radiator fins are shadowed by the spacecraft body and the Ka/X reflectors before entering eclipse. On eclipse exit the fins are exposed to the Sun over the orbits apogee the dip in temperature at this time occurs when the radiator fins are oriented end on to the Sun, thus minimizing the illuminated surface area.

Figure 4-25 TTO Hot Eclipse Transient Lump Mass Results

The results for the Ka/X reflector are presented in Figure 4-30. They closely resemble the results for the TTO Cold Eclipse case with slightly higher temperatures, different eclipse timing and duration, and inverted profiles for the East and West reflectors. The deep drops in temperature correspond to shadowing periods of that reflector by either the spacecraft body or the Earth. The periods when the spacecraft shadows a reflector by

geometry also defines the orientation when the other reflector is a near a minimum of surface area exposed to the Sun.

The West reflector drops below the Target temperature range, reaching -156.8°C at the lowest point. This is still within Survival limit of -160°C but only marginally. All other temperatures are within Target range.

Figure 4-26 TTO Hot Eclipse Transient Ka/X Reflector Results

The UHF helices results are shown in Figure 4-27. Both Antennas reach temperatures below the Target range of -90°C , but within the Survival range of -100°C . This may be counter-intuitive when compared to the TTO Cold Eclipse case where the Helix temperatures don't violate the low limits. However, the longer eclipse in this case is the main reason for the lower extremes. The difference in durations (100 vs. 58 minutes) allow the helices to reach lower temperatures during the warmer December season.

In contrast, the high temperatures reached here are greater than the high temperatures reached in the June Eclipse case.

Figure 4-27 TTO Hot Eclipse Transient UHF Helices Results

The results for the batteries for the TTO Hot Eclipse case are shown in Figure 4-28. The various spikes are the transient results of the cyclical heating and cooling of the West and East panels as well as the more obvious effect of the battery discharge load during eclipse.

This is a crucial result for the TCS design. Even in the hottest season, with the long eclipse and discharge duration, and full payload operations, the TCS is able to dissipate all the heat and maintain battery temperatures within Target ranges. This is in addition to meeting the Target temperatures for TWTA, ABI and spacecraft lump temperatures with significant margin.

Figure 4-28 TTO Hot Eclipse Transient Battery Results

4.3.1. GEO Eclipse Transient Results

The model was run in a typical GEO eclipse scenario as a comparison to both of the TTO eclipse cases. Though there are two eclipse seasons in GEO, (around the March and September equinoxes) they are thermally similar enough that one case covers both seasons. The results are presented in this section.

Figure 4-29 shows the lump mass temperature results. All temperatures are within Target ranges and the TWTA pattern closely resembles that of the TTO December eclipse case. This is because in both the GEO and the TTO December Eclipse, the Sun lies in the orbit plane, and the spacecraft is oriented the same way with respect to the Sun. The timing of the peaks and valleys is changed due to the difference in eccentricity of the two orbits, but the same cycle of Sunlight and shadowing emerges in both cases. Also the range of temperatures experienced is very similar between the two.

The same pattern is not evident in the TTO June eclipse case due to the drop in payload dissipation that results from the limited payload operations mode over the South Pole. Another scenario with full operations through the June eclipse orbit may reveal a similar pattern.

Figure 4-29 GEO Eclipse Transient Lump Mass Results

The results for the Ka/X Reflectors shown in Figure 4-30 confirm that the reflectors are identical in design, as the two plots are near reflections of each other. The temperatures experienced are similar to both TTO eclipse cases although the GEO reflectors stay within Target ranges at all times, only reaching a minimum of -149.2 °C in the worst case (on the East reflector), whereas both the TTO eclipse cases experienced minimum temperatures cooler than that.

Figure 4-30 GEO Eclipse Transient Ka/X Reflector Results

Figure 4-31 shows the results for the UHF Helices. As with the reflectors, the GEO results are similar to both TTO eclipse cases, with very similar profiles and temperature extremes.

Figure 4-31 GEO Eclipse Transient UHF Helices Results

The results for the batteries are shown in Figure 4-32. The rise due to battery discharge is significant, but within Target range. The West batteries again experience a larger rise than the East when those panels are directly illuminated through the orbit. The magnitude of the spike is below the levels reached by the discharge, and well within the Target limit.

Figure 4-32 GEO Eclipse Transient Battery Results

4.4. Initial Conditions Analysis

The selection of initial conditions in transient analyses sometimes has a bearing on the final results. All transient cases above were run starting from a steady state solution at the simulation start time conditions. To verify that the final transient results are independent of the initial conditions, solutions were also run using 20 °C as the starting temperature for all nodes.

The results for the GEO Cold Case are a good example of how the two different initial conditions still reach the same periodic solution, and how fast that solution is reached.

The simulation results for the GEO Cold Case with 20 °C Initial Conditions are show in Figure 4-33 through Figure 4-36.

Figure 4-33 GEO Cold Case Lump Mass Results with 20 °C Initial Conditions

Figure 4-34 GEO Cold Case Reflector Results with 20 °C Initial Conditions

Figure 4-35 GEO Cold Case UHF Helices Results with 20 °C Initial Conditions

Figure 4-36 GEO Cold Case Battery Results with 20 °C Initial Conditions

4.5. Mesh Variation Analysis

The selection of mesh size can impact final solution results at the cost of increased solution solve time. The TTO December Eclipse case was run with increased meshing on select components.

Table 4-4 Element Sizes for Regular and Fine Meshes

	Regular Element Size	Increased Mesh Element Size
North/South Panels	750 mm	250 mm
East/West Panels	500 mm	250 mm
Earth/Anti-Earth	750 mm	250 mm
TWTA Fin Surface	25 mm	12.5 mm

The simulation was run with the increased mesh for only two full orbits, to save on computing time. The results are shown in Figure 4-41 through Figure 4-44.

As the mesh did not change for the reflectors or helices, the results for these components match the original results. Where the finer mesh appears to have an impact is in the results for the TWTA (Figure 4-41) and the batteries (Figure 4-44).

The change in mesh has resulted in a lower peak TWTA temperature: it reaches just below 20 °C, and is quite flat in that part of the curve, which is considerably different from the results in Figure 4-25 that show greater temperatures and a dip that is thought to represent the point in the orbit when the TWTA fins present a minimum area to the Sun. The temperatures are far below the TWTA target range, so in either case the TCS is sufficiently controlling the temperature.

The other curious result is that the prominent spike in the West batteries all but disappears when using the finer mesh. The batteries show similar extreme with the

discharge load applied. This suggests that the larger spikes in the West vs. East batteries in previous results could be due an effect from the mesh size.

With the coarse mesh, battery temperatures reached by the spikes are always less than the peak temperatures reached due to battery discharges. Because of this, they do not constitute the worst case and consequently have no impact on the TCS design. Since the computing time for the increased mesh was four times longer than the coarser mesh, and the finer mesh *improves* the results, the coarse mesh results are deemed sufficient to evaluate the design.

Figure 4-37 Fine Mesh Lump Mass Results

Figure 4-38 Fine Mesh Reflector Results

Figure 4-39 Fine Mesh UHF Helices Results

Figure 4-40 Fine Mesh Battery Results

4.6. Orbit Propagator Sensitivity

The orbit propagator also has the option to increase the sensitivity of the analysis by varying the number of locations at which the environmental conditions are computed. The general results use 12 locations with intermediate values interpolated linearly with time.

A more sensitive case was run using 48 locations to test the model's sensitivity to this parameter. Again using the TTO December eclipse case, the results for this test are shown in Figure 4-41 through Figure 4-44. The results are similar to the original test results with some improvement in curve smoothness, but little significance in the temperatures reached.

Figure 4-41 Increased Orbit Sensitivity Lump Mass Results

Figure 4-42 Increased Orbit Sensitivity Reflector Results

Figure 4-43 Increased Orbit Sensitivity UHF Helices Results

Figure 4-44 Increased Orbit Sensitivity Battery Results

4.7. Solar Array Transient Results

As the solar cells should always be facing the Sun, the solar array thermal environment changes very slowly throughout the course of an orbit. To characterize the extremes, two simulations were run with the solar array models. For both cases RAAN = 90 was selected so that eclipses were in the extreme seasons. As shown in Figure 4-45 and Table 4-5, the solar arrays meet thermal requirements even in the most extreme cases.

Figure 4-45 Solar Array Transient Results

Table 4-5 Solar Array Extreme Temperatures

	Target [°C]	June [°C]	Dec. [°C]
High	100	37.4	42.8
Low	-95	-34.1	-45.4

Part 5: General Conclusions and Recommendations

5.1 Conclusions

The thesis used Siemens NX 8.5, to model a typical commercial geostationary (GEO) communications satellite thermal control system (TCS). The results of the thermal mathematical model in a the Telesat Tundra orbit (TTO) proposed for the Polar Communications and Weather (PCW) mission compared well against the model's performance in the GEO environment. Most temperatures were able to be maintained within a 20 °C target range. The NX software is a validated and well established industry standard software package. Further validation of the model could be performed by comparing the model's GEO results to actual GEO spacecraft data.

The goals of the thesis were to:

1. Identify worst case hot and worst case cold conditions.
2. Characterize eclipse durations and thermal effect.
3. Characterize impact of the TTO orbit on thermal design.
4. Analyze the suitability of typical GEO TCS to the Tundra orbit PCW system.

The hot and cold cases and the eclipse definitions were developed in Part 2. As with GEO spacecraft, Albedo and IR were found to be negligible. Unlike GEO, the TTO has many possible orientations with respect to the Sun, depending on mission design and ground footprint requirements. This thesis varied the right ascension of the ascending node (RAAN) with the assumption that a TCS that controls the worst hot and cold possible cases would also be suitable for the cases in between.

The worst cases for the TTO orientations occur at the two solstices, June and December, when solar flux is at minimum and maximum, respectively. For $RAAN = 0^\circ$, there are no eclipses at the solstices, while $RAAN = 90^\circ$ will have eclipses in those seasons. Both of these cases were simulated for June and December and compared to the analogous GEO conditions.

The analysis in Chapter 4 showed that a typical GEO thermal control system is a suitable baseline design for PCW in a TTO orbit. Further, the Sun-nadir steering required by the TTO suggests a less volatile thermal environment as the radiator panels on the North and South faces of the spacecraft are never illuminated by the Sun, unlike GEO where the solstice seasons cause one of the radiators to be illuminated.

A less volatile thermal environment is desirable especially for the Meteorological (MET) instrument which requires cryogenic cooling to maintain operating temperatures for the detector. A more stable external environment in TTO means that a design suitable for GEO missions can be used, as the case for the proposed Advanced Baseline Imager (ABI).

Many of the results show surface temperatures of MLI that are below the lower temperature thresholds, specifically the west panel MLI when Sun-nadir steering is active, and the anti-Earth panel that has little instrumentation associated with it.

Bringing these temperatures into required ranges can be accomplished by selecting MLI with fewer layers (therefore less insulating) on those panels. This would allow more spacecraft internal heat to dissipate through those panels, warming those surfaces. It would also have the added effect of reducing the heat dissipation load on the radiators. In

subsequent design iterations, perhaps the radiating area could be reduced to save mass and cost.

The West batteries showed an anomalous temperature spike that did not reflected during similar conditions on the East. The spikes remained within Target temperatures at all times and the discrepancy disappeared when the mesh was refined. The design impact is negligible, as the TCS design is driven by worst cases. The increased computing time of repeating the solutions with the fine mesh was deemed to have little additional value as the TCS design and final result would be unaffected.

The TTO presents as a cooler, more thermally stable environment than GEO. Though the PCW mission may call for highly variable payload dissipation, the thermal variation is well within the capability of GEO TCS design. The final conclusion of this thesis is that a typical GEO TCS design is suitable for PCW in TTO with no need for additional control techniques.

5.2 Recommendations for Future Work

The Government of Canada released a Request for Information on PCW in late 2013, so the mission will continue to be further refined. Work on the TTO orbit thermal environment would provide greater understanding of the mission alternatives leading up to a potential mission.

Work that could build on this thesis could include:

- Mission refinement: selection of the orbit orientation in space would provide greater certainty in the annual thermal environment, and bound the eclipses, and the hot and cold worst cases.

- Attitude control: this thesis assumed ideal Sun-nadir steering was achievable, which may not be the case. Though margins were added to the thermal design to cover a worst case steering, characterization of the attitude control capability would allow designers to develop more stringent margins.
- Model refinement: detail can be added to the model, including more complex modeling of the internal spacecraft environment in order to verify the internal TCS. Specifically, a model with all if the TWTA radiators included would be valuable. Specific refinements would be:
  - Integrate Solar Arrays
  - Include all TWTAs and radiator fins
  - Break Spacecraft lump mass into individual components
- Validate the 200 W heater budget by incorporating heaters in the model. This could be coupled with further refinement of the model and include design iteration as required.

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