12-16-2021

# State-of-the-Art of Thermal Control Solutions to Establish a Modular, Multi-Orbit Capable Spacecraft Thermal Management System Design Methodology

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# STATE-OF-THE-ART OF THERMAL CONTROL SOLUTIONS TO ESTABLISH A MODULAR, MULTI-ORBIT CAPABLE SPACECRAFT THERMAL MANAGEMENT SYSTEM DESIGN METHODOLOGY

by

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A thesis submitted in partial fulfillment of the requirements for the degree of Master of Science in Mechanical Engineering at Embry-Riddle Aeronautical University

Embry-Riddle Aeronautical University

Daytona Beach, Florida

December 2021

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This thesis was prepared under the direction of the candidate's Thesis Committee Chair, Dr. Sandra Boetcher, Professor, Daytona Beach Campus, and Thesis Committee Members Dr. Rafael Rodriguez, Professor, Daytona Beach Campus, Dr. Patrick Currier, Professor, Daytona Beach Campus, and has been approved by the Thesis Committee. It was submitted in partial fulfillment of the requirements for the degree of Master of Science in Mechanical Engineering.

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# Acknowledgement

I would like to thank my advisor, Dr. Sandra Boetcher, and my thesis committee members, Dr. Patrick Currier and Dr. Rafael Rodriguez for all of their support along the way both in my degree program and this thesis process.

Additionally, I'd like to thank my mother, Beatriz Aguiar-Consolo, my father, Robert Consolo Sr., my sister, Carina Consolo, Scott and Justin Fichera, and Serafina and Mia Cardenas for their support since the beginning. Your push on me to chase my dream has led me here and I'm forever grateful.

I'd like to also thank Copperline Coffee Cafe for providing a home in which nearly 90% of this thesis and research was created, written, and edited during the COVID-19 era that left many without facilities to work or study.

Lastly, I'd like to thank my fraternity, Phi Delta Theta Florida Mu Chapter, and the numerous friends I've made along the way here because if it wasn't for all of you, I may have never made it through school and kept on the path I'm on today. You all have made an impact in my life and this work is dedicated to all of you.

# Abstract

Today, the exploration and exploitation of space continues to become a more common occurrence. All types of spacecraft (S/C) utilize various types of thermal management solutions to mitigate the effects of thermal loading from the unforgiving vacuum of space. Without an appropriately designed thermal system, components on-board the S/C can experience failure or malfunction due to fluctuations in temperatures either beyond the designed operational parameters or unstable oscillating temperatures.

The purpose of this study is to perform a comprehensive review of technologies available today that are being used for thermal management onboard S/C in addition to investigating the means to analyzing the environment allowing the establishment of a design methodology that would support the development of efficient and effective future spacecraft thermal control systems. A combination of thermal solutions are investigated that would best assist onboard components in maintaining operable thermal ranges. Modern day methods of analyzing and understanding these environments were looked at to provide an insight as to what may be available for both the new and experienced developer.

Analytical methods varied, dependent on a reference point, but the outcomes were similar in that the primary concern of heat loading in space is radiative heating from internal and external sources. Numerically, industry has continued to find new ways of understanding environments prior to launch whether it be through analytical estimation or numerical tools. Thermal control solutions consisted of coatings, insulation, heat pipes, phase change material, conductive materials, thermal devices, actively pumped fluid loops, radiators, and combinations of these systems.

With numerous technologies identified, a series of charts were created to provide comparatives among the various aspects of selection guiding the start of design. Lastly, utilizing the knowledge gained from such a wide-net review of thermal control solutions available today, both in space and terrestrially, a design methodology was established.

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# 1 Introduction

Today, the exploration and exploitation of space continues to become a more common occurrence. Space has become a utility towards the advancement of technologies on Earth, both common use and the highly complex. Additionally, space provides an opportunity to exploit opportunities available only in orbit such as communication, scientific/militaristic observation, weather monitoring, navigation, remote sensing, surveillance, and data-relay services (Gilmore, 2002).

All types of spacecraft (S/C) utilize various types of thermal management solutions to mitigate the effects of thermal loading from the unforgiving vacuum of space. Additionally, internal components generate heat as they dissipate the energy from power required to operate the spacecraft. This power requirement has doubled every 5-6 years as demands of satellite services has increased (Hengeveld, Mathison, Braun, Groll, & Williams, 2010). Thus, the S/C requires additional methods to dissipate this heat and maintain temperatures that will allow components to operate sufficiently.

The future of the industry is showing higher demand for high power and high bandwidth components. For example, Shivakumar (2014) highlights that at the ISRO Satellite Centre, there is a huge demand of high throughput I-6K communication satellites capable of handing 10–15 kW power, high resolution mapping, and observation missions. Additionally there are desires for components like 10 Gbps data transmission systems requiring large bandwidth and processing capability (Shivakumar, 2014). Needs like higher power and higher bandwidth come with higher heat generation and higher heat loading which further exemplifies the need for newer, more efficient thermal control in order for the industry to keep up with advancing technology. Even in the age of early spaceflight, it was known that there is a need for more powerful, reliable, flexible, and efficient thermal control systems on spacecraft (Wise, 1986). As the mission constantly evolves, Wise (1986) describes the thermal design challenge that engineers face such as the need for weight minimization as mission lifetimes continue to expand and are directly correlated to available fuel, with spacecraft life equal to 1 month/kg of $N_2H_4$.

Providing insight into the satellite industry, Kopacz et al. (2020) highlights the newest and fastest growing segment deemed ”Small Satellites” or satellites that are less than 500kg in mass. This subsection, as shown in Figure 1, breaks out into three classifications of satellites: Mini, Micro, and Nano (Kopacz et al., 2020). Small satellite launches from 2000 to 2016 are shown in Figure 2. During this timespan, launches of small satellites increase by nearly 10 times with 2016 seeing the launch of 92 small satellites (Kopacz et al., 2020). Additionally, it can be found that earth-observation small satellites were able to achieve similar ground sampling distances (a measure imagery resolution) as satellites 7 times heavier (Kopacz et al., 2020), which is a testament to increasing technological power on a significantly smaller scale.

| Satellite Classification (N., 1992) | Mass (kg) |
|---|---|
| Large Satellite | > 1000 |
| Medium Satellite | 500 to 1000 |
| Mini Satellite | 100 to 500 |
| Micro Satellite (Cube Sat) | 10 to 100 |
| Nano Satellite | 1 to 10 |

Figure 1: Satellite Classification by Weight (Kopacz et al., 2020)

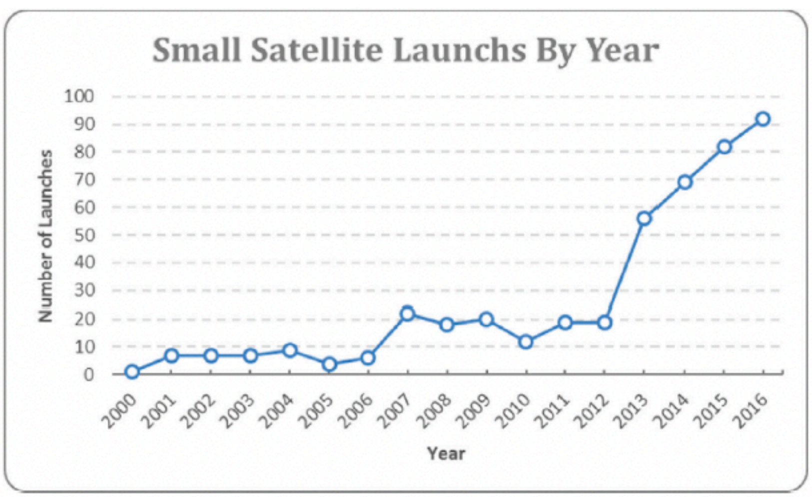


Figure 2: Small Satellite Launches by Year (Kopacz et al., 2020)

The challenge of mitigating thermal loading for S/C by constructing an effective thermal management system is exasperated by numerous additional challenges, including a micro-gravity environment, atmospheric drag, atomic oxygen that can degrade surfaces, a vacuum environment that induces outgassing and cold welding, micrometeoroids, and charged particles (Hengeveld et al., 2010). Swanson and Birur (2003) also identify that emerging trends in spacecraft and instrument design continue to complicate an already very challenging thermal control problem. Drivers of future requirements for future thermal control include (Swanson & Birur, 2003):

1. Dimensional stability of structures
2. Cryogenic heat acquisition and transport
3. Tight temperature control
4. Integrated thermal/mechanical/optical systems
5. Common thermal design for small satellites
6. High heat flux heat acquisition for tight temperature control
7. Challenging thermal sink situations
8. Minimization of mass and auxiliary power use
9. Thermal control of spacecraft in extreme high temperature environments

S/C have two parts: a payload and a bus that supports that payload. S/C buses are traditionally the structural component to a S/C and contain numerous subsystems that will ensure the payload operates throughout the mission such as the thermal control system (TCS). Zanoni et al. (2016) reminds us that thermal control is not only important for the craft itself but plays a critical role in the actual execution of experiments. Certain experiments, like the one Zanoni et al. (2016) was working on, required vacuum conditions with low temperature environments to appropriately execute, so they decided to use thermal shielding which is the concept where shields are built on the vehicle that would physically block solar radiation from reaching the critical components thus passively keeping them cool. The TCS is designed to maintain S/C and component temperatures within operational thresholds by balancing energy input and energy generation utilizing energy storage and energy dissipation (Hengeveld et al., 2010). This is represented accordingly by Equation 1 below.

$$\dot{E}_{in} + \dot{E}_{gen} = \dot{E}_{stored} + \dot{E}_{out} \tag{1}$$

With the thermal requirements of spacecraft primarily based around the operating ranges of its components, temperatures onboard should remain between -15$°C$ and 50$°C$ for electronics, 0$°C$ and 20$°C$ for rechargeable batteries, and between 0$°C$ and 50$°C$ for mechanisms like gyroscopes, momentum wheels, solar array driving motor, etc (Alcayde et al., 2021). To make the best choices for a passive thermal control system, it is important to be understand the degradation characteristics of the components as well. Mason (1988) took a look at the Infrared Astronomical Satellite mission over a period of three years and evaluated the state of its systems in orbit. During this period, an exterior temperature of 200 K and a shaded temperature of  100 K was recorded and maintained without any significant degradation (Mason, 1988). Additionally, post expenditure of onboard helium cryogen supply, the satellite remained at 100 K with no degradation in performance (Mason, 1988).

Without an appropriately designed thermal system, components on-board the S/C can experience failure or malfunction due to fluctuations in temperatures either beyond the designed operational parameters or unstable oscillating temperatures which can be due to external factors, internal factors, or a combination of the two. Active and passive solutions can provide the ability to moderate heat and can be used in combination with each other to provide greater protection from larger loading. For external loading, passive solutions are typically easiest to implement for S/C and include thermal insulation and reflective material coating to the vehicle exterior.

A thorough analysis and integrated protections from the environment provides the opportunity for thermal engineers to adequately ensure that heat flux is negated from entering through the exterior of the vehicle and treat exterior walls as adiabatic when evaluating thermal loading. With both environments separated, the problem sets are also separated, having to mitigate the thermal effects of heat dissipation from the components themselves independently. Each component produces heat as it operates and with extensive insulation creating an adiabatic external wall will also prevent heat from being emitted from the vehicle, making it important to have systems in place that can dissipate and reject the heat from the interior as well.

Passive methods that are available to mitigate internal loading include phase change material (PCM), heat pipes/sinks, and/or a combination of these through conductive heat transfer. For internal thermal loading, a more active thermal management approach may also be taken, using mechanically pumped single-phase or two-phase fluid loops, throttling on-board components to manage heat dissipation, and/or rotating the S/C to attain an even distribution of heat exposure across the entire exterior of the S/C. All methods are in conjunction with a radiator that would take the transported heat and reject that heat to space.

Even with the utilization of active systems, passive thermal solutions may be required to dissipate the heat toward the active solution and sufficiently permit the actual removal of heat. Additionally, PCMs provide for the opportunity to not only remove heat but to actually retain that energy, a phenomena known as thermal storage, which can be extremely helpful for systems in orbit that are going to experience volatile swings in temperature ranges throughout their orbit (Collette et al., 2011).

With many different modalities available for thermal management, its difficult to decide what combination of thermal management solutions allow for the most optimal performance of the S/C. Design parameters are not readily standardized and environments vary greatly due to a multitude of factors. This can lead to it being difficult to evaluate what environments the vehicle and its components will be exposed to, making it extremely difficult to conclude which modality would be optimal and could lead to over-design, under-design, and/or additional incurred cost overall.

With the three most common near earth orbits, as seen in Figure 3 in order of increasing altitude, Low Earth Orbit (LEO), Molniya Orbit, and Geostationary Orbit (GTO) (Gilmore, 2002), becoming more frequently traveled, it would be appropriate to focus on these environments to establish a standardized design constraint or best practice that would guide the future designs of spacecraft thermal management. Orbital mechanics directly determine the thermal exposure that a S/C may be exposed to so its important to understand them thoroughly. Low Earth Orbit, or LEO, are less than approximately 2000km in altitude and generally have the shortest orbital period typically on the order of about 90 minutes (Gilmore, 2002). This orbit can vary in eccentricity and inclination, limited only by the fact that the orbit is not much larger than Earth's diameter (Gilmore, 2002). Molniya orbits are highly elliptical with apogee, or highest point of orbit, of about 38,900km, a perigee, or lowest point in orbit, of about 550km, and are highly inclined at about 62°(Gilmore, 2002). Satellites in this orbit travel very slow near apogee and thus can provide coverage of the northern hemisphere for up to 8 hours of its 12 hour orbital period (Gilmore, 2002). Lastly, geosynchronous orbit are circular, have very low inclinations (< 10°), and operate at an altitude of about 35,786km (Gilmore, 2002). This orbit provides a orbital period matching the Earth's rotation which allows a satellite to maintain a position over a specific point of Earth for as long as desired providing opportunities for weather observation, communication, and surveillance (Gilmore, 2002).

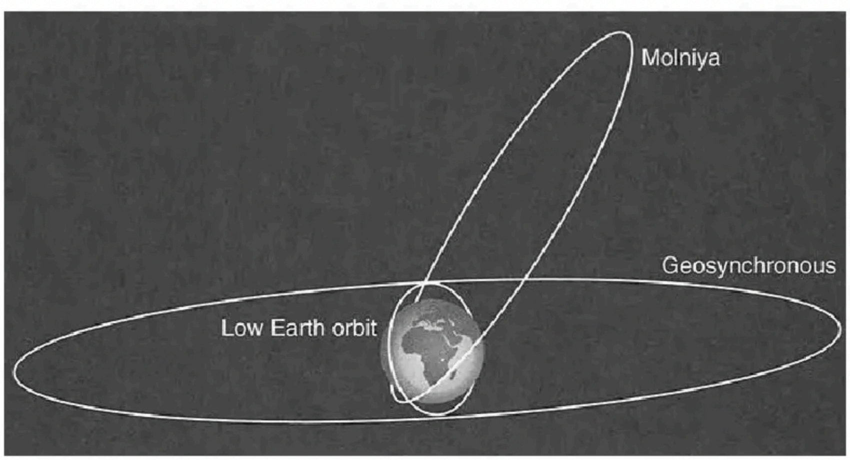


Figure 3: Orbit Types (Gilmore, 2002)

The development of a good design methodology can allow for a streamlining of the design process, simplifying the problem for smaller providers, and minimizing future unnecessary mistakes leading to operational failures. This study intends to answer the following:

1. What methods of analyzing or understanding environments are currently available or being used in industry?

2. What thermal management technologies, both those with flight heritage and those that are new innovations, are available today, and how do they perform under both extreme ends of the thermal loading?

3. What would be an appropriate design methodology when designing a thermal control system for a spacecraft?

The purpose of this study is to perform a comprehensive review of technologies available today that are being used for thermal management onboard S/C in addition to investigating the means to analyzing the environment and making the determination on a design methodology that would support the development of efficient and effective future spacecraft thermal control systems. A combination of active and passive thermal solutions are investigated that would best assist onboard components in maintaining operable thermal ranges in these varied environments. Modern day methods of analyzing and understanding these environments were looked at as to provide an insight as to what may be available for both the new and experienced developer.

# 2 Review of Thermal Environment Analysis

## 2.1 Orbital Thermal Environmental Factors

As established in Equation 1, thermal control is the act of balancing between energy being generated or absorbed and the energy being rejected or stored. The energy input, $\dot{E}_{in}$, is composed of direct solar flux ($q''_{sol}$), albedo flux ($q''_{alb}$), and outgoing longwave radiation ($q''_{OLR}$) (Hengeveld et al., 2010) as highlighted previously. Other energy inputs are insignificant in impact and are not included in this analysis (Gilmore, 2002). The total rate of flux input is summarized in the equation below.

$$q''_{external} = q''_{sol} + q''_{alb} + q''_{OLR} \tag{2}$$

According to Anderson, Justus, and Batts (2001), direct solar flux is the greatest source of heating for most spacecraft. In addition to the solar flux, manufacturers need to account for albedo, or the fraction of incident solar energy that is reflected of earth and back in to space, and for outgoing longwave radiation (OLR), which is radiation that has been previously absorbed by earth and then emitted back out to space (Anderson et al., 2001). Albedo is highly variable and is dependent on several factors ranging from cloud cover to the distribution of reflective properties of the surface (Anderson et al., 2001). Additionally, OLR is not constant either but variations are much less severe than albedo with the primary influencers of OLR being temperature of the earths surface and the amount of cloud cover. It is generally sufficient to assume a graybody spectrum corresponding to a temperature in the 250 to 300K range for OLR. Figure 4 depicts this problem set graphically.

Alcayde et al. (2021) presents the problem of spacecraft (S/C) thermal environments similarly to Anderson et al. (2001) by separating it into its individual components. With this knowledge, its important to note that the problem here is contingent on a S/C's orbital period. This is

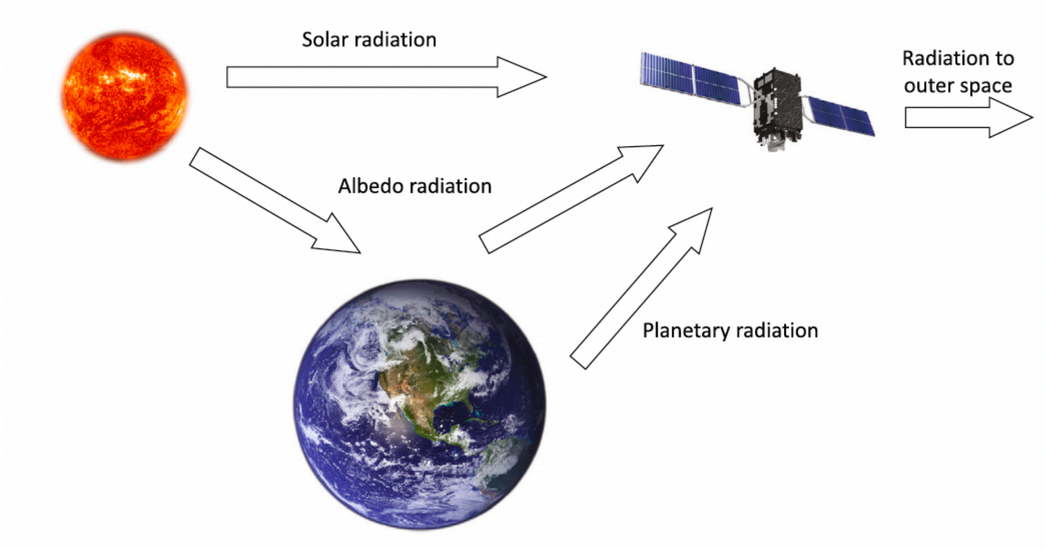


Figure 4: Spacecraft External Thermal Environment (Alcayde et al., 2021)

because there are two operational situations in which the S/C is under the effect of the sum of different variables (Alcayde et al., 2021), the period in which the S/C is in the sun, and when it is in the shadow of the body it orbits around, as shown in Figure 5. It is broken down simply as:

**Sunlight Zone:** Radiation Received = Solar rad. + Albedo rad. + Planetary rad.

**Shadow Zone:** Radiation Received = Planetary rad.

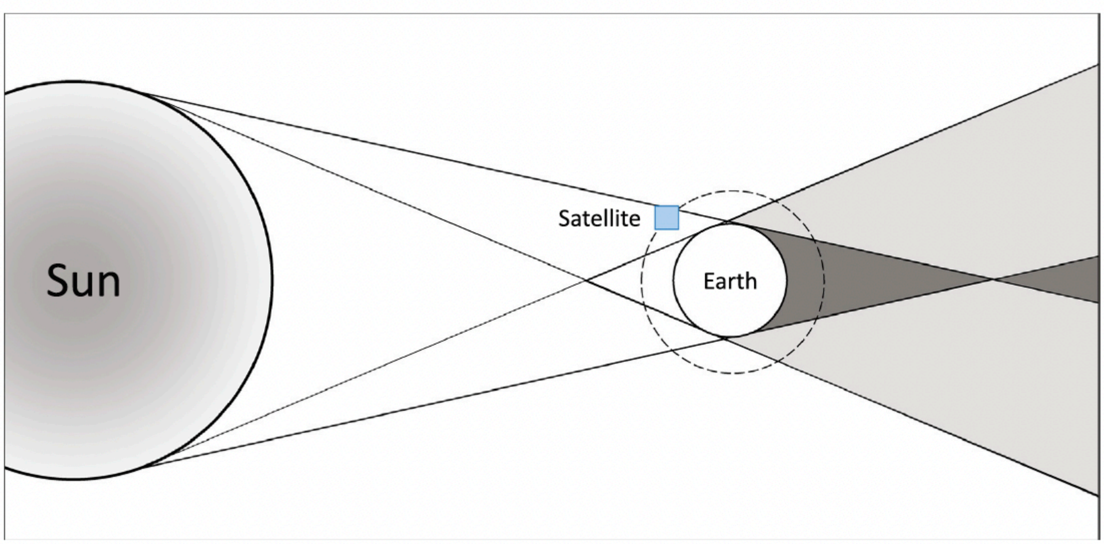


Figure 5: Spacecraft External Thermal Environment (Alcayde et al., 2021)

### 2.1.1 Solar Flux

Corpino, Caldera, Nichele, Masoero, and Viola (2015), utilizing the PiCPoT nanosatellite as the reference mission, conducted an analysis to develop a temperature profile of the satellite under dynamic conditions to evaluate the vehicles compliance with its component operative limits. Corpino et al. (2015) defines the external boundary conditions of the spacecraft by the position of the Earth on the ecliptic (epoch) and by the external radiation sources such as direct solar radiation, albedo, and the Earth infrared (IR) radiation. The direct solar radiation varies based upon the solar constant which is the intensity of sunlight radiation perpendicular to a surface at the Earths mean distance from the Sun (1 AU).

During the development of the International Space Station, a baseline set of thermal range parameters were developed to guide designs of components and thermal systems, but in 1992 these parameters came under review as lack of data made these parameters overly conservative, leading to design difficulties that were being experienced in the winter solstice (”Hot Case”) and summer solstice (”Cold Case”) environment parameters (Anderson et al., 2001). With the success of the Earth Radiation Budget Experiment (ERBE), bringing in extensive data on the environmental parameters, NASA created the ”Guidelines for the Selection of Near-Earth Thermal Environment Parameters for Spacecraft Design” (Anderson et al., 2001). These values, not including the solar cycle variation, are as follows (Anderson et al., 2001):

**Hot Case:** $S_{Hot} = 1414 W/m^2$

**Mean Case:** $S_{Avg} = 1367 W/m^2 =$ Solar Constant

**Cold Case:** $S_{Cold} = 1322 W/m^2$

While the solar constant, the average solar flux at the average Earth-Sun distance, is meant to be a constant in calculations, it does vary about 3.4% over an orbit. This value trends higher leading up to and following the winter solstice and an inverse trend leading up to and following the summer solstice due to its slightly elliptical orbit. An additional $\pm$ 5 $W/m^2$ could be used to account for solar cycle variations and uncertainties in the measurements that these values are based on, made from 1969 to 1980 (Anderson et al., 2001).

Xie, Gao, Wu, and Qin (2016) conducted research into the thermal conditions that the main radiators of the Alpha Magnetic Spectrometer are exposed to while attached to the International Space Station (ISS). The ISS flies at an altitude between 370km and 460km and when modeled, it is seen that both radiators exhibit, when fully illuminated, an solar flux between 1367 $W/m^2$ and 0 $W/m^2$.

#### 2.1.2 Albedo Flux and Planetary Radiation

Albedo values are typically shown as a fraction or a percent and is known to be highly variable, dependent on the distribution of reflective properties of the surface, and the amount/type of cloud cover with reflectivity increasing with increased cloud cover (Anderson et al., 2001). ”Albedo radiation has approximately the same spectral shape as the Sun’s spectrum which approximates a blackbody with a characteristic temperature of 5777K” (Anderson et al., 2001). All planets in our solar system exhibit temperatures greater than zero, and thus, radiate heat into space. Planetary radiation or in the context of our problem, the outgoing longwave radiation emitted by Earth, is a combination of radiation emitted by atmospheric gases and radiation emitted by the Earth’s surface and cloud tops but is partially absorbed in the atmosphere (Anderson et al., 2001). For the purpose of spacecraft thermal analysis, it is generally sufficient to assume a graybody spectrum providing a temperature of approximately 250 to 300K (Anderson et al., 2001). Planetary radiation is varied by the emittance of the planet in question in addition to your crafts distance from the emitting surface.

Gilmore (2002) included two tables detailing the albedo and Earth emitted IR that may be experienced at various orbital inclinations. Gilmore (2002) does specify that one table takes a more conservative, 3.3 standard deviations approach ($3.3\sigma$) to the analysis providing values that would only be exceeded 0.04 percent of the time while the second provides less conservative, 2 standard deviations values ($2\sigma$) that could be exceeded 5 percent of the time. The albedo and IR values recommended in these tables are based on the NASA/Marshall Space Flight Center study that considered 28 data sets of 16-second-resolution sensor data collected monthly from the previously mentioned Earth Radiation Budget Experiment (ERBE) (Gilmore, 2002). These sensors were flown on the ERBE at a low inclination, 610km altitude orbit and on the National Oceanographic and Atmospheric Administration (NOAA) 9 and 10 satellites at high inclination, 849km and 815km altitudes respectively (Gilmore, 2002). The study performed a statistical analysis to identify maximum and minimum albedo and Earth IR heating rates that a S/C would be exposed to in time periods ranging from 16 sec to 24 h (Values were found to not change significantly in periods greater than 24 h) (Gilmore, 2002).

The two tables that these tables present can guide the design of thermal control systems for critical and non-critical systems in a generalized form. These tables can be found in the Appendix under Tables 3, 4, 5, and 6 (Gilmore, 2002).

### 2.1.3 Mathematical Calculation of Near-Earth Thermal Environments

As shown in Equation 2, the heat flux boundary condition for the environment is the combination of solar flux, albedo flux, and the outgoing longwave radiation coming from Earth. This relationship can be approximated for an arbitrary surface facing Earth to include solar absorption ($\alpha$), solar irradiation ($S''$), Earth's radius ($R_{TOA}$), spacecraft altitude ($h$), albedo ($\rho_{alb}$), solar zenith angle ($\Theta$), longwave emissivity ($\varepsilon$), and Earth emitted radiation ($E''$) (Hengeveld et al., 2010). This relationship is depicted in the equation below.

$$q''_{external} \approx \alpha \cdot S'' + \alpha \cdot (\frac{R_{TOA}}{R_{TOA}+h})^2 \cdot \rho_{alb} \cdot \cos(\Theta) + \varepsilon \cdot (\frac{R_{TOA}}{R_{TOA}+h})^2 \cdot E'' \tag{3}$$

This equation is written utilizing the Earth as its reference point but Alcayde et al. (2021) presents the same problem but utilizing the Sun as it's reference point. Changing the reference point, while the equation outlined in Equation 2 remains the same, the means of obtaining the variable does change. Solar radiation ($J_s$) is defined by the power output of the sun (P) known to be 384.6 yotta Watts ($3.846X10^{26}W$) over $4\pi$ multiplied by the distance from the sun (d) squared (Alcayde et al., 2021). Albedo is utilizes the previous solar radiation value, multiplying it by the albedo parameter (a) which is assumed to 0.33 for Earth and the visibility factor (F) which varies with altitude and angle between the orbital plane and Earth-Sun vector (Alcayde et al., 2021). Lastly, planetary radiation, whose calculation drastically varies to the planet in question, is defined by a leading constant for that planet (237 for Earth), multiplied by the radius of the planets effective radiating surface ($R_{rad}$) over the distance of the satellite from the Earth's center ($R_{orbit}$) squared (Alcayde et al., 2021).

Another factor that separates Alcayde et al. (2021) calculation from Equation 3 by Hengeveld et al. (2010), is that it takes into account the specific surface that is exposed to the radiation in question and its area (A). Similarities are that absorptivity and emissivity is appropriately accounted for by the $\alpha$ and $\sigma$ values respectively. This equation can be illustrated as such (Alcayde et al., 2021):

$$q''_{external} \approx \alpha A_{solar} J_s + \alpha A_{albedo} J_a + \sigma A_{planetary} J_p \tag{4}$$

$$q''_{external} \approx \alpha A_{solar} (\frac{P}{4\pi d^2}) + \alpha A_{albedo} (J_s a F) + \sigma A_{planetary} (237(\frac{R_{rad}}{R_{orbit}})^2) \tag{5}$$

### 2.1.4 Numerical Analysis Tools for Spacecraft Thermal Environments

To most appropriately design a multi-mission platform, an understanding of the orbital parameters to which you will operate in help determine a multitude of variables that will play a critical role in your thermal environment calculations. For observation satellites, whether observing Earth or any other body in space, your orbit will be pertinent to being able to appropriately observe your target site(s) and depend on the bodies geometry as well as the orientation of the spacecraft. SPICE or Spacecraft, Planet, Instrument, Camera-matrix, Events program allows users an opportunity to compute the observation geometry of their robotic flight providing quantities of interest such as altitude, latitude and longitude, and lighting angles where an instrument field-of-view intercepts a surface (Acton, Bachman, Semenov, & Wright, 2018). These values output by the program could be repurposed to help derive values for external environments and, for uncontained/open spacecraft buses, help determine the external influence on component heating Acton et al. (2018)

Efforts have been taken to try and further improve the ability to numerically analyze what S/C may experience in an orbital period. Being that the heat transfer problem at hand is fully dependent on the orbital mechanics of the S/C, and that a thorough analysis requires a multi-nodal system, this part of the design process can be a timely and computationally expensive processes. This problem extends to the analysis of individual components as well. Anh et al. (2016) utilizes the method of equivalent linearization to the single-nodal model, differential nonlinear equation, of the heat transfer of a small satellite in Low Earth Orbit (LEO). Various approaches are used and find the temperature responses found using linearization and Grande's approach are very close to those obtained by the fourth order Runge-Kutta algorithm as shown in Figure 6. An important note here is while this method does a great job at obtaining a majority of the results, the outliers are still unaccounted for and those maximum and minimum outliers can be the thermal loading values that could cripple a system.

To best design thermal control systems for use in orbit, it is important to have accurate testing conducted here on the ground which can be difficult due to the adverse environment present. Simulated heat flux measurements are a primary aspect to spacecraft thermal tests and require highly accurate and calibrated heat flux measurement tools (Sheng, Hu, & Cheng, 2016). For smaller satellites, a traditional thermal test with steady state condition is inappropriate because of the smaller thermal inertia compared to a larger spacecraft (Sheng et al., 2016). A transient heat flux experiment is required than and this requires a transient heat flux meter with high ac-

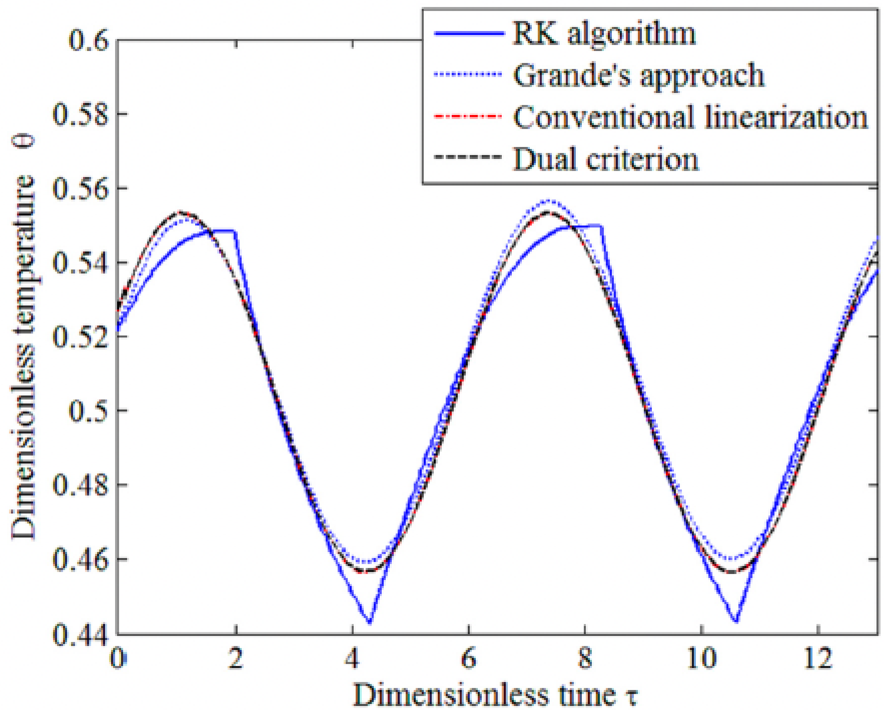


Figure 6: Temperature Variation Using Various Numerical Methods (Anh et al., 2016)

curacy and fast response time (Sheng et al., 2016). Sheng et al. (2016) present a novel transient radiative heat flux meter than can be used to measure up to 1400 $W/m^2$ with a 5% uncertainty and response time of 10 s that provides smaller provider to more adequately test and design their on-orbit systems.

Delkov, Kishkin, Lavrov, and Tanasienko (2016) found that most mathematical models are considered only from the standpoint of heat transfer and do not appropriately take in to consideration the hydraulic component. Determining the flow of the working medium in the elements as a reverse cycle system may exhibit greater suitability for flight because condensation processes in both cycles occur at different pressures and different temperatures playing directly into the rate of radiation at the radiator which is dependent on the temperature of the surface of the radiator to the fourth degree (Delkov et al., 2016). González-Bárcena, Fernández-Soler, Pérez-Grande, and Sanz-Andrés (2020) utilized real data obtained via atmospheric soundings and radar/satellite to provide further definition to the thermal environments experienced by balloons in the ascent phase on research missions. While the environment is different as these missions remain contained within the atmosphere, the definitions of various thermal variables as well as the cooling effects experienced could reflect on conditions the may be experienced in orbit and the calculations that must be conducted to attain an appropriate value.

With the extreme cost associated with launching anything to space, it can at times be impossible to afford hardened components to survive the environments in orbit thus driving the extreme importance of a thorough and complete thermal analysis. Kovács and Józsa (2018) conducted a thermal analysis on a satellite with a similar story. The SMOG-1 satellite developed by the Budapest University of Technology and Economics that will be flying in a sun-synchronous orbit at no higher than 600km, an inclination of 98° to ensure an acceptable orbit life and reduce the amount of time it will be space debris as it will use common commercial electronics in an effort to reduce cost. After the setup of a thermal network model, it was found that the minimum and maximum values of solar flux vary to the point in the orbit between 1322 and 1414$W/m^2$. Even with placing the battery at the center of the frame and insulating it, the analysis still showed it violating its tight charging operational range of 0 to +45$°C$ but maintains above the bottom boundary of its discharge operational range of -10$°C$ which provides the requirement of intermittent charging of the battery (Kovács & Józsa, 2018). This reinforces the idea that batteries are typically the tightest thermal requirement on systems and can cascade to larger issues if not addressed as loss of power will likely lead to mission failure.

”The complexity of the mathematical models used for simulation of the indicated processes, the large cost of the corresponding thermal experiments and tests, and the known limitations of the traditional methods used for processing and analysis of the results of thermophysical experiments have made pressing the problem on the development of new methods and means for obtaining a maximum body of data on the characteristics of a thermal system with the use of results of reliable inexpensive experiments.” (Budnik, Nenarokomov, & Titov, 2018). Budnik et al. (2018), acknowledging this, presented the thermophysical properties of their version of multi-layer insulation utilizing a new means of analysis. In ground based testing, with exposure to temperatures between 20-450$^{\circ}C$, showed that their process of testing experimentally when compared to their numerical calculations had a fairly small discrepancy providing evidence of the accuracy of their estimates (Budnik et al., 2018).

Delkov et al. (2016) continued that programs like ANSYS, Thermica, ESATAN-TMS, SINDA, Radsol, and other analysis programs suffer from a number of issues, including but not limited to high commercial cost, that it is not always possible to integrate the systems with external applications, that it is not possible to include the fluid dynamics features of the circuit for optimization and modeling, and the considerable time needed to perform the calculations. This instills a requirement to develop mathematical models that address these concerns and is designed to compare the energy characteristics of different types of thermal control systems (Delkov et al., 2016). This drive to establish independent mathematical models to numerically analyze systems has led to numerous independently sourced analysis tools that individuals and industry leaders utilize in-house but also leads to many variations in how the problems are analyzed and which variables are considered or take precedence over others.

While this does bring a level of uncertainty to many analysis results, it does also spur a level of innovation to the means of analyzing these systems. An example of such is an analysis and prediction software constructed by Guoliang and Guiqing (2004) for the means of reentry thermal conditions which, while a different problem set, could be applied to radiative conditions of open space. In order to ensure a safe reentry vehicle, conservative designs are usually selected. With consideration of the high-performance and low-cost requirements imposed on future space vehicle development, the conservative design is no longer acceptable for the vehicle general design. In order to improve the performance prediction and the design level of the heat protection, Guoliang and Guiqing (2004) have developed a comprehensive analysis and estimation system to show the history of thermal variation consisting of a numerical simulation, database, expert prediction, and image display Guoliang and Guiqing (2004).

To reduce the number of elements of S/C thermal models, a matrix method was developed by Fernández-Rico, Pérez-Grande, Sanz-Andres, Torralbo, and Woch (2016) based on the lumped parameter method. This method intended to produce a sufficient thermal model reduction for steady-state cases without reducing the main characteristics of the model. It has proven to be fast, intuitive, and maintain all the physical characteristics of the original model (Fernández-Rico et al., 2016) when compared to a more detailed model. Further pursuit of a more accurate and readily available thermal analysis tool for space environments, Reyes et al. (2020) conducted an analysis of a thermal coating utilizing an in-house constructed tool and compared to a couple of commercially available tools. The developed code is built to solve the energy balance for the critical conditions in both steady and transient states, and after use, showed agreement with commercially developed tools but did produce a more conservative estimate. Additionally, when compared to experimental data, errors of 5.4% and 4.4% were shown when estimating the maximum and minimum temperatures respectively (Reyes et al., 2020). Something to consider with spacecraft thermal analysis is that when attempting to determine the values of the parameters in the mathematical model, to reach a good fit between it and and the experimental test data, is frequently done manually due to the expertise required to make an appropriate compromise (Torralbo, Perez-Grande, Sanz-Andres, & Piqueras, 2018).

Very often, a numerical simulation of heat transfer processes on spacecraft must utilize a heat transfer mathematical model with lumped parameters but this brings the added difficulty of determining the coefficients of the model providing adequacy (Nenarokomov, Alifanov, Krainova, Titov, & Morzhukhina, 2019). Direct measurement is typically impossible and estimates aren't very accurate and are more frequently contradictory. Thus, a requirement is created to determine these values via calculations and experimentally. Nenarokomov et al. (2019) present a theory to experimentally identify absorptivity and emissivity in the real mode of operation and use these results to create long time predictions of radiative properties of materials.

An added level of difficulty comes from the very niche nature of nearly every mission. With that acknowledged, trying to capture as many possibilities or mission capabilities can lead to simplifications of the thermal problem. The problem lies in that analytical analysis is essential at an early stage of design, so Pérez-Grande, Sanz-Andrés, Guerra, and Alonso (2009) presented a simple analytical method to understand temperature variations and thermal stability of small compact spinning satellites. Due to the effect of spinning, the outer temperatures of the spacecraft are uniform, so this analysis can be conducted with just two nodes, one being the outer and one being the inner that includes all equipment within it. To be able to obtain general results that apply beyond this niche case, the equations are reduced to a second order non-homogeneous differential equation for the inner node temperature fluctuation.

An additional example of niche analysis is seen by Smirnov, Ivashnyov, Nerchenko, and Kazakova (2011) as they analyze the temperatures of the gas interior to the capsule surrounding the various experiments. This gas serves as the thermal control system, absorbing the heat emiited by the containers and transfers it to space. Utilizing a method of semi-permeable bodies, the containers reach a temperature difference of 30K and the developed model for this analysis can be used for predicting thermal conditions in new unmanned missions for microgravity experiments (Smirnov et al., 2011).

Pérez-Grande et al. (2009) makes the point of considering the thermal problem early on in the design process is highly valuable because your thermal tolerances will play an integral role in your operational parameters, potentially further determining how and when you will conduct maneuvers for example. Racca (1995) highlights this by calculating thermal characteristics of the lunar surface to determine the surface heat flux that may affect a lunar orbiting spacecraft. This can drive orbital considerations due to power and thermal constraints, for example, Racca (1995) states this when speaking to eclipse cases that put heavy strain on systems: ”Since the times of the eclipse events are well known the orbital maneuvers could be planned well in advance and therefore the best situation from the thermal and power point of view could be selected.” (Racca, 1995).

A very promising recent development in analysis capabilities stems from the acquisition of AGI, the creators of Satellite Tool Kit (STK), by ANSYS bringing an exciting new capability to merge operational information to produce a more accurate analysis of a platform in space throughout the entirety of its mission profile. Presented in a webinar hosted by AGI, an example problem is presented of an observation satellite in a low altitude ($\approx$ 700km) polar orbit that is operating in a notional attitude, meaning that the craft is oriented so the solar panels are always facing the sun through orbit and then oriented straight up in shadow, except when passing over the site of which the craft is attempting to observe as seen in Figure 7. This analysis takes into account all external factors (Solar, Albedo, and OLR) and utilizes the operational information input from STK to most appropriately calculate the constantly varying values of incoming radiation (Ingwersen, 2021).

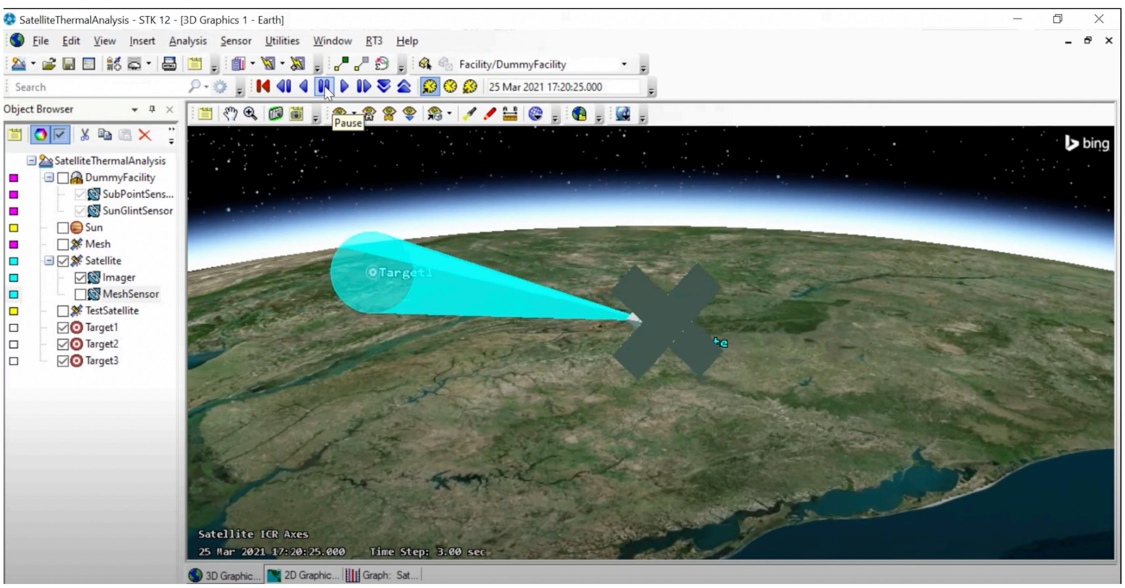


Figure 7: Satellite Tool Kit Satellite Mission Example (Ingwersen, 2021)

By following the steps taken in the webinar by Ingwersen (2021), the user would start by defining the 3-D geometric model in ANSYS workbench, establishing the materials in use and any protection means in place and where. Next, the user defines the mesh which can be generated automatically in the steady state thermal option in workbench as seen in Figure 8. Then, through the use of a custom script, the mesh coordinates are exported to STK, providing the program coordinates for each node and the material and the normal direction of each facet (Ingwersen, 2021). Once in STK, the mesh is imported utilizing a custom script and nodes can be shown graphically as desired on the model.

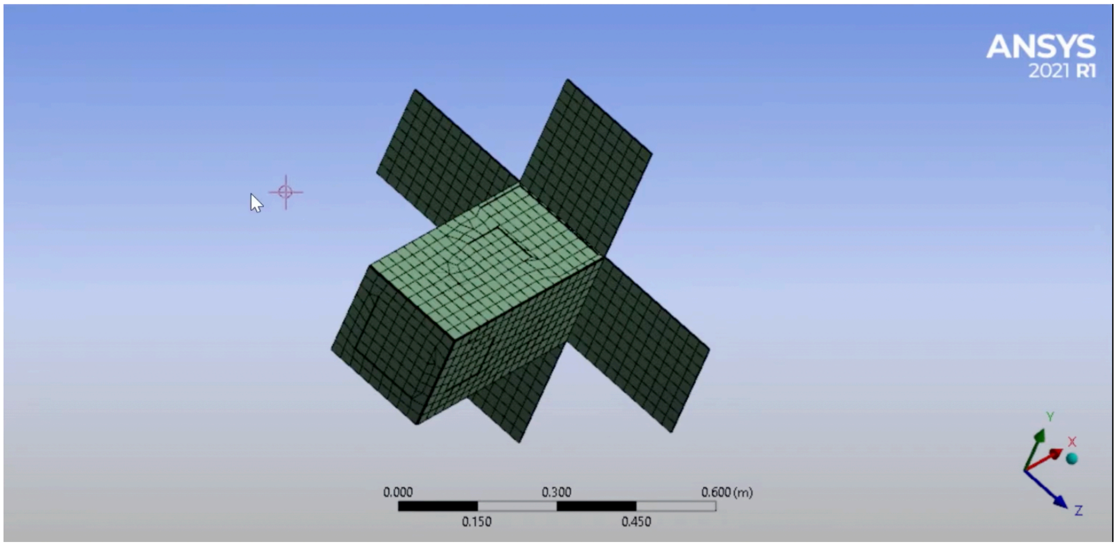


Figure 8: ANSYS Mesh Example (Ingwersen, 2021)

In order to analyze the flux impacts, a sensor object is utilized for each point, which incorporates the shadowing from the objects body and creates a body mask file for it, evaluating flux impacts at each point and exporting these results to a .CSV file using a custom script. These .CSV files contain X, Y, Z coordinates and total energy ($W/m^2$) for each node, and a file is created for each point in time for the orbit. This data can then be read into ANSYS Workbench, opened in mechanical, and a transient thermal analysis can be conducted resulting in temperatures for all nodes at various phases in flight as shown by Figure 9. The solver files output can then be brought back to STK to show temperatures throughout the mission profile in it's various phases of flight as shown by Figure 10.

The area of thermal analysis in space is an ever growing field with a lot of room for innovation and cost optimization, both computationally and economically. The ANSYS acquisition of AGI and the coworking capability between their two programs providing this new capability to combine a premier mission operational parameter platform with a well known and vetted thermal

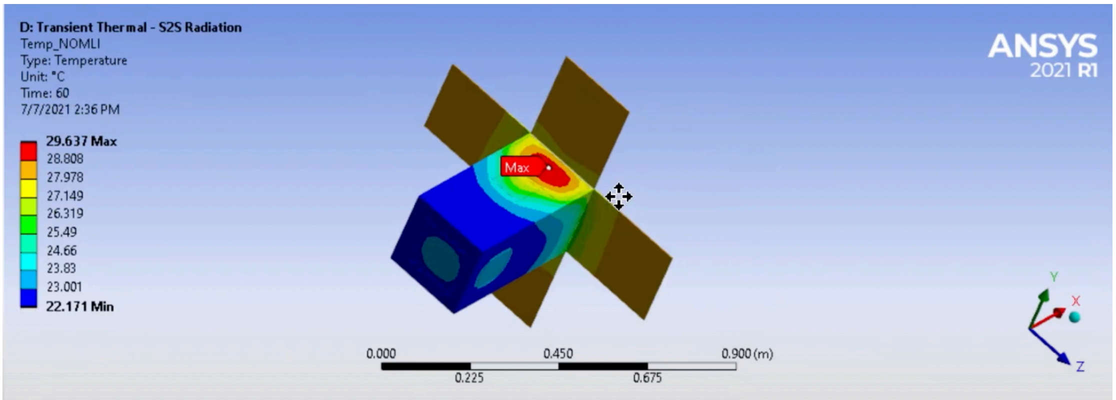


Figure 9: ANSYS Transient Thermal Analysis Solver Example (Ingwersen, 2021)

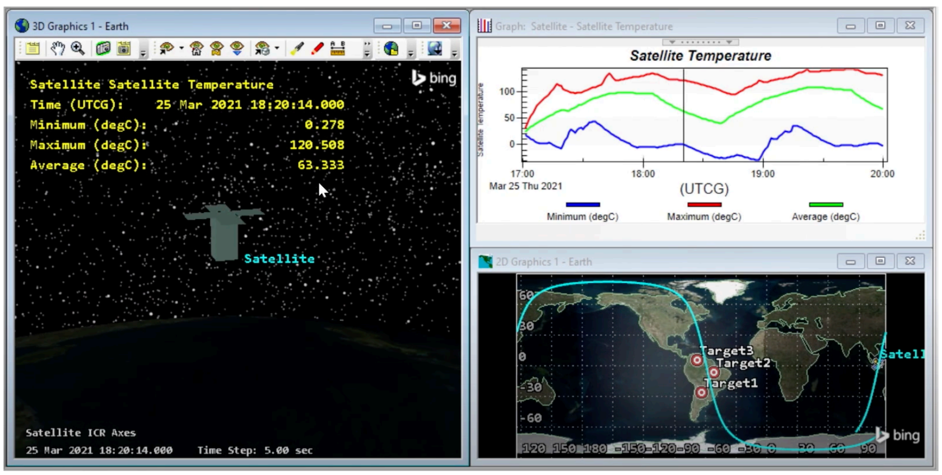


Figure 10: STK Integration of Solver Data Example (Ingwersen, 2021)

analysis software provides a new means to companies that may already operate utilizing either one or both of these program already.

# 3 State-of-the-Art of Spacecraft Thermal Control

J. Wang, Li, et al. (2021) defines a thermal management system (TMS) as ”...a subsystem for a spacecraft which maintains on-board thermal properties such as temperature, temperature difference, and humidity within the design requirement.” Additionally, parallels are drawn between aviation and aerospace, such as the fuel cooling loop of an airplane which can be seen as similar to the single phase mechanically pumped fluid loop that has been utilized in many spacecraft as an active liquid-based TMS (J. Wang, Li, et al., 2021).

Understanding that there are extreme environmental challenges that come with space exploration, Kahn et al. (2017) describe some of the key technical drivers where the environmental constraints drove the design of the system. The Sylph concept, a Europa satellite, and NEAScout, a craft built for a fly-by of a near earth asteroid, highlights size constraints, radiation challenges, and deployment challenges. Smallsats have to be fully capable of completing their mission objectives yet required to fit within limited mass and volume allocations either due to the small size of the core S/C bus or due to the restrictions of being a secondary payload on a launch vehicle. (Kahn et al., 2017).

Corpino et al. (2015) states that thermal control can be achieved via either active or passive methods but passive tends to be a more ”widespread option” which is due to its reduced complexity, weight, and cost. When speaking to methodology, it is highlighted that passive control ”relies mainly on the definition of the architecture and layout of the satellite, and makes use of materials, coatings, or surface finishes, thermal insulation and heat sinks...” (Corpino et al., 2015).

Being able to conduct a thorough and detailed review of the various system solutions that exist today will provide for an all-encompassing capture of potential options that commercial and government entities have access to in the design of their S/C. This wide-net review of systems will require the study to critically evaluate each system for its applicability in spaceflight, considering factors such as effectiveness, integration to the S/C, mass and sizing, and cost impacts. This evaluation should not be limited strictly to aerospace solutions but rather, take ideas and designs utilized in terrestrial applications as well, ensuring the study does not overlook potential alternative solutions that industry may not have on hand at this moment.

Both active and passive thermal management solutions have each equally gone through an evolution over the past few years. With many small satellite developers utilizing off-the-shelf components, that may not necessarily be rated for the harsh environment of space, have to take higher scrutiny of the thermal conditions of their S/C to avoid failure modes. Taking lessons learned from industry and utilizing new, innovative technologies can assist in ensuring the success of these missions while still maintained reduced cost and mass.

Additionally, this review shall not be focused strictly on solutions that possess flight heritage. New innovations is a significantly important component as the most recent relevant review was completed in 2010 that conducted a detailed investigation of insulation, variable heat-rejection surfaces, heat switches, high-conductivity materials, mechanically-pumped-fluid loops, heat pipes, and heat-pumping technologies. (Hengeveld et al., 2010). Recent developments have brought about technology such as phase-change material (PCM) as a new modality, advancements on the previous technology such as high-conductivity materials, and the combination of previous technology and the newly introduced PCM to create hybrid solutions that may be strictly passive or the combination of active and passive systems.

## 3.1 External Thermal Control

### 3.1.1 Thermal Coatings

A potentially overlooked aspect to the design of a spacecraft are the surface qualities. ”In the absence of atmosphere, the temperature of a spacecraft is controlled by the optical properties of its components.” (A. K. Sharma, 2005). Additionally, with the use of thermal coatings, insulation, and other surface layer protections for spacecraft, the surface qualities could impact the effectivity of each (A. K. Sharma, 2005). Passive thermal systems that are used to protect the exterior of the spacecraft are exposed to harsh elements such as ultraviolet irradiation, particle irradiation, and atomic oxygen causing physical damage and reduced effectiveness from degradation (A. Sharma & Sridhara, 2012). To better understand this interaction a long term radiation test simulating 3 years in orbit was performed, finding the degradation of solar absorptance and emissivity of various thermo-optical materials and found that while emissivity changes were negligible while solar absorptance did experience some negative changes due to degradation (A. Sharma & Sridhara, 2012).

By utilizing a derivation of Equation 5, Alcayde et al. (2021) completed a transient analysis which takes into account the time, the vehicle movement, and the temperature evolution over the mission time-frame. This provides the corresponding radiations dependent on whether the S/C stands in the sunlight or shadow and utilizes that information, through a MATLAB tool, to analyze multiple coating materials performance in temperatures in a whole orbit of the Earth. A list of those materials identified as potentially suitable for spaceflight are found in Table 1. It was observed that materials with high values of absorptance would stabilize at higher temperatures and vice versa for those materials with lower absorptance values (Alcayde et al., 2021). Taking into consideration the operating range for components, with the tightest range being the rechargeable batteries at 0° to 20°, the materials based on black coatings best accommodated the S/C in question (Alcayde et al., 2021).

With external power generating components like solar arrays, these are expected to capture radiation as part of its operation which will naturally influence its temperature. A proper thermal management system is required to guarantee the cells can operate in acceptable efficiency range. To accommodate this, Bianco et al. (2015) proposed a new method of thermal regulating solar cells using polyimide foam and analyzes them comparatively to external coating and multilayer insulation as seen in Figure 11. Results of the thermal analysis showed that the tempera-

Table 1: Spacecraft Coating Materials (Alcayde et al., 2021)

| Material | Absorptance, $\alpha$ | Emittance, $\epsilon$ | Absortion coefficient, $\alpha/\epsilon$ |
|---|---|---|---|
| | Optical solar reflectors | | |
| Silvered fused silica | 0,07 | 0,8 | 0,0875 |
| Indium - Tin - Oxide (ITO) | 0,07 | 0,76 | 0,0921 |
| Aluminized Teflon (0,5 mm) | 0,14 | 0,4 | 0,35 |
| Aluminized Teflon (10 mm) | 0,15 | 0,85 | 0,1765 |
| Silvered Teflon (2 mm) | 0,08 | 0,68 | 0,1176 |
| Silvered Teflon (10 mm) | 0,09 | 0,88 | 0,1023 |
| | Black coatings | | |
| Catalac black paint | 0,96 | 0,88 | 1,091 |
| Delrin black plastic | 0,96 | 0,87 | 1,1034 |
| Martin black velvet paint | 0,91 | 0,94 | 0,9681 |
| Parsons black paint | 0,98 | 0,91 | 1,0769 |
| Vel-black | 0,99 | 0,95 | 1,0421 |
| | White coatings | | |
| Barium sulphate with polyvinyl alcohol | 0,06 | 0,91 | 0,0659 |
| Catalac white plastic | 0,24 | 0,9 | 0,267 |
| NASA/GSFC NS-74 White paint | 0,17 | 0,92 | 0,267 |
| Magnesium oxide aluminium oxide paint | 0,09 | 0,92 | 0,0978 |
| White polyurethane paint | 0,27 | 0,84 | 0,3214 |
| | Anodized aluminium samples | | |
| Anodized aluminium black | 0,76 | 0,88 | 0,8636 |
| Anodized aluminium blue | 0,60 | 0,88 | 0,6816 |
| Anodized aluminium chromic | 0,44 | 0,56 | 0,7857 |
| Anodized aluminium gold | 0,48 | 0,82 | 0,5854 |
| Anodized aluminium red | 0,57 | 0,88 | 0,6477 |
| Anodized aluminium yellow | 0,47 | 0,87 | 0,5402 |
| | Film and tapes | | |
| Aluminium tape | 0,21 | 0,04 | 5,25 |
| Aluminized aclar film (1 mm) | 0,12 | 0,54 | 0,22 |
| Aluminized kapton (aluminium outside) | 0,14 | 0,05 | 2,8 |
| Goldized kapton (gold outside) | 0,25 | 0,02 | 12,5 |
| | Metals | | |
| Buffed aluminium | 0,16 | 0,03 | 5,33 |
| Buffed copper | 0,30 | 0,03 | 10 |
| Polished aluminium | 0,24 | 0,08 | 3 |
| Polished Beryllium | 0,44 | 0,01 | 44 |
| Polished Gold | 0,30 | 0,05 | 6 |
| Polished Silver | 0,04 | 0,02 | 2 |
| Polished Stainless steel | 0,42 | 0,11 | 3,818 |
| Polished Tungsten | 0,44 | 0,03 | 14,67 |
| | Vapor-deposited coatings | | |
| Aluminium | 0,08 | 0,02 | 4 |
| Gold | 0,19 | 0,02 | 9,5 |
| Silver | 0,04 | 0,02 | 2 |
| Titanium | 0,51 | 0,12 | 4,33 |
| Tungsten | 0,60 | 0,27 | 2,22 |
| Polished Silver | 0,04 | 0,02 | 2 |
| | Solar cells | | |
| Galium arsenide-based solar cells | 0,88 | 0,80 | 1,1 |
| Crystalline silicon-based solar cells | 0,75 | 0,82 | 0,915 |

ture reached by the polyimide foam is suitable for operation of the solar cell while also reducing weight of the system (Bianco et al., 2015).

Decreasing size of satellite platforms has led to thermal control issues because of the high surface-to-volume ratios and a much smaller thermal mass which makes these platforms much more vulnerable to rapid temperature fluctuations. Böhnke et al. (2008) investigates the use of a coating with high thermal emissivity on top of a layer with high reflectivity and experimentally

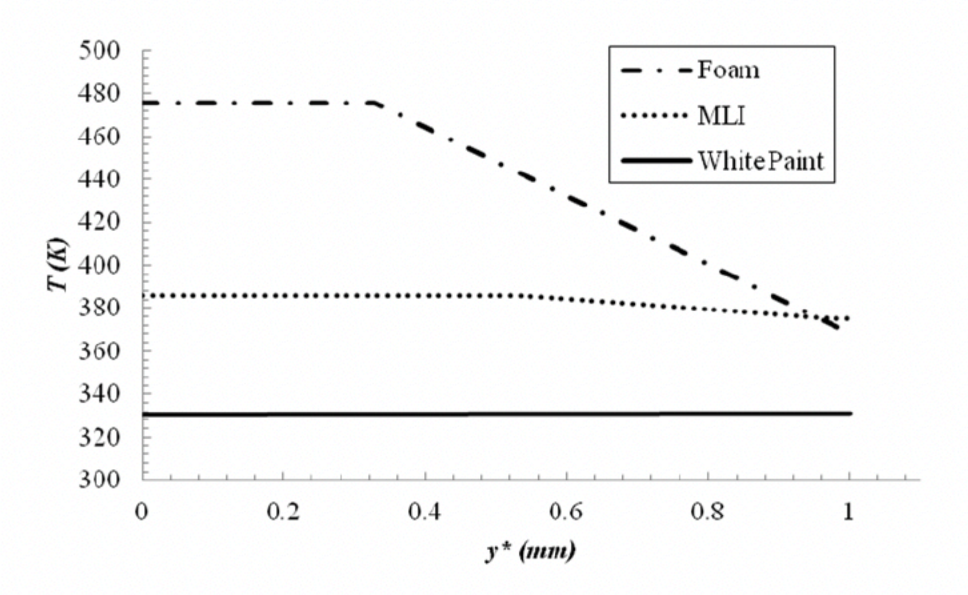


Figure 11: Temperature Profile Comparison (Bianco et al., 2015)

tested to better understand the equilibrium temperatures of the surfaces in space. These functional surfaces using structured silicon with sputtered aluminum to provide reflectance in addition to PEC-VD-deposited silicon dioxide for emittance can be optically tailored for the needs of the specific module or mission (Böhnke et al., 2008). It is shown that these structured surfaces heat up slower than their unstructured counterparts and also provide better emission properties (Böhnke et al., 2008).

Similar to the problem faced by Böhnke et al. (2008), Bonnici et al. (2019) presents a satellite platform goes even smaller in scale weighing in at only 250 grams. Due to it's smaller size, allowing a smaller thermal inertia and driving a small time constant, leading to the potential of much larger temperature swings (Bonnici et al., 2019). Utilizing a similar approach of coatings as well as complex geometry to mitigate thermal loadings that may be experienced, thermal responses were analyzed utilizing a parametric approach, taking the orbital parameters like altitude and beta angle, into account showing that by controlling the surface finish and the beta angle can place the satellite into a thermal environment that the common electronics can survive (Bonnici et al., 2019). The temperature response with varying altitudes and beta angles can be seen further in Figure 12.

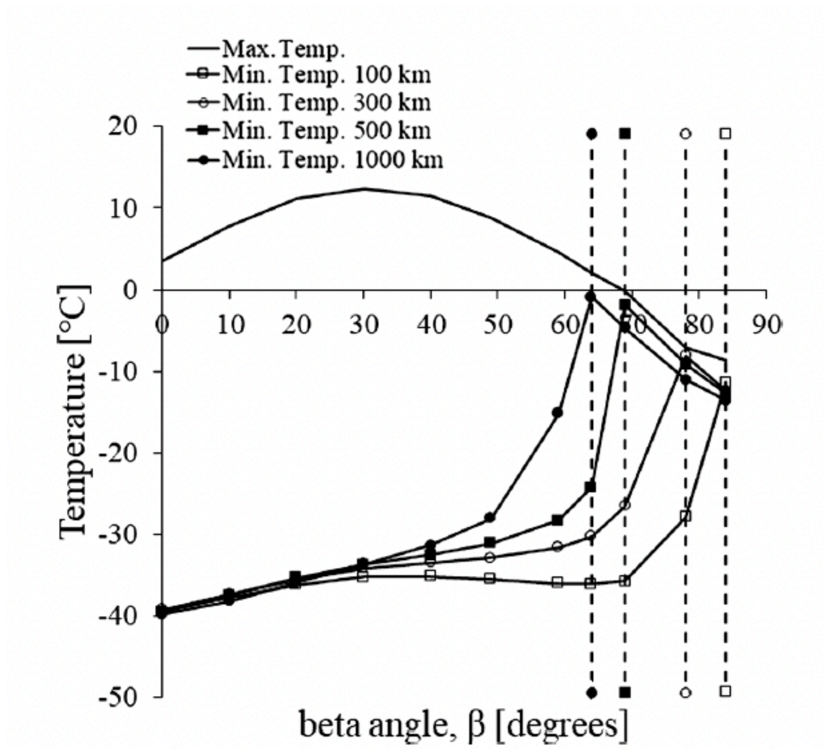


Figure 12: Temperatures as a Function of Orbit Altitude (Bonnici et al., 2019)

Speaking to surface finishes, the decision comes down to a combination of solar absorptivity and infared emissivity of the surfaces to which Corpino et al. (2015) adds: ”Two or more coatings may be combined to obtain the desired values of absorptivity and emissivity...”. Moving to thermal insulation, its goal is to reduce and regulate thermal effects between two adjacent surfaces at different temperatures.

The development of new, enhanced coatings provides for wider operational bandwidth with regard to thermal environments. The use of Bonechar, a carbonaceous calcium phosphate material that is produced through the calcination of bovine bone under a reduced oxygen environment, is being used on a ESA Solar Orbiter S/C under the name SolarBlack (Doherty et al., 2016). SolarBlack underwent numerous tests in preparation for its flight, and showed that it was a stable thermooptical surface, that exhibited high levels of absorptivity and emissivity while maintaining the electrical conductivity and flexibility of uncoated metallic substrates (Doherty et al., 2016).

Result showed absorptivity of 0.96 and 0.95 and an emissivity of 0.81 +/- 0.03 at room temperature applied on titanium and aluminum respectively as well as an equilibrium temperature at the surface of 600C at a perihelion of 0.28 AU (Doherty et al., 2016). Comparatively, the same titanium surface without SolarBlack coating showed temperatures exceeding 850C (Doherty et al., 2016).

Similarly, Somasundaram et al. (2018) produced a new means of passive thermal control for spacecraft by integrating black nickel coatings exhibiting high IR emittance to copper and stainless steel substrates. After environmental testing, the test article showed no significant change in solar absorptance and IR emittance making it an appealing choice as a thermal control coating (Somasundaram et al., 2018). Johnson, Heidenreich, Mantz, Baker, and Donley (2003) continue the work of researching of thermal coatings investigating the use of a potassium silicate binder with a zinc oxide pigment in hopes of it maintaining reflectance over long periods of exposure and thus keeping temperatures down. Temperature is dictated by "...a balance between heat lost through emittance of thermal IR radiation, heat gained through absorption of radiation, and heat internally generated within the spacecraft at an equilibrium state." (Johnson et al., 2003). While drops in reflection and scattering efficiency was seen with longer wavelengths, an optimized particle size for the zinc oxide pigment is believed to be able to provide 2-10 times greater efficiency than the current sizes (Johnson et al., 2003).

Thermal control coatings continue to enhance as new binders and pigments are trialed, altering the thermo-optical properties in an effort to reduce solar absorptance and maximize thermal emittance. Kiomarsipour, Razavi, and Ghani (2013) furthers this work by investigating two novel thermal coatings, pre-synthesized MCM-41 and Zn-MCM-41, incorporated to a potassium silicate binder. Per their experimentation, Kiomarsipour et al. (2013) show that these coatings reflect almost all of the UV radiation which makes them significantly less susceptible to solar radiation than traditional coatings. This is shown all while lower pigment to binder ratios and dry film thickness are achieved allowing these coatings to have reduced weight, lower porosity, and improved mechanical properties (Kiomarsipour et al., 2013). In an effort to introduce a new type of thermal control coating, Mikhailov, Yuryev, and Lapin (2019) conducted a comparative analysis of $BaSO_4$, ZnO, and $TiO_2$ reflective powders. The study showed that with respect to its optical properties and radiation stability, $BaSO_4$ was able to exhibit better performance than its competitors and can be further improved by modifying it with some additional materials (Mikhailov et al., 2019).

With thermal control coatings playing such an important role in satellite thermal control, it is important to understand their degradation characteristics as to better understand the lifetime of the system and vehicle. T. Liu, Sun, Meng, Pan, and Tang (2016) presents a degradation model for some example coatings used in low earth orbit, taking it to consideration the environmental factors there that cause degradation, such as solar absorptance, and the respective failure modes that come with it. Results of the analysis show that solar absorptance degrades coatings significantly over time and can be used as a crucial performance parameter in characterizing coatings (T. Liu et al., 2016).

### 3.1.2 Thermal Insulation

Kang (1999) helps introduce a simple, lightweight, insulation system that have high thermal resistance in a vacuum and utilize this trait to reduce heat loss from a spacecraft or prevent excessive heating of a surrounding from an internal component. This system, called Multi-layer Insulation (MLI) uses multiple layers of radiation shielding to reflect back a large portion of radiant heat flux reaching the spacecraft.

Thermal insulation or more widely known as Multi-Layer Insulation is used across the industry in space, as put by Corpino et al. (2015), for the following purposes:

(1) to prevent excessive thermal flux from/to components;

(2) to reduce temperature variation due to environmental radiative fluxes that vary with time;

(3) to minimize temperature gradients

Corpino et al. (2015) states that when it comes to these smaller space platforms like nanosatellites, ”simplicity and flexibility” will be the primary contributors to the choice of what thermal control techniques best fit the mission.

Technology utilized in space typically has applications in numerous industries and Hengeveld et al. (2010) shows that comparison looking to terrestrial HVAC applications. Additionally, Hengeveld et al. (2010) provides a broad overview as to how environments in space are calculated to provide the context as to why these materials are considered. MLI consists of up to 25 layers of thermal control material that is utilized to have an optimal combination of optical and insulative properties (Hengeveld et al., 2010). While MLI is known to be effective and reliable through flight heritage, it requires a tedious design and installation process due to its inherent fragility (Hengeveld et al., 2010). Taking into consideration aerogels, while possessing a limited flight history, aerogel composite blankets have shown promise with the ability to handle up to 200 psi of compression force before the thermal performance is affected, in addition to its ability to handle repeated flexure and handling (Hengeveld et al., 2010). Based on performance alone, aerogel technology is well suited for a wide-range of applications (Hengeveld et al., 2010).

When exposed to adverse environments like those you may find in Low Earth Orbit, you run the risk of material degradation over time which would generate debris. To best understand this interaction, Gordo, Frederico, Melicio, Duzellier, and Amorim (2020) set up an experiment that would subject materials to vacuum, ultraviolet, and thermal cycles allowing temperature cycling of $\pm$ 200 $^{\circ}C$ without using liquid nitrogen (LN2). Material degradation leading to debris, like paint flaking and Multi-layer insulation layers becoming fragile, were observed in the experiments (Gordo et al., 2020).

While technology on the ground typically takes lessons learned from spaceflight, there are also things that the space industry can learn from the ground as well. R. Hu et al. (2020) provides a state-of-the-art review of personal thermal management (PTM) technologies that include cooling, heating, insulation, and thermoregulation that are more flexible and extensive than traditional solutions. One concept introduced is infrared-transparent visible-opaque fabric (ITVOF) as it led the way for PTM by implementation of radiative cooling followed by wearable heaters, flexible thermoelectric devices, and sweat management textiles.

One of the more difficult cases to analyze for future spacecraft missions is the radiative heat transfer environment that will be experienced by spacecraft thermal insulation which is partially due the transient nature and non-linearity of heat transfer in space causing a reduction in the acceptability of traditional theoretical and experimental methods (Krainova et al., 2017). To develop a new approach to studies and create experimental methods more similar to full-scale flight tests, Krainova et al. (2017) look to generalize a previously developed radiative transfer model and compare these theoretical predictions with new experimental data that take into account the thermal contact between a fibrous spacer and one of the foil layers.

Multi-layer insulation blankets have frequently served as one of the main components of a satellites thermal control system but in analysis, due to the use of reflective thin films such as aluminized Kapton, have assumed an infinite heat transfer coefficient (Mesforoush, Pakmanesh, Esfandiary, Asghari, & Baniasadi, 2019). This assumption considered there to be equal temperatures on two sides of the shield and the effect of thermal resistance was neglected in the total resistance (Mesforoush et al., 2019). ”The thermal performance of MLI is affected by several parameters such as layer density, optical properties of shields, spacer material, perforation coefficient, MLI size, the seam characteristics, etc.” (Mesforoush et al., 2019). Mesforoush et al. (2019) is able to conclude through numerical and experimental analysis that the thermal conductivity of the shield film is an effective parameter in thermal performance of MLI and the previously established assumption of two-dimensional element is questionable.

A study on the Alpha Magnetic Spectrometer focused in on the effects that the thermal blankets used to stabilize temperatures have on various components (F. Yang, Sun, & Cheng, 2020). In simplest terms, when external heat flux increases, the thermal blankets restrain the increase in the temperature, but when that heat flux decreases, it reduces the heat dissipation (F. Yang et al., 2020). Because of this effect, it is seen that the blanket does solve a low-temperature warning problem on one of its components, but alternatively, prevents heat dissipation from the interior and drives high-temperature warnings for multiple internal components (F. Yang et al., 2020).

Prosuntsov and Praheeva (2021) investigated high temperature open cell carbon materials, finding ways to reduce the weight of spacecraft insulation by adjusting the porosity level and characteristic pore size of the material by building geometric models and modeling the radiative heat transfer of the representative volume. Through this analysis it is determined that carbon materials with a porosity of 90-95% and a characteristic pore size of no more than $100\mu m$ (Prosuntsov & Praheeva, 2021). The use of materials with this proposed structure could significantly reduce the specific density of the insulation on the spacecraft (Prosuntsov & Praheeva, 2021).

## 3.2 Internal Thermal Control

Revisiting Equation 1 will allow a greater understanding of the requirements for internal thermal systems. The two aspects not yet considered to is energy generated ($\dot{E}_{gen}$) and energy stored ($\dot{E}_{stored}$). Energy stored comes from two places. First, it is a function of satellite mass and thermal properties. This has the ability to significantly affect the variation in component temperatures that occurs over the orbit of the satellite as it is exposed to time-varying boundary conditions (Hengeveld et al., 2010). Energy storage could also help as the S/C can retain heat to stay warm when in the eclipse portion of the orbit and facing harsh low ends of the temperature profile. It is also important to note that the S/C structures with low capacitance yield significant spikes in temperature over one orbit while those with high capacitance have a smoother temperature profile which is desired but typically comes with a mass penalty and thus higher cost to launch (Hengeveld et al., 2010). The second aspect of energy storage is through the use of Phase Change Materials (PCM). PCMs open the opportunity to retain some of the energy generated by the components, taking advantage of the high latent heat capacity of certain materials, to the be utilized later when heating of components is required to maintain operational temperature ranges.

Energy generated primarily comes from the high waste heat of components being used on board (Ambrose, Feild, & Holmes, 1995). This heat loading can lead to its own issues as it means that components onboard are not operating efficiently and the temperatures may restrict the operation of certain components unless certain conditions are met. The desire is to be able to remove this heat through the use of a internal thermal management system.

### 3.2.1 Heat Pipes

Wrobel and McGlen (2021) describes the operation of a conventional heat pipe (HP): ”When heat is introduced to the evaporator section of the HP, liquid phase working fluid within the evaporator wick, evaporates. As the vapor flows into the adiabatic section of the heat pipe, a pressure drop occurs that enables acceleration of the vapor, creating a high mass flow rate of the working fluid to the condenser region, which due to condensation of the vapor, is at a lower pressure enabling fluid flow. The condensate is then pumped back to the evaporator due to the capillary force, which is generated within the capillary structure within the wick. The phase-change phenomena employed in HPs allow them to achieve very high values of equivalent thermal conductivity, typically from 5,000 $W/m{\cdot}K$ to 100,000 $W/m{\cdot}K$, as compared with pure Copper, 387 $W/m{\cdot}K$. This value depends upon the mechanisms used to add heat at the evaporator, and remove it at the condenser, where high conduction and convection thermal resistances may exist.” (Wrobel & McGlen, 2021). Figure 13 further details the operation of a heat pipe.

The most common style of heat pipe used for spacecraft thermal management today are axial-grooved ammonia-aluminum heat pipes, but these require special die manufacturing and extrusion technique to produce, which makes these heat pipes typically limited to academic research and industrial applications (Z. Li, 2018). Z. Li (2018), identifying this as problematic, presents a new heat pipe structure that is made of a spiral coil wick structure in a simple piping container. Two experimental test articles were manufactured and showed that the new spiral wick

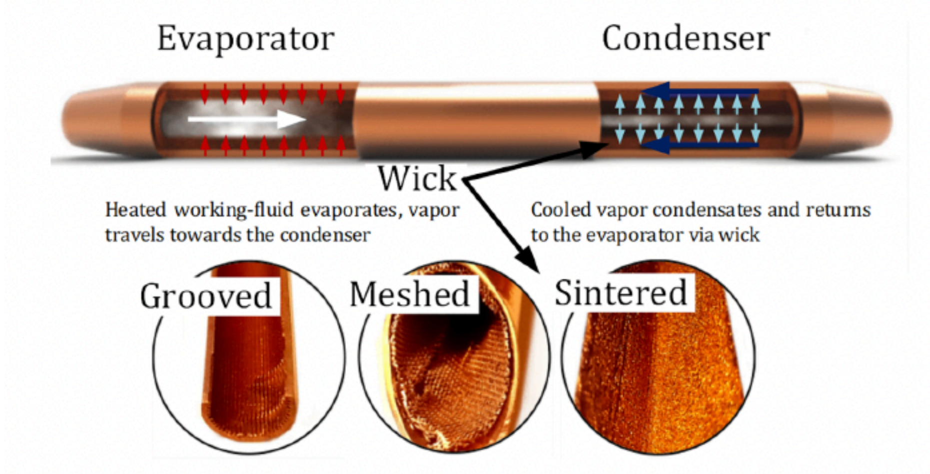


Figure 13: Schematic representation of a Copper-Water Heat Pipe with Alternative Wick Construction (Wrobel & McGlen, 2021)

design successfully functioned as a capillary wick and that the proposed arterial heat pipe realized high effective thermal conductivity of the same order of magnitude of an aluminum axial-grooved heat pipe (Z. Li, 2018).

Jasvanth, Adoni, Jaikumar, and Ambirajan (2017) designed, developed, and tested a two-phase ammonia loop heat pipe capable of transporting up to 500W of heat. The heat pipe consists of an evaporator, compensation chamber, fluid transport pipes, and a condenser. Jasvanth et al. (2017) present the effect of heat load and adverse system level elevations on the operating temperature and conductance of the loop heat pipe, testing at heat loads up to 600W and elevations of 1000mm. Thermal conductance was shown to decrease in the evaporator with an increase in elevation and at heat loads upwards of 300W led the heat pipe to constant conductance (Jasvanth et al., 2017).

When constructing spacecraft systems for long duration flight, it is important that thermal control systems can maintain functionality for the mission lifetime. Kianfar, Joodaki, Dashti, and Asghari (2021) evaluated heat pipes in space applications using various prediction methods to determine lifetime in different operating temperatures. After analysis, it is shown that the evaluated heat pipes failure time ranged between 12 and 60 years matching up with world standards.

Pietrasanta, Postorino, Perna, Mameli, and Filippeschi (2020) identifies pulsating heat pipes as another candidate for spacecraft thermal control but a lack of data exists for its use in relevant environments. To address this gap in the research, Pietrasanta et al. (2020) tested the heat pipe in thermo-vacuum conditions at different heat loads and environment temperatures revealing that the performance and operating ranges of pulsating heat pipes may be overestimated if tested under ambient conditions.

Tang et al. (2018) presents a new method of cooling called a multi-heat source and double-end cooling (MSDC) developed for spacecraft applications. Currently, heat pipes are typically only used with single heat sources but this design utilizes an ordinary, cylindrical heat pipe and employs dual end cooling. Experiments were conducted to determine effects of various heat inputs and flow rates in a horizontal orientation. Results exhibit that the heat transport capability of a heat pipe is significantly improved with the proposed method with temperatures reaching steady values without shock to the system even under high load (Tang et al., 2018). Due to its stable operation at high flux, low cost, and sizing, the heat pipe with MSDC is an appealing solution for spacecraft thermal control.

Vasiliev and Vasiliev (2005) propose an advancement on heat pipes referred to as a Sorption Heat Pipe (SHP). The SHP is the combination of a conventional heat pipe with the sorption phenomena and due to its insensitivity to some gravitational acceleration, is believe to be a good choice for space applications. Sorption is known as the capture of gas or vapor by a substance in a condensed state. The SHP exhibits significant heat transfer enhancement when compared to a conventional heat pipe with similar working fluid and dimensions (Vasiliev & Vasiliev, 2005).

R. Yang, Lin, He, Bai, and Miao (2017) identifies that one of the main causes of adverse performance in loop heat pipes is non-condensable gas which is typically generated by chemical reactions among the impurities, container wall, wick material, and the working fluid. To simulate this, Nitrogen was charged into two ammonia-stainless steel loop heat pipes to which R. Yang et al. (2017) then used a thermoelectric cooler to see the effect of it on the operation of the loop heat pipe with the non-condensable gas inside. Results showed that the thermoelectric cooler decreased steady-state operating temperature, expanded the allowable heat loading, conducted a successful startup with reduced temperature overshoot, and eliminated temperature oscillations thus showing it can effectively improve the thermal performance of the loop heat pipe (R. Yang et al., 2017).

H. Zhang et al. (2020) presents an enhancement to the traditional loop heat pipe by incorporating a pump into the loop to overcome the limited heat transfer distance and temperature oscillation seen originally. This light, small, high-speed pump was tested at various power inputs and various temperatures of the heat sink and showed that when the heated surface was limited to 80 $^{\circ}C$, the loop could handle the max heat dissipation of 370 W when the power consumption of the pump is 4 W and the temperature of the heat sink was -10$^{\circ}C$ (H. Zhang et al., 2020).

### 3.2.2 Phase Change Materials

One of the earliest examples of investigating the uses of phase change material (PCM) in spaceflight is exhibited by Fixler (1966), who presents an analytical and experimental investigation on whether PCM is suitable as a passive thermal control system on a satellite. Analytically, its understood that the overarching problem is that an allocation of PCM is exposed to different heat loading inputs from two opposite faces. On the external face, the heat flux consists of the net variable radiative energy from solar flux and planetary longwave radiation while the internal face, is exposed to the heat input caused by the equipment heat dissipation that may or may not be constant. (Fixler, 1966). An exact analytical solution can not be found so numerical methods must be used such as finite-difference method. With a solar absorptivity of 0.1 and a weight that is 0.312psf from the closest competitor, the passive PCM system certainly has advantage.

The rapid adaptation and acceptance of phase change materials has also led to insight and development to new means of heat storage and heat transfer. Belyavskii (2021) recognized the benefits of heat stores because the PCM inside prevents overheating by removing and retaining internally the heat of components but also assists in preventing excessive cooling by releasing heat stored back to the component. Belyavskii (2021) takes this knowledge and analyzes various geometries in an effort to optimize the heat store design and reduce the parasitic effect of the shell that contains the material, to which is constant for his analysis. It is found by Belyavskii (2021) that heat stores in the form of parallel plates of material, between which the heat-transfer passes, are best for use in the thermal control of S/C.

The proposition of using heat stores in spaceflight provides the opportunity to reduce mass of thermal control systems. Per Belyavskii, Novikov, Sorokin, and Shangin (2019), these thermal stores must meet the following requirements:

(1) High thermal capacity per unit volume or weight

(2) Repeated charging and discharging without loss of efficiency

(3) Reversible phase changes or reactions

(4) Isothermal heat transfer

Cao et al. (2020) provides a new problem set because hypersonic vehicles require similar thermal protection capabilities to S/C, there are certain load bearing capabilities that are required to prevent excessive deformation of the structures during these heating periods and assure safe operation. To handle this added requirement, Cao et al. (2020) took lessons learned on previously researched structurally and thermally integrate protection systems to move forward on a corrugate core system. The problem with the corrugated core though, is that the webbing serves as a thermal bridge passing heat loading through the insulation with minimal resistance. Thus, this team utilized shape stabilized PCM as an intermediary and maintains the advantages of ”lightweight and structural efficiency” (Cao et al., 2020) while alleviating thermal concerns from the webbing.

As PCMs continue to find a stronger foothold in S/C thermal control, Collette et al. (2011) acknowledged that there are further performance requirements that should be understood prior to settling on a material selection. To assist in their material selection, 19 different PCMs of various classes were analyzed as candidate materials. Additionally, best characteristics were outlined on what a PCM should possess for this specific application and can be found in Table 2 (Collette et al., 2011):

Table 2: Desired Characteristics of Candidate PCM (Collette et al., 2011)

| Property or Characteristic | Desired Value or Tendency |
|---|---|
| Heat of Fusion | High |
| Thermal Conductivity | High |
| Specific Heat | High |
| Density | High |
| Volume Change during Melting | Low |
| Vapor Pressure | Low |
| Melting and Freezing Behavior | Dependable and Reversible |
| Availability | Readily Available |
| Cost | Low |
| Compatibility | Compatible with container and filler material |
| Reversible Solid-to-Liquid Transition | High |
| Long Term Reliability During Repeated Cycling | High |
| Toxicity | Non-Toxic |
| Hazardous Behavior | Not Exhibited |
| Property Data | Readily Available and Well Documented |
| Flash Point | High |
| Coefficient of Thermal Expansion | Low |
| Surface Tension | Low |

After initial evaluation, all but 5 materials were disqualified and the remaining were submitted to thermal testing. Further, an investigation into implementation of PCM to a system, showing mathematically how to optimize a PCM implementation and then supporting these points by a look to various case studies that show the most promising implementations and mission scenarios where a PCM could give an advantage when compared to a traditional system. (Collette et al., 2011). In conclusion, Collette et al. (2011) found that while PCM was not often used for European space applications due to mass concerns, the new and improved materials being presented today show promise especially considering smaller space platforms like nano-satellites. While some cases did show a net mass gain, it also showed large power savings and effective damping of temperatures.

In a continuation of their previous work, Collette et al. (2014) utilized the previously built mathematical models and the literature collected to further test a choice set of PCMs in a laboratory setting while also putting effort forward to construct two different prototypes with previously established candidate paraffin wax PCMs: n-docosane and n-octadecane. The PCMs were evaluated for induced corrosion with the planned aluminum housing. It was found that the two selected PCMs had negligible effect on the aluminum while the hydrated salts had an opposite effect. Further, the two prototype PCMs continued to be evaluated, gathering more information to characterize thermal behavior of the PCM and the effectiveness of the two prototype builds for their respective environment choice.

Zhao, Xing, and Liu (2020) investigates the use of a low melting point alloy (Bi-Pb-Sn-Cd) as a candidate phase change material for thermal management systems. When compared to an organic PCM that possess a similar melting point, the alloy was found to have a much larger thermal conductivity and volumetric latent heat (Zhao et al., 2020). With similar dimensions, the alloy heat sink outperformed the other heat sinks at various input levels that could effectively reduce the temperature of the heater and extend the effective time 1.5 times (Zhao et al., 2020).

Additive manufacturing combined with PCM technologies opens new applications in spacecraft thermal control as shown by the work done by Y. Guo, Yang, Fu, Bai, and Miao (2021). In this work, a star sensor baffle is developed utilizing 3-D printing, composed of aluminum with a lattice structure, and combined it with a phase change material thermal energy storage medium to control temperature of the baffle Y. Guo et al. (2021). The temperature control strategy was expected to keep temperatures between -2 and 11 C. The work done validated the feasibility of this combination technology, showed that the baffle maintains the temperature range desired, and maintain an acceptable operation temperature (Y. Guo et al., 2021).

One of the biggest justifications for use of PCM in spacecraft thermal management is that without the thermal storage capability afforded by it, radiators must be sized to be large enough to release the maximum power (Kabir, Gemeda, Preller, & Xu, 2021). To reduce the size of that radiator, thus reducing overall mass of the system, Kabir et al. (2021) integrates PCM into a passive two-phase heat exchanger which reduces the size of the radiator can be sized for the average output vs. the maximum. In the continuation of research into PCM applications for spacecraft thermal control systems, Kansara, Singh, Patel, Bhavsar, and Vora (2021) investigate the effects of a low gravity environment on the melting and solidification of phase change materials implemented. The melting and solidification processes are simulated for materials at different values of gravitational acceleration ranging from g to g/80. The effect on natural convection is noticeable on all but one, glycerol which is believed to be due to the highly viscous nature and faster heat propagation of the material (Kansara et al., 2021). This information will allow engineers to best determine sizing of heat sinks operating under these environments and provide a modality for analyzing it.

T. Kim et al. (2010) identified that if a high heat dissipating component work at intermittent times, the radiator for this system will be sized to account for instant peak temperatures which could lead to an oversized radiator. An oversized radiator leads to unnecessary mass and added cost so the idea of integrating a solid-liquid PCM came about. Due to the heat of fusion of the PCM, the instant peak temperatures are kept to moderate levels, reducing radiator sizing and increasing radiator thermal capacity. Additionally, the use of PCM reduces heater power consumption due to the accumulated heat of the PCM. To address concerns of limited power resources, small heat capacity, insufficient radiator area, high-density packing of electronics, and mass limitations of small satellites, Yamada and Nagano (2015) presents the design of a heat storage panel, a thin carbon-fiber-reinforced polymer panel with PCM encapsulated in it. This panel was tested both here on the ground and in orbit on the Hodoyoshi-4 satellite and showed in both that the HSP has high potential for thermal control on small satellites.

A downside to conventional PCMs is that they possess a constant strong rigidity and makes it extremely difficult to integrate these materials in to complex structures with a control device which contributes to bad surface contact with devices resulting in high thermal resistance (W.-W. Li, Cheng, Xie, Liu, & Zhang, 2017) unless you possess molding that would allow you melt down the material and mold it to the complex structure. Problem is this method requires time and additional manufacturing processes. To broaden the use of PCM, W.-W. Li et al. (2017) present and investigate the idea of flexible PCMs with high thermal conductivity. They are able to successfully prepare this using olefin block-copolymer as a supporting material and shows through experimentation that these materials exhibit high chemical stability, high latent capacity and encapsulation in line with what you would see out of conventional PCMs, but also exhibit thermal sensitive flexibility at a defined temperature level. This enhancement to PCMs greatly improves the feasibility of using PCM in spacecraft thermal control applications (W.-W. Li et al., 2017).

Two primary issues with designing a cost effective PCM thermal storage system is selecting a suitable candidate material and to increase heat transfer between the storage material and the working fluid as performance is limited by poor thermal conductivity of the latent heat of the material (M. Liu et al., 2012). M. Liu et al. (2012) provide multiple inorganic materials that are PCM candidates for thermal storage systems used terrestrially at solar power plants for potential use in high temperature situations, exhibiting melting temperatures of greater than $300°C$. These candidate materials could hypothetically be considered for space use if cooperative with housing materials. You can see the heat capacity and cost per kWh for these candidate PCMs in Figure 14. Zalba, Marın, Cabeza, and Mehling (2003) lists out all of the different inorganic substances and eutectics, organic substances, fatty acids, and commercial PCMs that could hold potential use as thermal heat storage devices. Additionally, Zalba et al. (2003) also compares organic and inorganic materials and states their advantages and disadvantages for heat storage while also pointing out the important characteristics of energy storage materials.

Drawing parallels from terrestrial applications provides the opportunity for technology on

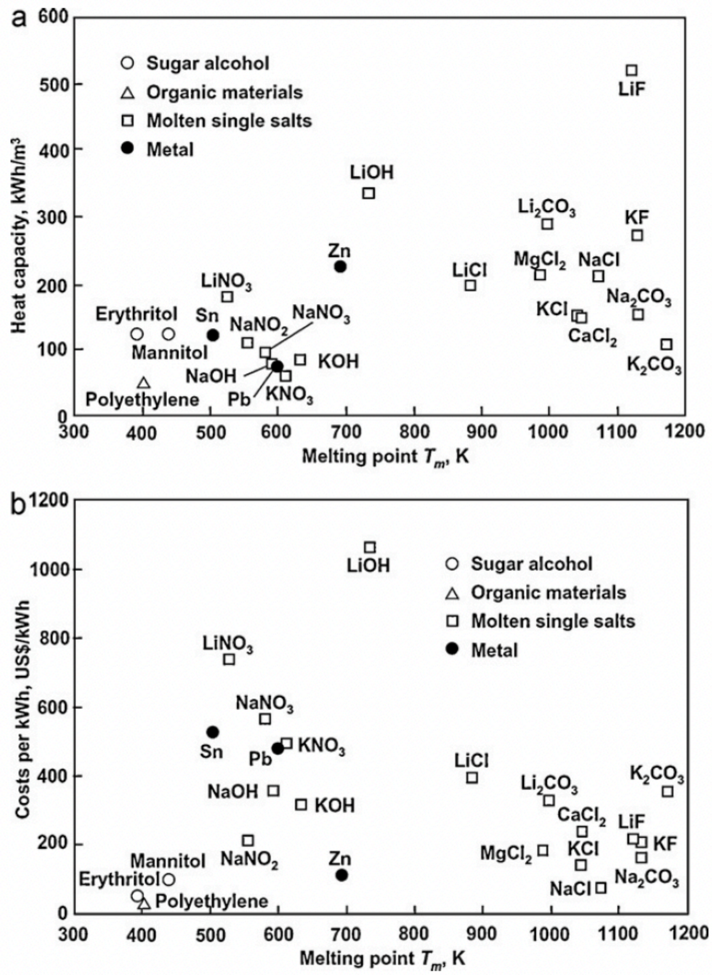


Figure 14: Heat Capacity and Cost of High Melting Point PCMs (M. Liu et al., 2012)

both sides to progress and advance. For example, just being able to justify the use of incorporating PCM into a spacecraft thermal control system by using the results of two studies showing PCM incorporation in building elements drove a 24.22% and 32.8% heat flux reduction respectively when compared to ordinary bricks (Mavrigiannaki & Ampatzi, 2016). You can also find justification of power savings due to a 10.8% cooling load reduction for an experimental cabin integrated with phase change frame wall (Mavrigiannaki & Ampatzi, 2016).

Peng, Guo, Li, and Feng (2021) evaluated gallium as a potential PCM, due to its thermal transport properties, to better understand its dynamic melting behavior and heat transfer performance under microgravity. While it is true that gallium, ice, and n-octadecane have different phase-change temperatures, this study focused from the point that melting begins and the overall heat capacity of the material. The use of gallium reduces the melting time by 88.3% and 96.4% when compared to ice and n-octadecane respectively and increase total energy storage by 20.7% and 123.3%. These results indicate that gallium is a suitable choice for effective for spacecraft temperature control (Peng et al., 2021).

Ren et al. (2020) present a novel multi-layer thermal protection system (TPS) using erythritol, a phase change material, as the insulative layer. Erythritol possess a melting temperature of 390.15 - 393.15 K and a latent heat of 340 kJ/kg (Ren et al., 2020). A mockup of this can be seen in Figure **??**. Utilizing numerical analysis, its heat performance is confirmed, showing that the addition of PCM reduces the temperature at each layer (Ren et al., 2020). It is also confirmed that a greater latent heat, larger specific heat capacity, and an acceptable phase transition temperature all further contribute to the decrease in temperature of all layers (Ren et al., 2020).

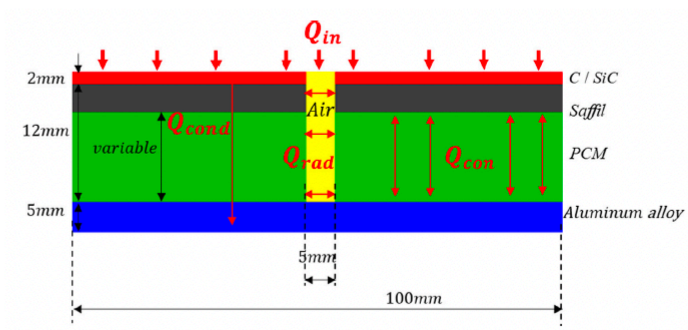


Figure 15: Multilayer TPS Structure (Ren et al., 2020)

Highlighting the limitations of PCMs, low thermal conductivity is frequently identified as one of the major hindrances to the effective heat transfer capability of the entire passive thermal control system (Righetti, Zilio, Doretti, Longo, & Mancin, 2021). Righetti et al. (2021) investigate whether the use of enhanced surfaces could mitigate this issue and improve the capabilities of the PCM. It is found that the use of an enhanced surface does not always lead to an improvement of the heat transfer performance and to truly design an efficient and compact PCM based thermal control system, an integrated approach is required that accounts for the material properties, the device requirements (heat flux, operation times and capacity), and the defined boundary conditions (natural or forced convection) (Righetti et al., 2021).

One of the primary benefits of PCM is its ability to operate independently of a power supply. This is extremely helpful in critical failure scenarios but there is not much data on the capability of PCM during these failure scenario and how well it may function without the assistance of auxiliary/support equipment. J. Wang, Cao, Yuan, Leng, and Sun (2021) conducted a study to address this with regard to information systems. A mathematical heat transfer model was set up considering a cooling system with PCM plates was implemented and then evaluated for the case of an emergency power failure of a heat releasing space. Results found that the cooling system was able to effectively maintain the air temperature of the room below 35 $^{\circ}C$ within 9h and until 16h (J. Wang, Cao, et al., 2021). It was also found that the lower the melting temperature, the better the air temperature was controlled with this study considering the melting temperature to be approximately 25 $^{\circ}C$ (J. Wang, Cao, et al., 2021).

fan Wu, Liu, long Cheng, and Liu (2013) identifies that externally sourced short-term high heat flux has the ability to cause faults in a spacecraft thermal control system leading to temperature anomalies or failure of internal equipment. To protect from this, fan Wu et al. (2013) proposes using a shape stabilized PCM that has high thermal conductivity and does not require tight packaging. Results have indicated that the PCM can effectively absorb the heat to prevent faults during extreme heat flux changes and has no negative effect on that spacecraft in normal heat flux (fan Wu et al., 2013).

### 3.2.3 Highly Conductive Materials

As the miniaturization of S/C hardware continues to take a greater foothold in the industry while power demands increase, advancements in thermal technology are required to handle higher loads with less contact. Thermal interface materials (TIM) is used to reduce contact resistance at the heat sink and allows for significantly better conductive heat transfer. Gwinn and Webb (2003) provide a state-of-the-art review on thermal interface materials discussing the advantages and disadvantages of various materials.

G. Hu et al. (2020) provides a new innovation in heat transfer technology that keeps modularization and functionality in mind. To allow for structural independence, G. Hu et al. (2020) investigates the integration of a conductive plate at the docking port of these modules that permit heat transfer module to module and still support repeatable connection-disconnection as seen in Figure 16. A model is developed and tested showing that the thermal control technology can satisfy the thermal demand of the modular satellite (G. Hu et al., 2020).

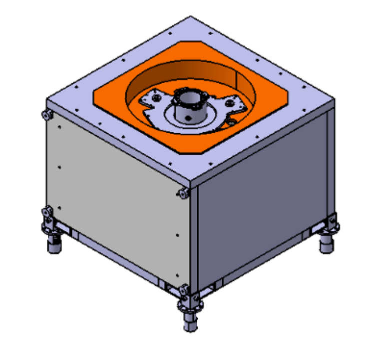

Figure 16: Modular Satellite with Conductive Material Docking Port Equipped (G. Hu et al., 2020)

#### 3.2.4 Thermal Devices and Heaters

Another means of regulating heat flow onboard a S/C is through the use of Thermoelectric Devices. Colomer et al. (2015) describes these as ”...solid-state systems consisting of a number of alternate (positive) and (negative) type semiconductor thermoelements, which are connected electrically in series by metal interconnects and sandwiched between two electrically insulating and thermally conducting ceramic substrates.” Having been used previously and proven functional in aerospace, instrumentation, medicine, industrial, or vehicles, these devices can act as a cooler or heater depending on the direction of the current being inputted (Colomer et al., 2015). These devices have the ability to be tunable, altering function based on the environment and the needs of the vehicle. The modes of operation can be seen in Figure 17 Through theoretical analysis, it is shown that ”thermoelectric devices could serve as a variable insulators for applications where the heat fluxes between two thermal bodies need to be altered conveniently." (Colomer et al., 2015). Applications like a S/C thermal control system, especially those on a smaller scale using high power micro-electronics, fit right into this description.

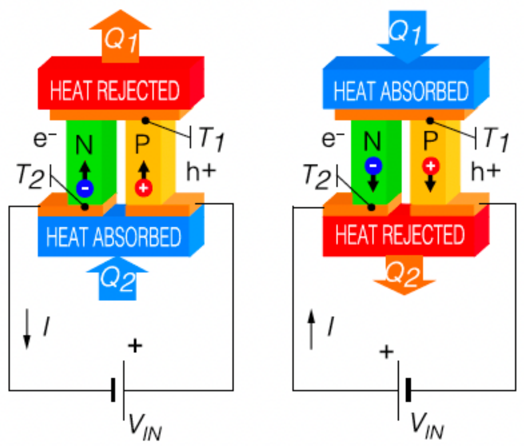


Figure 17: Schematic representation of Active Control Modes of Tunable Thermoelectric Device (Colomer et al., 2015)

As the scale of devices continues to decrease, yet they have been becoming more powerful. Producing more heat that conventional heat sinks combined with heat pipes or water cooling simply can not dissipate to the environment at such a scale. To provide a new solution for this dilemma, Klinar et al. (2020) presents the use of thermal control devices that can enable better control of the heat flux. Each device has its own operational parameters that defines the thermal resistance through it and thus affects the heat flux through it. Klinar et al. (2020) provide an extremely valuable depiction that shows how these devices vary and further details the operation of thermal switches, thermal transistors, thermal diodes, and thermal regulators and can be seen in Figure 18.

The design of a thermal control system typically accounts for worse case scenarios on both ends of heating and cooling which can lead to overencumbering equipment. H. Kim et al. (2019) investigated a new means of radiating heat out to space but altering the emissivity of the radiator dependent on surrounding conditions. The solid state passive switchable radiator would modulate the emissivity dependent on when the spacecrafts temperature was either lower or higher than

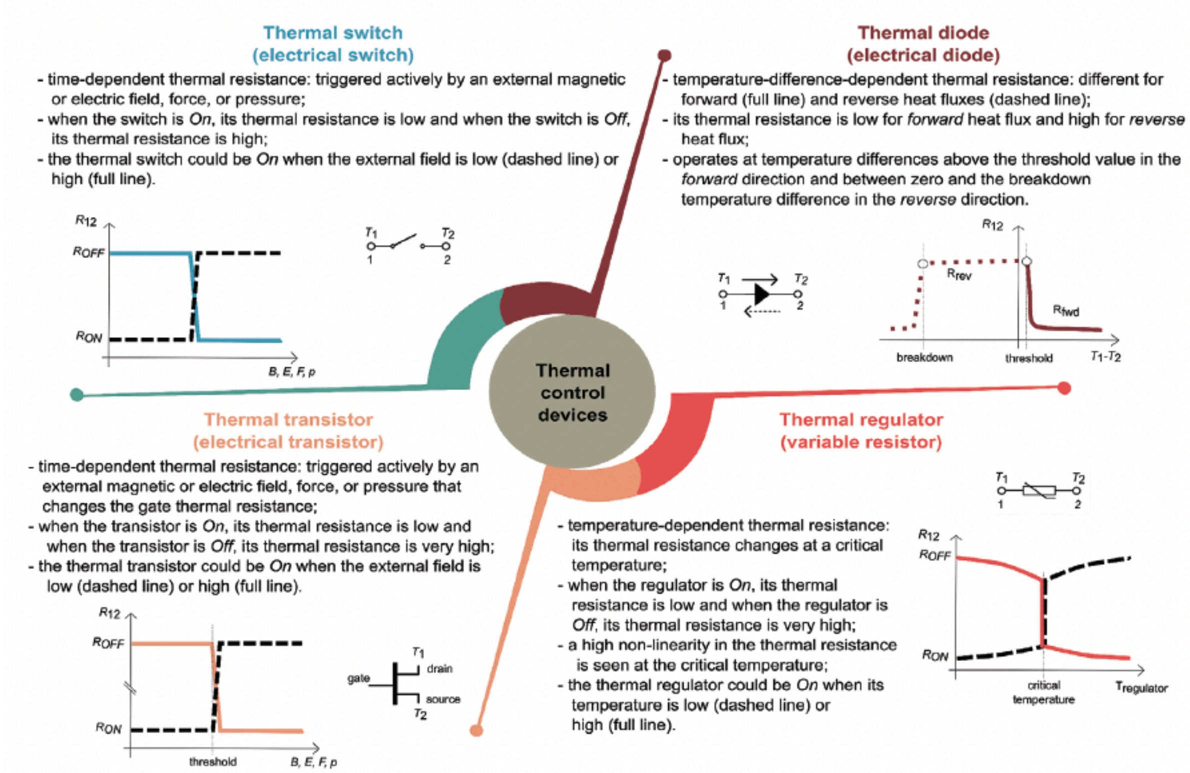


Figure 18: Operation and Types of Thermal Control Devices (Klinar et al., 2020)

ideal. This was accomplished using a thermochromic material, VO2, that would change phase change at 340 K and concurrently have a dramatic change in optical properties (H. Kim et al., 2019). During experimentation in a simulated space environment, it was shown that when below 340 K, the structure behaved like a simple reflector thus minimizing radiative heat loss but acted as a absorber greater than 340 K providing a radiative cooling effect (H. Kim et al., 2019).

The problem of needing heaters at low thermal loading brings the additional mass of batteries and supporting components further complicating overall spacecraft designs and space constraints. Stavely and Lesieutre (2013) present the idea of using contact aided cellular compliant mechanisms to an self-adaptive passive response to thermal switches. These cells deform to make contact when a compression force is applied, controlling the flow of heat from the component to the radiator through direct mechanical contact, introduce new thermal pathways, and increase the thermal resistance for low-conductivity modes (Stavely & Lesieutre, 2013).

### 3.2.5 Mechanically Pumped Fluid Loops

P. Zhang et al. (2019) breaks down how mechanically pumped cooling loops work and the various configurations they can be utilized in. Mechanically pumped fluid loops usually consist of a reservoir, preheater, mechanical pump, condenser, evaporator, connecting pipes as seen in Figure 19(P. Zhang et al., 2019) . To reduce the energy consumption in the system and optimize the heat distribution, a heat dissipating device with a heat exchanger is usually used as seen in Figure 20 (P. Zhang et al., 2019).

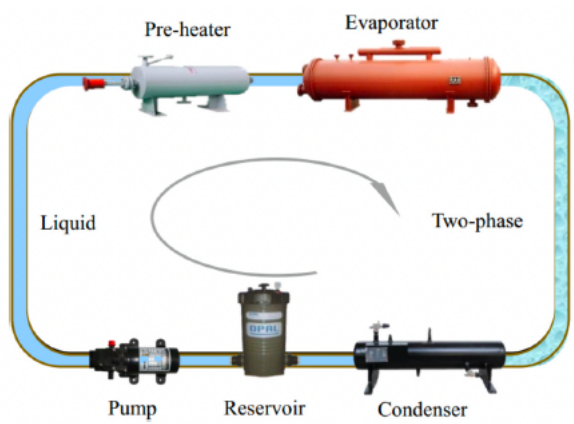


Figure 19: Schematic of Mechanically Pumped Cooling Loop (MPCL) (P. Zhang et al., 2019)

In the era of shuttle and space station design, Peterson (1987) was one of the first to challenge the solutions of the Gemini era by presenting a review on two-phase pumping mechanisms. The principal method of spacecraft thermal management during the Gemini program was a pumped single phase system, but the use of a fluid's sensible heat results in large temperature variation in the loop and single phase systems are typically designed specific to the components and predetermined thermal load thus making any changes to loading requires significant design modifications. Looking now to the modern day, Y. Zhang and Tong (2016) talk to active thermal control systems being studied for use on the Chinese Manned Space Station for their payload racks. These racks contain high value experiments, each putting out their own heat during operation. Y. Zhang and Tong (2016) presented the idea of a multi-scheme active control system that can choose between either liquid cooling, air cooling, or a combination of the two.

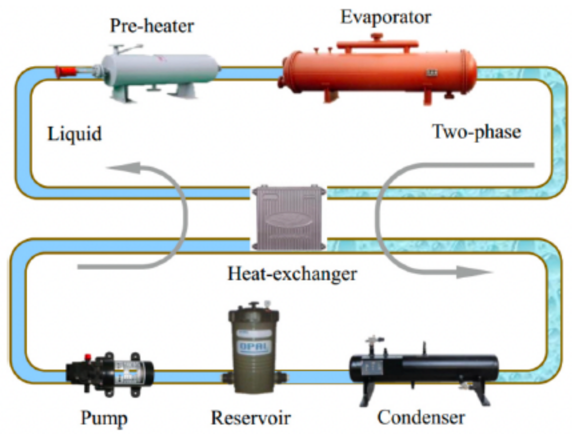


Figure 20: Schematic of MPCL with Heat Exchanger (P. Zhang et al., 2019)

Lee, Mudawar, and Hasan (2016) points out one of the most uncertain variables that readily influences the entire United States space agenda, budget constraints and policy changes, causing uncertainty to which missions would take precedence. With these considerations, there is an urgent need for a more versatile spacecraft that can endure varying gravitational and thermal environments. To address this, Lee et al. (2016) presents a hybrid thermal control systems (H-TCS) that can accommodate three modes of operation: single phase, two-phase, and heat pump. A two-phase pumped loop has advantages in cold environments due to the temperature of the working fluid needing to exceed the effective heat sink temperature to enable rejection at the radiator, while in warm environments a vapor compression heat pump is advantageous as it is required to reject heat (Lee et al., 2016). In operation, this system would operate single-phase for low heat loads, two-phase for high heat loads, and heat pump for hot environments (Lee et al., 2016).

While a mechanically pumped two-phase loop is a familiar thermal control choice, transient behaviors caused by start-up and heat sources loading can be problematic for certain payloads. Meng et al. (2020) presents a novel two-phase thermal-controlled accumulator design with passive cooling to buffer the oscillations and manage temperature. Results of testing this new accumulator showed that it was capable of managing small temperature oscillation and maintain a uniform temperature distribution across during the whole process showing promise for future use (Meng et al., 2020).

Essential liquids like water or ethylene glycol typically have lower thermal conductivity making it a less efficient choice as an absorber fluid. Shah, Gupta, Sonvane, and Davariya (2017) presents the idea of using nanofluids, a common fluid with ultra fine solid particles suspended in it, to improve thermal conductivity. Proof of this is seen experimentally when ethylene glycol has CuO particles added leading to 20% increase in thermal conductivity (Shah et al., 2017).

WANG et al. (2019) reports that a sublimator can be an effective heat rejection approach for spacecraft working in a vacuum environment. When integrated with a active fluid loop, the waste heat can be collected from multiple sources and transported to the sublimator but there is not much insight into the heat and mass transfer performances for the sublimator combined system (WANG et al., 2019). WANG et al. (2019) conducted a study to evaluated the influence of the working fluid mass flow rate on the heat load of the loop and reveals that the heat rejection ability did not always increase with the increasing of the mass flow rate.

J.-X. Wang et al. (2017) presents a novel closed loop spray cooling system that is believed to be gravity-immune and suitable as a thermal protection platform for a spacecrafts power transmission system. The gravitational immunity is accomplished via the ejector loop that generates a local low pressure to ensure rejection of the two-phase mixture in the spray chamber (J.-X. Wang et al., 2017). Figure 21 shows how the closed loop system functions. In ground-based experimentation, the largest critical heat flux, or the maximum heart dissipation capability of a certain operation condition, could be up to 705 $W/cm^2$ with high volumetric flow of 25.5 $L/h$ and optimal efficiency was calculated to be 9.34% with low volumetric flow of 14.5 $L/h$ (J.-X. Wang et al., 2017). Similarly to J.-X. Wang et al. (2017), H. Zhang, Li, Wang, Liu, and Zhong (2016) presents a cooling method known as spray cooling that has shown great performance on ground applications but has experienced difficulty being implemented in space due to the inability to dissipate the vapor-liquid mixture from the heat surface. In the H. Zhang et al. (2016) study, an ejected spray closed-loop cooling system for space applications that uses the negative pressure in a ejected condenser chamber to remove the two-phase mixture from the spray chamber and finds through ground-testing that:

1. The heat surface temperature, heat transfer coefficient, vaporization ratio and heat transfer efficiency rise up with the increase of the heat flux (69.76–311.45 $W/cm^2$) at the same spray volume flow. The heat transfer coefficient is higher when the spray inlet temperature is 69.2$°C$ compared with that at the spray inlet temperature of 78.2$°C$, which results in a lower heat surface temperature. What's more, the vaporization ratio is lower when the spray inlet temperature is 69.2$°C$. The heat transfer efficiency is also lower with the same heat flux imposed.

2. With the increase of spray volume flow (11.22–15.76 $L/h$), the heat transfer coefficient rises up and the heat surface temperature decreases at the same heat flux and spray inlet temperature. The heat transfer efficiency decreases and the vaporization ratio increases.

3. For the same spray cooling inlet temperature (78.2$°C$) and spray inlet volume flow (13.6 $L/h$), the influence of two different volume flows (99.5 $L/h$ and 77.5 $L/h$) through the ejected condenser nozzle on the heat transfer performance have little difference.

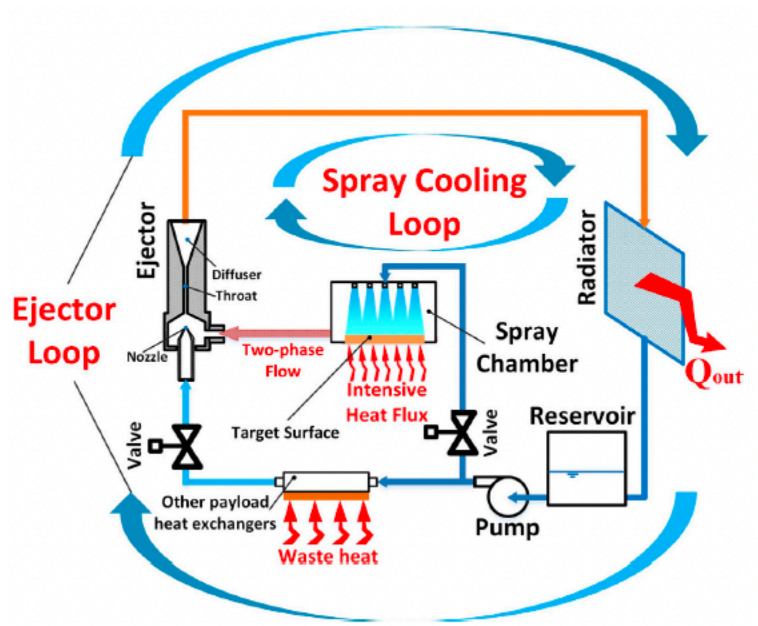


Figure 21: Schematic representation of the Space-Oriented Spray Cooling System (J.-X. Wang et al., 2017)

In an attempt to improve upon the concept of a traditional single-phase mechanically pumped fluid loop, a novel actively-pumped loop system is proposed utilizing a distributed thermal control strategy with each branch controlled by a thermal control valve with a paraffin-based actuator (J.-X. Wang et al., 2016). The coolant flow rate and the cold plates heat removal capacity are sensitively controlled by the heat loaded upon the cold plate J.-X. Wang et al. (2016). A validating system prototype was created with the objective of maintaining a controlled temperature of 43 $^{\circ}C$ to which it was able to complete within a range of $\pm$ 2 $^{\circ}C$ around the set-point J.-X. Wang et al. (2016).

### 3.2.6 System Combinations

Internal S/C heat loading primarily coming from the high waste heat of components being used on board lead to issues especially when common waste heat flux for these platforms were around 10 $W/cm^2$ and projections at this time showed future platforms exhibiting heat flux on the order of 100 $W/cm^2$. With higher fluxes present, a greater distance over which the heat pipe must carry the waste heat is required due to low flux radiators, leading to the innovative combination of a pumped heat pipe cold plate for high flux applications (Ambrose et al., 1995). This combination of a two-phase pumped loop and heat pipe integrates high flux, detachable S/C payloads to a pumped thermal control loop (Ambrose et al., 1995).

For the on-orbit service market, autonomous thermal management provides the benefit of not having to continuously monitor your system and allow it to complete its mission with minimal oversight. W. Guo, Li, Li, Zhong, et al. (2017) presents an example of this idea with a Shape Memory Alloy (SMA) based adaptive thermal control coldplate module (TCCM). With use of this assembly, a self adaptive flow rate and temperature co-adjustment can be done. This was demonstrated via an experimental test-bed setup to interface with a single-phase mechanical pumped fluid loop (SPMPFL). ”...by applying a modular thermal control system, in which parallel installed TCCM array is cooperated with a SPMPFL, the thermal design of a spacecraft can be simplified into determining heat sink quantity according to integral thermal load, the temperature fluctuation of electronics can be less considered or even ignored due to the TCCM itself is capable to arrange.” (W. Guo, Li, Li, Zhong, et al., 2017). In a continuation of their previous work, W. Guo, Li, Li, Wang, et al. (2017) designed and evaluated a novel concept referred to as a flexible thermal control system (F-TCS). The F-TCS is composed of a heat collecting bus, heat dissipating bus, connection brunches, and an inter-platform service module (W. Guo, Li, Li, Wang, et al., 2017). A mathematical model was established and it was numerically analyzed to understand its thermal control dynamic characteristics and performance. Additionally, a physical experiment was also set up to verify and provide for technical feasibility.

In a continuation of their previous work, T. Y. Kim, Hyun, Lee, and Rhee (2013) further develop the heat pipe-phase change material (HP-PCM) hardware concept. To quantitatively investigate the device, a typical spacecraft radiator for the intermittent, high heat generating, component is analyzed numerically and the temperature changes for cases with the PCM implemented and those without are compared to verify the thermal effect. This new but less complex hardware provides as expected results with maximum temperatures decreasing and minimum temperatures increasing while also reducing component operating temperatures by 28 $°C$ with only a 9.8% increase in mass.

For high-power components, waste heat can be so intense that the internal cooling system is not sufficient to handle peak loading. In these circumstances, it can be valuable to look for components with a more managable heat load or to utilize components in a way that draws power more efficiently and produces less waste heat in operation. A novel technique was developed to optimize the efficiency of high-power equipment on board a geo-synchronous satellite by altering the means of operation Solid-State Power Amplifier which resulted in a reduction of 1000 W DC power and a 700 W decrease in power dissipation as heat as a result (Doshi & Ghodgaonkar, 2018).

While on-orbit operations brings its own thermal problems, similar advancement of new technologies and drive for solutions has occurred with regard to the re-entry phase of flight. These technology typically has applications for both phases and with appropriate analysis, could be feasible for use. Examples of such could be active thermal protection systems such as convective cooling, film cooling, and transpiration cooling or passive systems like heat sinks, hot structures, and insulated structures (Uyanna & Najafi, 2020). When looking at the internal problem, typically you look to reject heat from the system. Instead, Borshchev, Sorokin, and Belyavskii (2020) presents the use of power units that will run on the waste-heat produced by its internal components. In analysis it is found that, in steam-turbine systems, the unit area of the cooling radiator is greater than for a power unit operating by means of an ideal Carnot cycle such as this potential implementation. Therefore, comparison of the mass of radiators and solar panels for spacecraft in Earth orbit indicate that solar panels are the preferred means of generating power instead of attempting to use waste heat in a turbine system (Borshchev et al., 2020).

Varatharajoo, Kahle, and Fasoulas (2003) proposed the unique idea of coupling two different spacecraft subsystems, the thermal control system and attitude control system, allowing for reduced cost and decreased mass overall. By utilizing an electrical conductive fluid in a closed loop system, the fluid can serve as both a heat conductor and momentum generator, influenced by magnetic or electric fields and temperature gradients (Varatharajoo et al., 2003). The electrical influence can be utilized to generate angular momentum around the fluids circulation axis which could contribute to attitude control (Varatharajoo et al., 2003).

### 3.2.7 Radiators

The use of radiator serves as a direct means of rejecting heat back out to space. To fulfill the last part of the energy balance equation (Equation 1), energy out ($\dot{E}_{out}$), it is easiest to utilize the already harsh cold of space due to a large temperature differential, to radiate heat out and away from the vehicle avoiding overheating conditions interior to the S/C.

Chen, Huang, and Chen (2016) highlights the problem that conventional radiators alone are simply incapable of handling the increased loading being presented by modern day S/C components. Chen looks to the use of PCM to hopefully reduce radiator area and heating power required. Chen et al. (2016) start numerically to prove a PCM solution can improve thermal conditions on the FORMOSAT-5 satellite, and then experimentally evaluate four different candidate materials: N-octadecane, N-eicosane, Glycerin, and Gallium, to provide further proof that this PCM device is feasible for implementation.

Inamori, Ozaki, Saisutjarit, and Ohsaki (2015) provides a novel radiative cooling system aimed toward nano- and micro-spacecraft with low power consumption, small mass, and low cost. With strict power and mass requirements for these small platforms, this paper presents the idea of using a high temperature superconducting coil (HTS) to create a propulsive force. With no capability for an active cooling system, a passive system consisting of radiative cooling to the 3K cosmic background radiation of deep space, shielding from the sun, and insulation against heat generation using magnetic holders (Inamori et al., 2015). Analysis of this system shows the HTS is cooled to 60 K in interplanetary orbits.

Radiators capable of dynamically changing the amount of heat that they reject offers the potential to reduce the amount of heating required onboard to maintain survival temperatures and also further expands the mission profiles that can be attempted by various crafts. Mulford et al. (2020) developed and tested a prototype of an origami-inspired radiator to demonstrate the capability of maintaining temperatures through the expansion and contraction of a collapsible radiator controlling radiative heat loss. It is found that as the radiator actuates from extended to collapsed, the heat transfer decreases but the fin efficiency increases and find that the four panel prototype exhibits a turn-down ratio of 1.31 over a limited actuation range. This is further confirmed as the numerical model suggests that a turn-down ratio of 2.27 is feasible for a full range of positions and future revisions with 8 panels and high thermal conductivity would yield a ratio of 6.01 (Mulford et al., 2020).

Nagano, Ohnishi, and Nagasaka (2011) investigates the use of a new, re-deployable radiator with environment adaptive functions. This radiator is made of highly conductive graphite sheets and a single shape memory alloy to provide a passive reversible actuator, which also changes its function to a solar absorber by deploying or stowing the fin upon changes in heat dissipation and environment. Nagano et al. (2011) conclude analytically that the radiator is able to save heater power and that much larger savings than expected were found with 90% of heater power saved near earth orbit.

Petroffe et al. (2019) presents another variation of an adaptive radiator, this one utilizing electroemissive devices as the means for adjusting the radiators functionality. Electroemissive devices or EEDs acts as a electro-active material that changes emissivity of the radiator based on the voltage being sent to the device. In ”cold” states the radiator exhibited a $9°C$ increase compared to the optical solar reflector radiator, and showed an equal or greater rejection ability in the ”hot” case comparison (Petroffe et al., 2019).

Similar to others investigating adaptive radiator designs, Xu, Zhao, and Liu (2021) continues that research by presenting the idea of coating the radiator with a tunable-emittance film that can appropriately tune the radiators emittance according to the thermal conditions it is exposed to. This is accomplished by a near-field radiation assisted smart skin that through a metal-insulator semiconductor structure, can change emission by applying voltage and in experimentation, it show that the effective emittance can exceed 0.7 (Xu et al., 2021).

While radiators with PCM have been proposed before, they have been rarely used due to concerns about poor heat transfer through the material from poor thermal conductivity and an absences of models to better understand and evaluate their performance. Roy and Avanic (2006) develop a simple model that will provide a theoretical basis for a space radiator with latent heat thermal energy storage. This model allows for analysis and optimization of the radiator for a given heat storage and dissipation capability. Numerical evaluation of a typical configuration shows mass reductions of 20-25% can be achieved for pulsed heat loads of 1 hour duration or less and also show radiator sized decreases by a factor of 4 or more (Roy & Avanic, 2006).

Sun, Wang, Chen, and Xia (2016) presents another variation of an adaptive radiative shield to handle extreme environments experienced by an orbital transfer vehicle. With 7 adjustable parameters, the thermal control effect is better than using traditional insulation and the optimum combination of parameters can be determined through numerical analysis (Sun et al., 2016).

## 3.3 Comparatives

Throughout this section, numerous technologies were discussed, but the problem remains that it can be difficult to know where to actually begin on designing a system for a spacecraft. To assist with this, a series of figures were created providing comparatives among the various aspects of selection. While these figures do not provide all the direct information that would address the niche nuances of any and all missions, they do provide the context necessary to begin a design with some base level of knowledge, allowing an effort to progress toward a solution that will work for their design.

Firstly, in assessing environments, the three orbits previously mentioned (Low Earth, Molinya, and Geosynchronous) are broken down to show typical radiative heat loading that can be found at the different points. Figure 22 shows the relationship of heat loading to distance from Earth, exemplifying how the influences of albedo and OLR diminish as apogee is approached. Additionally, Figure 23 shows the removal of the solar irradiation component which occurs during an eclipsed portion of flight, and results in a large decrease in energy values. It is important to note that once leaving Earth orbit, the problem does change and these values will change accordingly.

Next, when looking internal to the vehicle itself, the anticipated size of the vehicle can provide some initial idea as to the conditions that will be present inside due to the various components required for its operation. Utilizing sizing information and example mission information provided by Kopacz et al. (2020) and W.-J. Li et al. (2019), in addition to example component

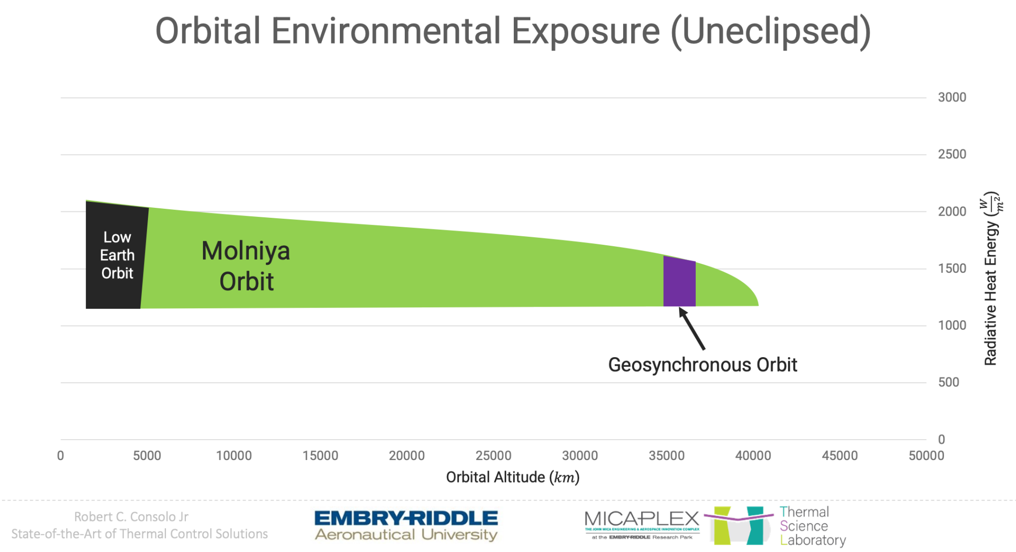


Figure 22: Orbital Environmental Exposure (Uneclipsed)

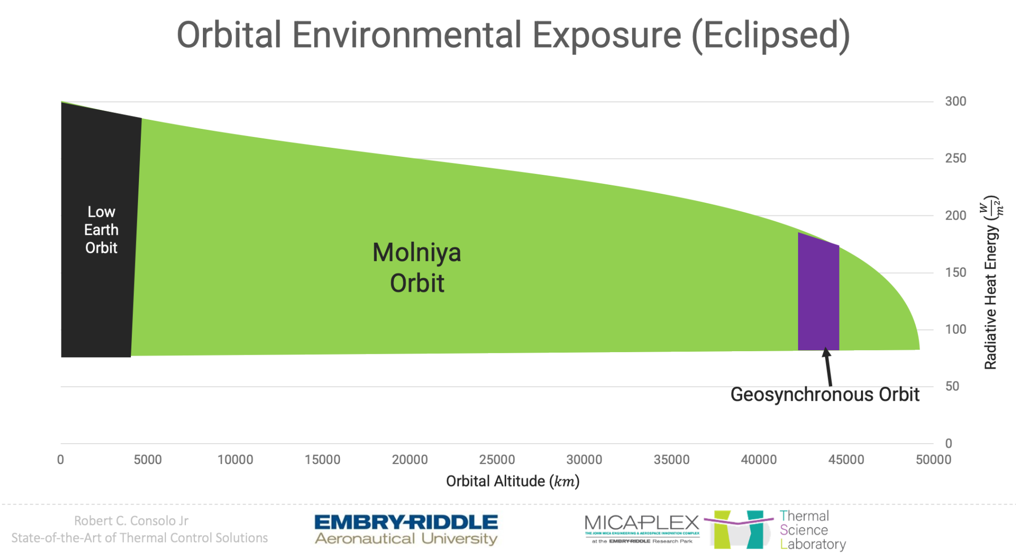


Figure 23: Orbital Environmental Exposure (Eclipsed)

selections from various sources like Corpino et al. (2015), potential waste heat loading, generated by the operation of supporting components, were assessed in correlation to vehicle mass in Figure 24.

Furthermore, an understanding of the supporting systems for the spacecraft as a whole and their operational parameters provides the context needed to better understand internal conditions.

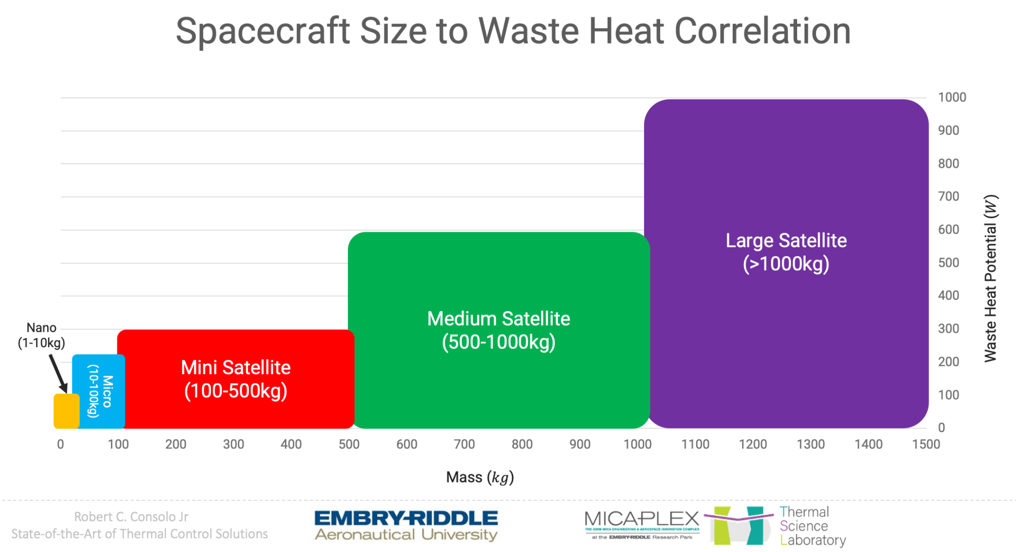


Figure 24: Spacecraft Size to Waste Heat Correlation

Power conversion systems tend to be the systems that not only put out the most heat due to their high power input and output, but also tend to have the tightest operating conditions (i.e. temperatures) as well, shown by both Alcayde et al. (2021) and Corpino et al. (2015). These requirements are shown graphically to further advise on how to proceed with a thermal management strategy dependent on the supporting infrastructure of the vehicle in question. With an established knowledge of the environments both externally and internally, deciding on which methods of the thermal management to pursue is the next hurdle.

Space allocation is one of the first disqualifying factors for selection due to certain, more complex, solutions require additional supporting equipment, and thus requiring more volume (i.e. thermal devices), more mass (i.e. highly conductive materials), or both (i.e. active systems). Utilizing this, while also acknowledging various reported sizes of test systems or integrated flight technologies, a comparative figure was created to show these constraints by investigating various implementations of the systems, both in test and in flight, gathering dimensional information on those that showed successful performance and drawing assumptions as to how some of these systems would act in tighter configurations.

Both operating requirements for internal components and space allocations for thermal management systems are graphically represented by Figure 25 and Figure 26. It is important to note that these ranges do encompass various implementations of a similar solution so it is important to assess which specific configuration works best for your system. For example, the novel spray cooling system would require less mass comparatively to a full flow, two-phase system due to a reduced fluid mass requirement and thus fall at the lower end of the active systems block.

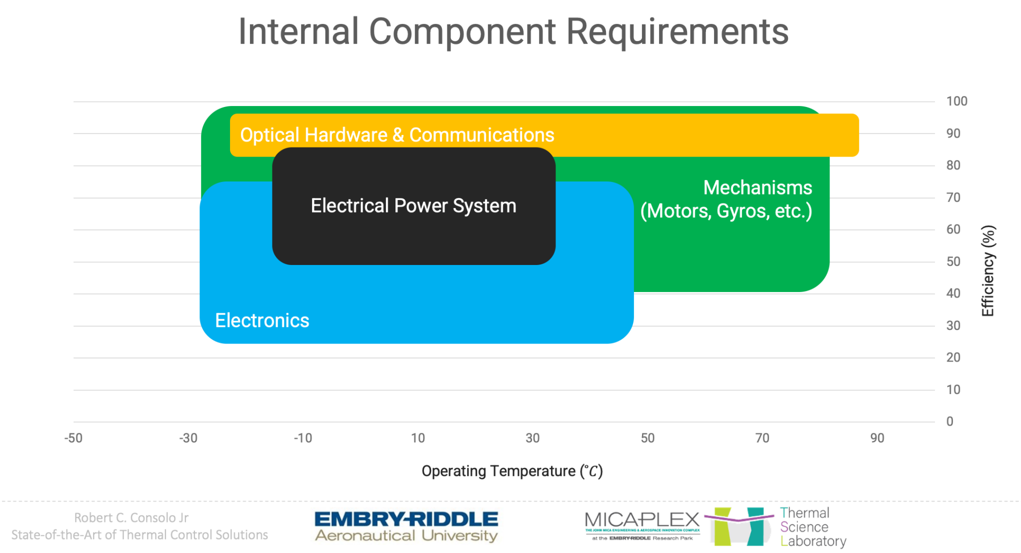


Figure 25: Internal Component Requirements

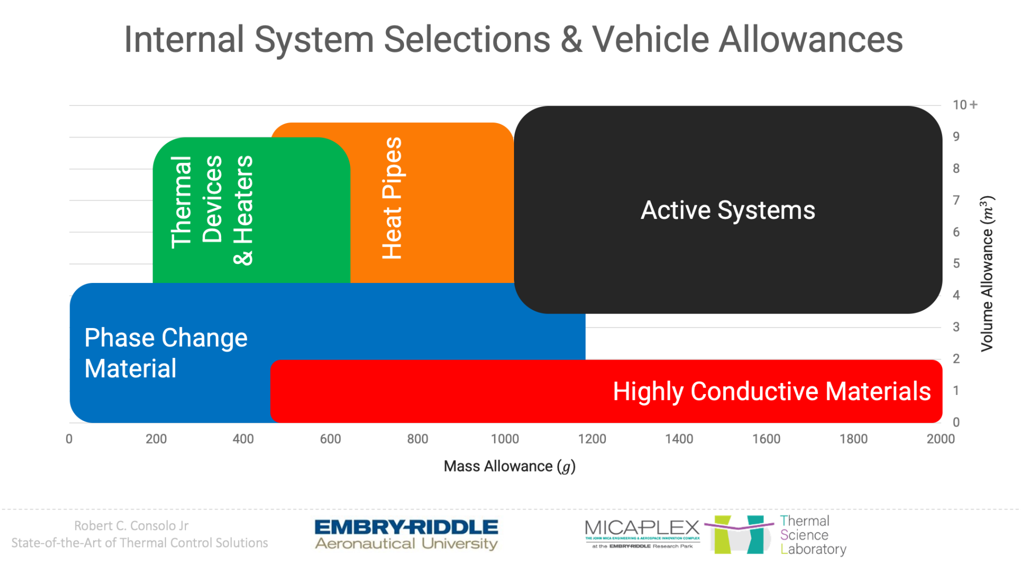


Figure 26: Internal System Selections  Vehicle Allowances

Lastly, with the added knowledge provided by the previous figures, more discrete selections of thermal solutions can be made for further analysis. Further comparatives were made by pulling the combined information from all the papers utilized in this section to critically analyze and establish likely performance parameters of both passive and active thermal management systems as well as rejection systems. The passive and active systems are evaluated by comparing heat loading on the system versus the temperature change required by the protected system, negative showing cooling and positive showing heating. Passive and Active system performance is graphically shown in Figures 27 and 28. Similar to the previous figure showing space allowances, these figures were created by investigating various implementations of the systems, both in test and in flight, providing a range of performance values that have been exhibited by these systems in practice. It is important that while these can serve as a guide to make decisions on how to proceed, all decisions should be followed by a thorough analysis of the configuration to ensure desired performance in flight.

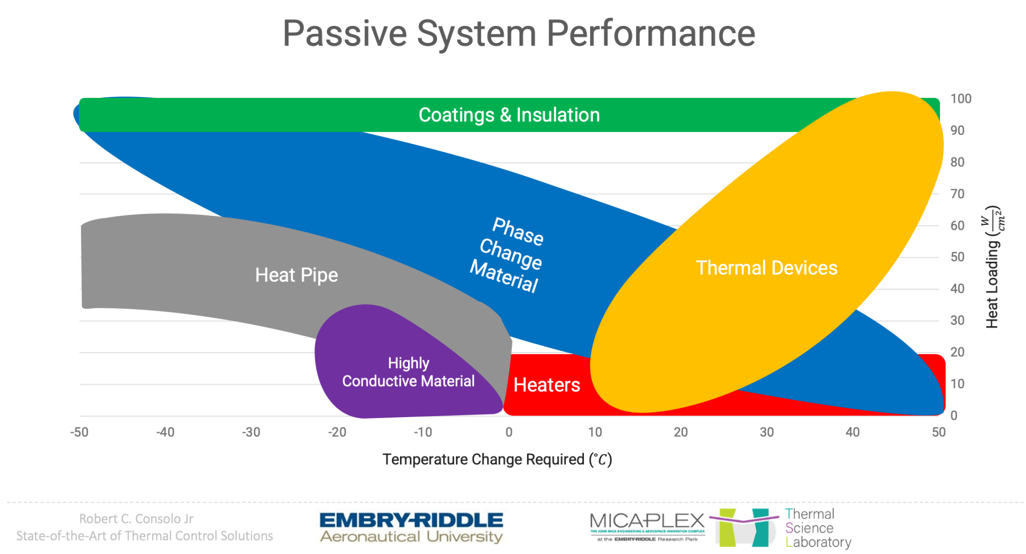


Figure 27: Passive System Performance

For rejection, a traditional radiator and a radiator integrated with phase change material are compared under steady state conditions, showing mass per unit heat transfer over a specified amount of time heated. The comparison is driven by data collected by Roy and Avanic (2006)

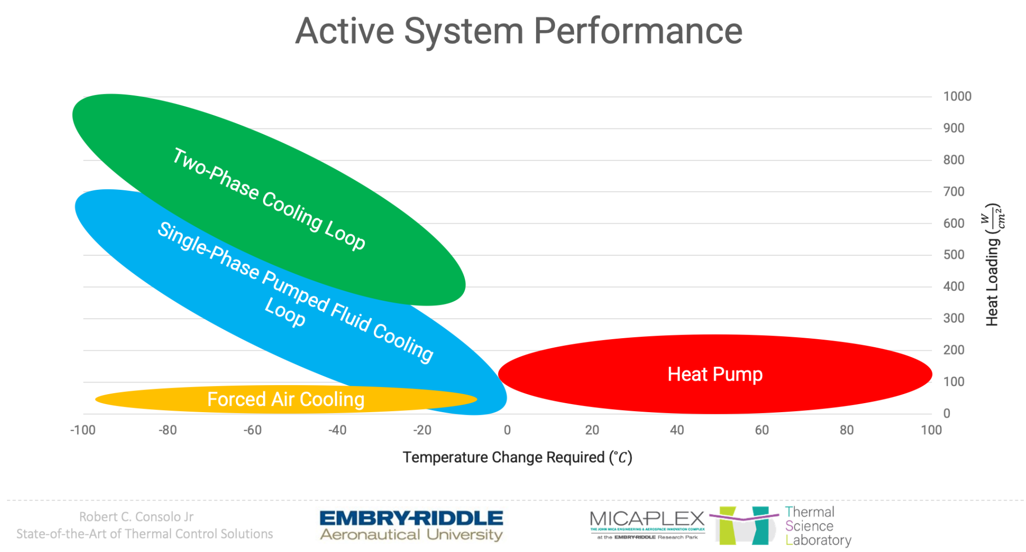


Figure 28: Active System Performance

with similar results being shown in testing by different variations of PCM implementation in other applications, thus the figure is simplified to show the difference in performance for a range of PCM implementations rather than each individually. This is graphically represented by Figure 29.

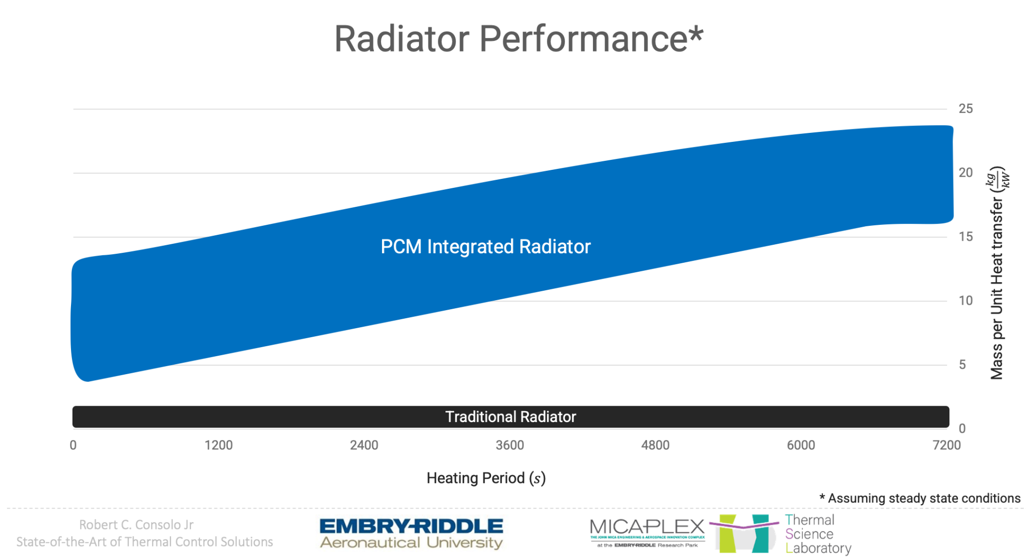


Figure 29: Radiator Performance

The figures provided above provide a starting point for smaller, inexperienced providers and allow those who are still contained to a concept phase to begin making selections for analysis that ideally will reduce redundant evaluation and minimize cost. Combined with a design methodology , these figures followed by a thorough analysis will be helpful to those furthering the design of new craft, providing solutions for more niche requirements of certain mission sets.

# 4 Design Methodology for Spacecraft Thermal Control

When evaluating a system such as spacecraft, it comes with a requirement for a high degree of accuracy and acknowledgement of risk. ”Uncertainty impacts the decisions engineers and managers make...when engineers and managers are risk-tolerant in the context of uncertainty, decisions are made that might ultimately result in systems that are over budget, delivered late, descoped, or even canceled.” (Thunnissen, Au, & Tsuyuki, 2007). Now on the other end of the spectrum, being risk-averse, uncertainty can lead to over-design, uncompetitive products that are not best suited to meet requirements (Thunnissen et al., 2007). Zaidi, Fitz-Coy, and van Zyl (2018) points out that additionally, with many of these smaller satellites using commercial grade components, there is a level of risk associated with it due to it operating in the harsh environment of space, for which it is not rated. ”In each [orbit], if no abrupt variation in solar weather occurs, then the dynamic thermal loading on a spacecraft can be considered deterministic and enforceable for a fixed duration.” (Zaidi et al., 2018).

The points made by Thunnissen et al. (2007) and Zaidi et al. (2018) bring us to step one of the design methodology:

1. Define the mission(s) for the spacecraft including, but not limited to, desired orbital parameters, vehicle sizing, scientific goals, and power requirements.

Defining the mission(s) for the S/C is the true gateway to a multi-mission/multi-orbit platform by providing the framework necessary to not only design a thermal control system that will work for your vehicle, but rather ensure that you do not over/under design which could lead to cost overruns, redesigning, and further infringing on flight opportunities. Desired orbital parameters and vehicle sizing both provide the information that would permit the start of environmental analysis. Utilizing, for example, Satellite Tool Kit and ANSYS as mentioned previously can provide total energy values for you S/C’s full mission profile and provide temperatures at its harshest highs and lows (Ingwersen, 2021) thus providing the information necessary to begin making protection system selections.

Vehicle sizing is also applicable in determining the volume availability for the thermal control system. Revisiting an excerpt earlier from Kahn et al. (2017), Smallsats have to be capable of completing their objectives yet fit within the limited mass and volume allowances provided by either the core S/C bus or the launch vehicle secondary payload requirements, and with small satellites on the rise, the problem of available space is more present than before. With respect to the launch vehicle, the problem continues beyond the actual volume of the S/C, because mass has a direct impact on the cost to launch. Hengeveld et al. (2010) shows this, referencing the SpaceX Falcon 1, which at the time would provide launch services to a 450kg payload to LEO for $7.9 million or $16,667 per kg.

In another attempt to analytically understand thermal influence on a satellite, Farrahi and Pérez-Grande (2017) developed a semi-analytic model that can predict the temperature variation and the influence of a satellite's spin rate on temperature variation. Comparing numerical results with the model solution, Farrahi and Pérez-Grande (2017) found less than 1 percent of error was found. It was also found that a satellite flying over a morning sun-synchronous orbit reaches it's steady-state sooner than the one flying over a midnight orbit with the same altitude.

Gadalla (2005) investigated the effects of rotating the spacecraft on temperature variation. The problem was broken down analytically and, inspite of the non-linearity and non-homogeneity of the boundary conditions, an exact solution is found. It is found that the temperature distribution on the exterior surface is nearly independent of the angular position (Gadalla, 2005). The analysis showed that spinning speeds significantly diminish and shift the location of temperature maximum and minimum on the interior but also cause larger temperature gradients as that speed increases (Gadalla, 2005). Once a rotation over 1000 rad/h is achieved, variation on the outer surface achieves independence of the angular position while the interior reflects a similar effect at only 20 rad/h. (Gadalla, 2005).

Yamada and Nagano (2015) speak to why the thermal condition of small satellites is more difficult to control stating that it's "...because of (i) limited power resources, (ii) small heat capacity, (iii) insufficient radiator area, (iv) high-density packing of electronics, and (v) mass limitations." To address these concerns, Yamada and Nagano (2015) presents the design of a heat storage panel, a thin carbon-fiber-reinforced polymer panel with PCM encapsulated in it. This panel was tested both here on the ground and in orbit on the Hodoyoshi-4 satellite and showed in both that the HSP has high potential for thermal control on small satellites.

A contributor to thermal rejection capability of S/C that may go unnoticed by some is the actual layout of the thermal control system on board. Escobar, Diaz, and Zagal (2016) presents a method, using genetic algorithms, to assist in automating the design of a satellites passive thermal control system. The genetic algorithm will evaluate a space provided to it defined by varying surface paint tiles made with different paint materials for possible candidate solutions and then evaluated using a thermal finite element method. To then validate this method, Escobar et al. (2016) set up a real physical experiment in a vacuum chamber and tested its thermal behavior. The result shower that after calibration, the model would obtain solutions with high levels of accuracy exhibiting mean squared errors of 1.45 K for sunlit faces and 2.4 K for shadowed faces (Escobar et al., 2016).

Lastly, scientific goals and power requirements are considered to be tightly intertwined as typically the scientific goals will dictate the components or payloads required onboard which then determines the power loading. Components utilizing power will naturally generate waste heat in operation as no system is 100% efficient (Ambrose et al., 1995). The heat loading caused by the waste heat will be a large contributor to your internal thermal solution decisions.

Once the mission is defined, the next ideal step is:

2. Establish the thermal operating range for each of the onboard components and plan your thermal management system around maintaining the range that is tightest and interior to all other ranges.

Batteries are one of the most critical components of a spacecrafts power system but the operating performance of these batteries are contingent on its temperature level (S. Wang et al., 2017). Large temperature differences in power cells lead to reduction of life cycle and a decrease in energy conversion efficiency leading S. Wang et al. (2017) to propose an integrated cooling method that combines forced internal gas cooling and a liquid cooling plate to mitigate thermal concers for a Li-ion battery. With a total of 576 W of heat generation coming from the battery, the maximum temperature and general temperature difference can be decreased by 3.45 K and 3.88 K respectively when compared to a traditional vacuum packaged cooling method (S. Wang et al., 2017). Additionally the temperature uniformity and control effectiveness can be increased by 2.42 times and 2.61 times respectively (S. Wang et al., 2017).

Alcayde et al. (2021) highlighted this earlier and showed a similar methodology by taking the thermal operating range for the electronics, batteries, and mechanisms, framing the analysis around the rechargeable batteries which can operate in $0°C$ and $20°C$. Batteries are typically going to be one of the most sensitive components onboard a spacecraft but also tends to be necessary to maintain operation in eclipsed portions of flight. With this tight range, it may require the utilization of heater but the problem of needing heaters at low thermal loading brings the additional mass of supporting components further complicating overall spacecraft designs and space constraints (Stavely & Lesieutre, 2013).

As we look terrestrially for more ideas in thermal management, the automotive industry frequently becomes an asset, especially the Electric Vehicle (EV) market due to the complexity of these systems and the impact that thermal can impose on the overall operation. Requirements for these propulsion systems such as power density, switching frequency, and cost are consistently becoming more stringent and as a consequence of these stringent conditions reliability of certain components are becoming jeopardized due to thermal issues. For example, approximately 60% of power semiconductor breakages are a consequence of thermal issues and an increment of $10°C$ in operation mean temperature can double its failure-ratio (Trancho et al., 2020).

Now with a defined mission providing external boundary conditions and component operating conditions providing internal requirements, it is possible to move into the next step of the methodology:

3. Evaluate the thermal problem of external and internal loading separately. Make thermal control selections appropriate to the conditions.

To understand this further, we must look back at Equation 1 highlighted again below as Equation 6.

$$\dot{E}_{in} + \dot{E}_{gen} = \dot{E}_{stored} + \dot{E}_{out} \tag{6}$$

To isolate the internal problem, we must evaluate and provide solutions for the external conditions of the S/C first. While making the assumption that the entirety of the S/C is enclosed, effort can be dedicated to preventing both the minimum and maximum energy loads from interacting with the S/C and ideally creating an adiabatic wall. An adiabatic boundary condition or, a wall that does not permit heat transfer, allows the developer to isolate the internal problem independent of the environmental factors. It is important to note that this assumption can readily change throughout the flight. As shown by Xu et al. (2021) and Böhnke et al. (2008), there are examples in industry that developers have made methods to control the emittance and absorptance of the vehicle whether it be by a specific combination of coatings and insulation (Böhnke et al., 2008) or by an electrically tunable film that can readily change based on the voltage supplied to it (Xu et al., 2021). With this acknowledged, once this assumption is made, the equation simplifies as represented in Equation 7:

$$\dot{E}_{gen} - \dot{E}_{out} = \dot{E}_{stored} \tag{7}$$

The energy generated represents the waste heat from the various components onboard which, if using high-power systems in support of scientific payloads, can equate to a rather high thermal loading opening the vehicle up to potential overheating. As for stored energy, this thermal loading plays a critical role for about 50% of the S/C's flight. Stored energy provides the opportunity to retain some amount of heat energy that can continue to keep components above the lower end of their operating range when exposed to the harsh cold space environment that is present when no longer exposed to the sun. With the assumption of the adiabatic wall, this simplifies to just represent the heat energy that is to be rejected ($\dot{E}_{out}$) by the radiator in the thermal control system. Additionally, if utilizing PCM in your thermal control system, the stored energy for these components can be accounted for here and with larger heat capacity in some PCMs, this can add up. By solving for the rate of energy generation and subtracting the capability of heat rejection by the radiator and/or any other rejection systems onboard can provide a loading that is retained by the vehicle, allowing us to calculate the internal temperature of the vehicle.

While a highly effective thermal control system is required for spaceflight in general, in some cases, a system can also be deemed too effective. Bertagne, Cognata, Sheth, Dinsmore, and Hartl (2017) speaks about a constant rejection system would put this particular case of a crewed vehicle in a predicament because, while in planetary surface operations (PSO) phase the environment is much warmer, during the trans-planetary coast (TPC) phase of this flight, the vehicle will be subject to much lower temperatures requiring the ability to modify how much heat is being rejected. While in PSO, the heat rejection requirement for the mission in question is constant at 5400 W while during the TPC phase, that requirement drops to 1000 W. This ratio of 5.4:1 is referred to the turndown ratio or the ratio between the maximum and minimum heat rejection capabilities of the TCS in the warmest and coolest environments respectively (Bertagne et al., 2017). It is expected that future missions will require ratios between 6:1 and 12:1 dramatically increased from our first experience with this problem with travel to the moon during the Apollo era (1.6:1). This drove the design and testing of a morphable radiator that would be able to control how much heat is rejected or retained as to keep components healthy for the entire duration of the flight.

To best understand the performance of various components or operational methodologies prior to their use in spaceflight, it is beneficial to look at their performance outside of the industry in addition to inside and assimilate their comparisons. Looking as far as the operating room, where the patient can be viewed as a heat storage body and thermoregulatory mechanisms are put in place prevent inadvertent perioperative hypothermia (Torossian, 2008). With respect to passive warming methods, insulating blankets are used here as well, reducing heat loss by about 30% (Torossian, 2008). Additionally, the use of active methods such as radiant heaters and warmed fluids are used in combination with passive methods to maximize heat preservation (Torossian, 2008). A operational characteristic that you don't see all too often in the space industry is the idea of pre-warming (Torossian, 2008) such as taking action prior to a previously identified cooling situation that can allow for you to preemptively take steps to minimize cooling impacts and afford more time prior to exceeding the bounds of your operational parameters.

4. To promote fault tolerance and mission success, utilize a combination of systems or operational methodologies especially if utilizing an active system.

Active thermal control systems are a primary means of transporting heat away from multiple sources to a rejection system (typically a radiator). To transport this heat, via a selected working fluid, requires multiple supporting components in which some are moving parts like pumps

Murthy, Sharma, Badarinarayana, and Lakshminarasimhan (2011) provides further justification to thermal control system choices speaking to a geo-stationary communications satellite. After identifying the spacecraft will be exposed to temperatures ranging from 2.76 K to 5800 K, Murthy et al. (2011) choose to wrap the craft in MLI and optical solar reflector windows while heat pipes and high emittance surfaces are used to spread the concentrated heat coming from components. Additionally, heaters are provided to handle the lower boundary of the temperature range that the spacecraft will be exposed to.

There are cases where certain electronic components will only periodically operate. For example, on observation satellites, these components will only operate when traversing over a specific region in which it is attempting to capture an image of. With temperatures rising to their peaks during operation, using radiators to discharge that heat, and hitting their lowest at the end of cooling, using heaters to keep temperatures up. To supplement these systems, T. Kim (2013) presents the idea of using a thermal buffer mass with high thermal capacities that can suppress temperature rises, reduce power consumption in cooling phases, and mitigate thermal shocks from rapid changes. T. Kim (2013) finds that as the thermal capacity of the mass increases, changes to the heating phase temperature profiles are small, but the cooling rate does slow down and reduces the duty cycle of the compensation heaters.

Lastly, to promote modularity in S/C and open opportunities to further extend the lifespan of vehicles and extend their mission;

5. Promoting modularity in the S/C opens up the possibility of replacing outdated/damaged components and keep vehicle at 100% through the act of On-Orbit Servicing. Utilization of certain passive components to interface with active systems (if applicable) allows designs of the system to consist of modular boxes that can readily be replaced to provide new services for a new mission or repair the S/C in the event of failure.

The concept of on-orbit servicing was first proposed in the 1960s and has been implemented in many case to include: Skylab, Hubble Space Telescope, Solar Maximum Satellite, and the International Space Station (W.-J. Li et al., 2019). The development and use of the space shuttle drastically pushed forward large advancements in the development of on-orbit servicing technology (W.-J. Li et al., 2019). The idea of multi-mission modular spacecraft was introduced by Goddard Space Flight Center in 1975 with the key point of the spacecraft bus being serviceability in space (W.-J. Li et al., 2019). On the iBOSS mission, an integrated interface provided heat transportation using high-thermal conductivity material that would form a heat loop among different blocks and support thermal management of the total system. This modular integrated interface could be applied to develop a modular spacecraft or modular thermal components (W.-J. Li et al., 2019).

Active solutions can be difficult to make modular due the working fluid itself. Having fluid in the system necessitates a requirement to have multiple quick-disconnect (QD) connections on the fluid lines entering and exiting that module of the system. These QDs can be difficult to manipulate and, if the system is not appropriately safed before any repair operations, run the risk of potentially exposing the highly pressurized lines to the vacuum of space, draining your fluid loop of all working fluid and leaving the system dry. This doesn't mean that it is impossible to create a modular active thermal control system.

For example, the International Space Station has shown this to be possible with their Pump Module (PM) for their External Thermal Control System (ETCS). The pump module serves as the heart of the mechanically pumped fluid cooling loop having two in operation (one for each thermal cooling loop) during nominal operations. When one has failed, there is a limited operation state in which the vehicle can remain operational until an Extravehicular Activity (EVA) has taken place to repair the module and bring the failed cooling loop online (Bruckner & Manco, 2014). Bruckner and Manco (2014) speaks to this replacement unit further: "The pump module is an Orbital Replacement Unit (ORU) designed so that it could be exchanged with spare units ...The pump module ORU houses the pump, controls, and valves required to maintain proper temperature in the heat rejection loop. A canned motor cartridge pump circulates this heat transfer fluid through the thermal control system." While the ability to replace modular ORUs to keep various systems functioning is a great benefit, the hazards of a pressurized fluid system, especially one filled with a hazardous working fluid, comes at a cost of time and risk. "Astronauts Doug Wheelock and Tracy Caldwell-Dyson performed three EVAs totaling over 25 hours to replace the failed pump module with a spare ORU. The EVAs proved to be much more difficult and dangerous than anticipated due to frozen ammonia that jammed the quick disconnect fluid couplings used to connect the pump module to the rest of the thermal control loop. During the de-mating procedure on the failed unit, frozen ammonia particles were liberated from the connectors and caused both a micrometeorite type of hazard for the two astronauts and a health concern upon re-entry to the ISS as the frozen ammonia particles were stuck to the outside of the spacesuits. On August 17, 2010 function of cooling Loop A was restored and ISS operations returned to nominal.” (Bruckner & Manco, 2014).

With this in mind, the design methodology for modularity focuses on passive systems interacting with an active transport system similar to the concept highlighted by W.-J. Li et al. (2019) in the iBOSS mission. A prototype schematic of a modular thermal control system was drawn up by the Embry-Riddle Aeronautical University Thermal Science Laboratory after being approached by a company that was desiring a thermal control system that would be capable of keeping components operable both on the core bus and on modular ”arms” of the S/C that would each be providing its own service to various other craft. These arms would contain various electronic components and ideally receive power from the core spacecraft bus. This S/C was expected to be capable of thermally protecting itself in a variety of near-earth orbit configurations. With this knowledge, it was identified that a modular active system would be complex and costly, both monetarily and in mass. The idea was then discussed to use a hybrid modular system utilizing both passive and active solutions.

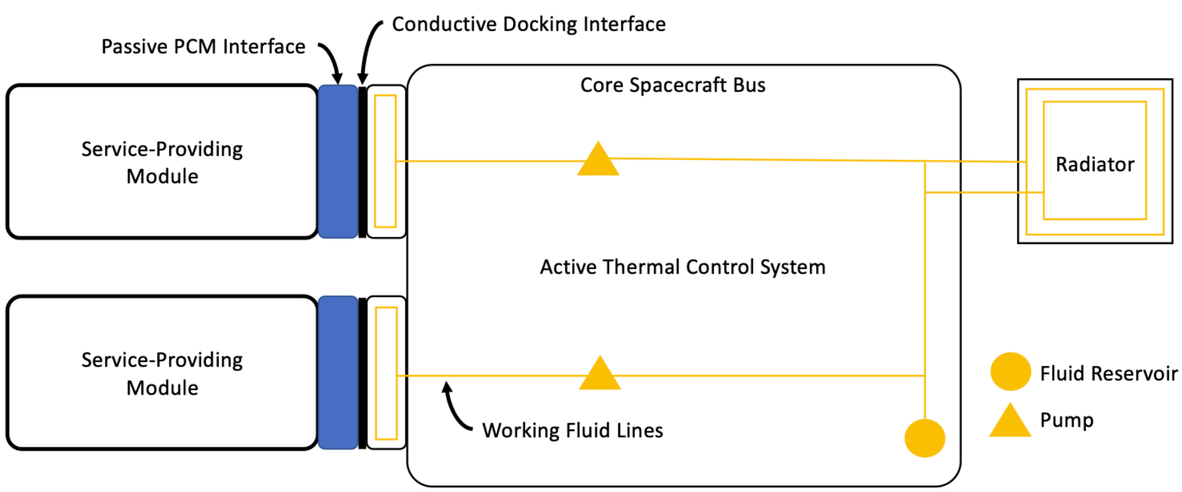


Figure 30: Schematic representation of Modular Hybrid Thermal Control System

This design asked that each ”arm” module be designed with a PCM interface at its docking end and called for the docking interface to contain a conductive material to allow for less thermal resistance when transferring loading over to the core bus. At the core bus docking interface, cooling lines would be in place to collect heat energy from the PCM and transport that heat to a radiator on an always eclipsed side of the vehicle ensuring exposure to the coldest temperatures and dropping the temperature of the now heated working fluid. In the meantime, it would be expected that the PCM would retain some amount of heat from operation to ensure that when eclipsed, the various components in these ”arm” modules would be kept within their operating range at the harsh minimum temperatures. This design can be further referenced in Figure 30. The next step for the Thermal Sciences Laboratory is to await the final mission definitions and vehicle geometries from the company as to begin the analysis of environments at various orbits, thus beginning the design methodology from step 1. Utilizing information from this comprehensive review, the team can also begin evaluating various space-capable PCMs if the service-providing components have been selected.

# 5 Conclusion

The purpose of this study was to perform a comprehensive review of technologies available today for thermal management onboard S/C in addition to investigating the means to analyzing the environments in orbit and making the determination on a design methodology that would support the development of efficient and effective future spacecraft thermal control systems. A combination of active and passive thermal solutions were investigated that would best assist onboard components in maintaining operable thermal ranges in these varied environments in addition to a breakdown of the supporting equations that best determine the thermal conditions spacecraft are exposed to. Modern day methods of analyzing and understanding these environments were looked at as to provide an insight as to what may be available for both the new and experienced developer.

Analytical methods varied based around a reference point but the outcomes were similar in that the primary concern of heat loading in space is radiative heating from varied sources such as the sun, the planets, as well as cosmic background radiation if traveling beyond Earth orbit. Numerically, industry has continued to find new ways of understanding environments prior to launch whether it be through analytical estimation, numerical tools created through known outlets like MATLAB, or new modern analysis platforms as provided by ANSYS/STK.

Thermal control solutions consisted of coatings, insulation, heat pipes, phase change material, conductive materials, thermal devices, actively pumped fluid loops, radiators, and combinations of these systems. Such a large selection exists due to the largely niche nature of spaceflight as the goals set by one vehicle are likely not be shared directly with another, making each design a new project requiring its own system to accommodate its needs.

With numerous technologies identified, the problem remained that it can be difficult to actually begin on a design so a series of figures were created providing comparatives among the various aspects of selection. Lastly, utilizing the knowledge gained from such a wide-net review of thermal control solutions available today, both in space and terrestrially, a design methodology was established:

1. Define the mission(s) for the spacecraft including, but not limited to, desired orbital parameters, vehicle sizing, scientific goals, and power requirements.
2. Establish the thermal operating range for each of the onboard components and plan your thermal management system around maintaining the range that is tightest and interior to all other ranges.
3. Evaluate the thermal problem of external and internal loading separately.
4. To promote fault tolerance and mission success, utilize a combination of systems or operational methodologies especially if utilizing an active system.
5. Promoting modularity in the S/C opens up the possibility of replacing outdated/damaged components and keep vehicle at 100% through the act of On-Orbit Servicing. Utilization of certain passive components to interface with active systems (if applicable) allows designs of the system to consist of modular boxes that can readily be replaced to provide new services for a new mission or repair the S/C in the event of failure.

This methodology looks to be a means of further guiding new providers and developers through the design, analysis, development, and launch of a more efficient and effective system that can best equip their S/C for mission success.

# 6 Appendix

Table 3: Earth IR and Albedo, 3.3$\sigma$ Values, Cold Case (Gilmore, 2002)

| | | Inclination (deg) | | | | | |
|---|---|---|---|---|---|---|---|
| | | 0-30 | | 30-60 | | 60-90 | |
| Surface Sensitivity | Time Period | Albedo | IR ($W/m^2$) | Albedo | IR ($W/m^2$) | Albedo | IR ($W/m^2$) |
| Albedo | 16 sec | 0.06 | 273 | 0.06 | 273 | 0.06 | 273 |
| | 128 sec | 0.06 | 273 | 0.06 | 273 | 0.06 | 273 |
| | 896 sec | 0.07 | 265 | 0.08 | 262 | 0.09 | 264 |
| | 30 min | 0.08 | 261 | 0.12 | 246 | 0.13 | 246 |
| | 90 min | 0.11 | 258 | 0.16 | 239 | 0.16 | 231 |
| | 6 h | 0.14 | 245 | 0.18 | 238 | 0.18 | 231 |
| | 24 h | 0.16 | 240 | 0.19 | 233 | 0.18 | 231 |
| IR | 16 sec | 0.40 | 150 | 0.40 | 151 | 0.40 | 108 |
| | 128 sec | 0.38 | 154 | 0.38 | 155 | 0.38 | 111 |
| | 896 sec | 0.33 | 173 | 0.34 | 163 | 0.33 | 148 |
| | 30 min | 0.30 | 188 | 0.27 | 176 | 0.31 | 175 |
| | 90 min | 0.25 | 206 | 0.30 | 200 | 0.26 | 193 |
| | 6 h | 0.19 | 224 | 0.31 | 207 | 0.27 | 202 |
| | 24 h | 0.18 | 230 | 0.25 | 210 | 0.24 | 205 |
| Both | 16 sec | 0.13 | 225 | 0.15 | 213 | 0.16 | 212 |
| | 128 sec | 0.13 | 226 | 0.15 | 213 | 0.16 | 212 |
| | 896 sec | 0.14 | 227 | 0.17 | 217 | 0.17 | 218 |
| | 30 min | 0.14 | 228 | 0.18 | 217 | 0.18 | 218 |
| | 90 min | 0.14 | 228 | 0.19 | 218 | 0.19 | 218 |
| | 6 h | 0.16 | 232 | 0.19 | 221 | 0.20 | 224 |
| | 24 h | 0.16 | 235 | 0.20 | 223 | 0.20 | 224 |

Table 4: Earth IR and Albedo, 3.3$\sigma$ Values, Hot Case (Gilmore, 2002)

| | | Inclination (deg) | | | | | |
|---|---|---|---|---|---|---|---|
| | | 0-30 | | 30-60 | | 60-90 | |
| Surface Sensitivity | Time Period | Albedo | IR ($W/m^2$) | Albedo | IR ($W/m^2$) | Albedo | IR ($W/m^2$) |
| Albedo | 16 sec | 0.43 | 182 | 0.48 | 180 | 0.50 | 180 |
| | 128 sec | 0.42 | 181 | 0.47 | 180 | 0.49 | 184 |
| | 896 sec | 0.37 | 219 | 0.36 | 192 | 0.35 | 202 |
| | 30 min | 0.33 | 219 | 0.34 | 205 | 0.33 | 204 |
| | 90 min | 0.28 | 237 | 0.31 | 204 | 0.28 | 214 |
| | 6 h | 0.23 | 248 | 0.31 | 212 | 0.27 | 218 |
| | 24 h | 0.22 | 251 | 0.28 | 224 | 0.24 | 224 |
| IR | 16 sec | 0.22 | 331 | 0.21 | 332 | 0.22 | 332 |
| | 128 sec | 0.22 | 326 | 0.22 | 331 | 0.22 | 331 |
| | 896 sec | 0.22 | 318 | 0.22 | 297 | 0.20 | 294 |
| | 30 min | 0.17 | 297 | 0.21 | 282 | 0.20 | 284 |
| | 90 min | 0.20 | 285 | 0.22 | 274 | 0.22 | 250 |
| | 6 h | 0.19 | 269 | 0.21 | 249 | 0.22 | 221 |
| | 24 h | 0.19 | 262 | 0.21 | 245 | 0.20 | 217 |
| Both | 16 sec | 0.30 | 298 | 0.31 | 267 | 0.32 | 263 |
| | 128 sec | 0.29 | 295 | 0.30 | 265 | 0.31 | 262 |
| | 896 sec | 0.28 | 291 | 0.28 | 258 | 0.28 | 259 |
| | 30 min | 0.26 | 284 | 0.28 | 261 | 0.27 | 260 |
| | 90 min | 0.24 | 275 | 0.26 | 257 | 0.26 | 244 |
| | 6 h | 0.21 | 264 | 0.24 | 248 | 0.24 | 233 |
| | 24 h | 0.20 | 260 | 0.24 | 247 | 0.23 | 232 |

Table 5: Earth IR and Albedo, $2\sigma$ Values, Cold Case (Gilmore, 2002)

| | | Inclination (deg) | | | | | |
|---|---|---|---|---|---|---|---|
| | | 0-30 | | 30-60 | | 60-90 | |
| Surface Sensitivity | Time Period | Albedo | IR ($W/m^2$) | Albedo | IR ($W/m^2$) | Albedo | IR ($W/m^2$) |
| Albedo | 16 sec | 0.09 | 270 | 0.10 | 267 | 0.10 | 267 |
| | 128 sec | 0.09 | 267 | 0.10 | 265 | 0.10 | 265 |
| | 896 sec | 0.10 | 261 | 0.13 | 252 | 0.14 | 252 |
| | 30 min | 0.12 | 257 | 0.16 | 242 | 0.17 | 244 |
| | 90 min | 0.13 | 249 | 0.18 | 238 | 0.18 | 230 |
| | 6 h | 0.15 | 241 | 0.19 | 233 | 0.19 | 230 |
| | 24 h | 0.16 | 240 | 0.19 | 235 | 0.19 | 230 |
| IR | 16 sec | 0.30 | 195 | 0.33 | 183 | 0.35 | 164 |
| | 128 sec | 0.29 | 198 | 0.33 | 184 | 0.34 | 164 |
| | 896 sec | 0.26 | 209 | 0.28 | 189 | 0.27 | 172 |
| | 30 min | 0.23 | 216 | 0.25 | 200 | 0.25 | 190 |
| | 90 min | 0.20 | 225 | 0.23 | 209 | 0.24 | 202 |
| | 6 h | 0.18 | 231 | 0.23 | 212 | 0.23 | 205 |
| | 24 h | 0.17 | 233 | 0.23 | 212 | 0.23 | 207 |
| Both | 16 sec | 0.15 | 236 | 0.19 | 227 | 0.20 | 225 |
| | 128 sec | 0.16 | 236 | 0.19 | 227 | 0.20 | 225 |
| | 896 sec | 0.16 | 237 | 0.20 | 226 | 0.20 | 227 |
| | 30 min | 0.16 | 237 | 0.20 | 225 | 0.20 | 226 |
| | 90 min | 0.16 | 237 | 0.20 | 225 | 0.21 | 224 |
| | 6 h | 0.17 | 237 | 0.20 | 226 | 0.21 | 226 |
| | 24 h | 0.17 | 236 | 0.20 | 226 | 0.20 | 225 |

Table 6: Earth IR and Albedo, $2\sigma$ Values, Hot Case (Gilmore, 2002)

| | | Inclination (deg) | | | | | |
|---|---|---|---|---|---|---|---|
| | | 0-30 | | 30-60 | | 60-90 | |
| Surface Sensitivity | Time Period | Albedo | IR ($W/m^2$) | Albedo | IR ($W/m^2$) | Albedo | IR ($W/m^2$) |
| Albedo | 16 sec | 0.29 | 205 | 0.36 | 201 | 0.38 | 197 |
| | 128 sec | 0.29 | 211 | 0.35 | 202 | 0.37 | 199 |
| | 896 sec | 0.26 | 225 | 0.29 | 213 | 0.28 | 213 |
| | 30 min | 0.24 | 234 | 0.27 | 223 | 0.26 | 223 |
| | 90 min | 0.22 | 246 | 0.26 | 229 | 0.24 | 219 |
| | 6 h | 0.20 | 252 | 0.25 | 231 | 0.23 | 224 |
| | 24 h | 0.20 | 252 | 0.25 | 232 | 0.23 | 224 |
| IR | 16 sec | 0.17 | 285 | 0.17 | 280 | 0.17 | 280 |
| | 128 sec | 0.17 | 284 | 0.17 | 279 | 0.17 | 279 |
| | 896 sec | 0.18 | 279 | 0.18 | 264 | 0.18 | 263 |
| | 30 min | 0.18 | 274 | 0.20 | 258 | 0.20 | 258 |
| | 90 min | 0.19 | 268 | 0.21 | 254 | 0.21 | 242 |
| | 6 h | 0.19 | 261 | 0.21 | 242 | 0.21 | 216 |
| | 24 h | 0.18 | 258 | 0.21 | 241 | 0.21 | 215 |
| Both | 16 sec | 0.21 | 260 | 0.23 | 240 | 0.24 | 237 |
| | 128 sec | 0.21 | 260 | 0.23 | 240 | 0.24 | 238 |
| | 896 sec | 0.21 | 261 | 0.23 | 241 | 0.24 | 240 |
| | 30 min | 0.21 | 258 | 0.23 | 240 | 0.23 | 242 |
| | 90 min | 0.20 | 258 | 0.23 | 241 | 0.23 | 232 |
| | 6 h | 0.19 | 255 | 0.23 | 242 | 0.22 | 230 |
| | 24 h | 0.19 | 257 | 0.23 | 241 | 0.23 | 230 |

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