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14. ABSTRACT
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To provide a baseline for the TCS design and to help bound the problem for the development of robust and modular thermal designs, a bus sizing exercise was conducted to determine an upper and lower bound for the internal heat load of the system. This exercise also provided mass and volume estimates. In addition, the range of external heat loads for small satellites in low earth orbit were evaluated. From this analysis, the worst hot and cold cases conditions were identified. Using these two cases, various design parameters were evaluated, two different design approaches were compared, and the feasibility of a one-size-fits-all approach is assessed. Finally, critical design parameters are identified.

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ISSUES AND IMPLICATIONS OF THE THERMAL CONTROL SYSTEM ON THE "SIX DAY SPACECRAFT"

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ABSTRACT

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INTRODUCTION

The traditional approach to satellite design is a customized and highly optimized satellite bus. The primary driver is to minimize mass but often at the expense of time and money. As a result the time from concept development to satellite deployment is three to ten years. In addition, each system costs tens of millions to billions of dollars. As a result, space assets are considered strategic in nature. Therefore, the Department of Defense (DOD) is looking for ways to drastically reduce the development time and cost of satellites to extend the benefits of space from the strategic commanders to the battlefield warfighters. This is the primary goal of the Operationally Responsive Space (ORS) program.

To meet the goals of ORS, the satellite must be adaptable to different missions, changing threats, and emerging technologies. The ultimate goal is to have an asset launched and operational within six days after call up from the battlefield commander: hence the "six day satellite."

One system in particular poses a great challenge to meeting the goals of ORS: the satellite's Thermal Control System (TCS). Traditionally, every aspect of the satellite, the mission, and the components must be known before the thermal design can be completed. The overall goal of the engineer is to reduce the mass of the system by trading cost and engineering time. As a result, every design is unique and requires extensive design, modeling, analysis, and test programs, which is one of the contributing factors to the time and expense required for satellite development. To meet the goals of responsive space, future satellite systems will have to be robust, modular, and adaptable.

THERMAL CONTROL DESIGN

One approach to achieve the goals of ORS in the near term is to separate the design and engineering of the payload from the bus. The bus would provide a specific range of baseline capabilities to meet the needs of most missions and payloads. The goal is an 80% design solution. Any additional capability required by the payload would have to be provided by the payload itself. Integration of the bus and payload would occur through standard mechanical, electrical, thermal, and software interfaces. It should be noted that, according to this philosophy, there will be some payloads that can not be economically accommodated by the standard bus, and a unique system will have to be designed.

This philosophy is not new and has analogies in the computer and automotive industry. For example, a supplier, such as Dell, has a standard model that will meet the needs of the majority of the market. For those users that need additional features, such as a faster processor or better graphics the appropriate upgrades are made to the standard model before the unit is shipped to the customer. For the user that requires a top of the line system, often times a custom built unit provides the only economic solution.

In the past, there has been a number of standard bus development programs focused at reducing development time and cost. The disadvantage with most standardized bus programs is that the bus quickly becomes obsolete and must be completely redesigned as new technologies are introduced. One of the goals of the ORS program is the development of technologies that provide robust and flexible bus designs similar to those in other industries. For example, plug-and-play USB connectivity similar to the PC based version is being investigated for the command and data handling system[1]. Plug-and-play addresses the software and electrical interfaces, but other efforts are needed to address the mechanical and thermal interfaces

Regardless of the design philosophy, a certain level of fidelity of the bus design is needed before the basic requirements for the TCS can be identified. One hindrance is that the missions, payloads, and requirements for ORS are still somewhat nebulous. As a result, bus architectures and specific components have not been identified, which makes it difficult to derive even initial thermal system requirements. However, there is one preliminary assumption about the bus that can be made. Because of launch vehicle limitations, ORS missions will likely be relegated to 450 kg class satellites. Using this basic assumption, the capabilities that a small satellite bus can provide can be determined.

INTERNAL HEAT LOAD

To evaluate the internal heat load that the TCS must be able to accommodate, a bus sizing exercise was conducted. The purpose of the exercise was not to specify exact components for the bus but to identify the design space for the system based on current and near term technologies. For each subsystem, two levels of capability were identified. Similar to the Dell analogy, subsystem components were sized to provide a baseline capability and an upgraded one. From these components, the mass, volume, and power of the subsystem were estimated. The results were an upper and lower bounds for the design of the TCS and are only summarized here. A more detailed analysis can be found in Reference 2.

Low Capability Bus

The low capability bus (LCB) represents a minimum level of capability that is required for small satellite missions. It is important to note that these system requirements do not represent any particular mission or system. Instead, they are a first order approximation based on general mission needs and were used to begin subsystem design and analysis. The capability for each subsystem is summarized below.

Attitude Determination and Control (ADC)
- Attitude knowledge of 0.1° - 1°
- Pointing accuracy of 1° - 5°
- Slew rate of 0.05 - 0.1 °/s

Telemetry, Tracking, and Command (TTC)
- S-band system
- Data rate of 1 Mbps

Navigation and Guidance (NG)
- 12 channel GPS receiver

Command and Data Handling (CDH)
- Space Plug-and-play Avionics–USB (SPA-U) based system
- Legacy system compatibility
- Power management for USB based components

Power Management (PM)
- 500 W system
- Triple junction deployed solar array
- Lithium-ion batteries
- Peak power tracking (PPT)

Structure
- Aluminum honeycomb

Propulsion
- No onboard propulsion system

There are a few important points to note. First, the control scheme for the ADC system is based on a momentum bias system with magnetic torque rods providing additional control. Second, a PPT control system is used for power regulation. The advantage of a PPT system is that it acts like a low impedance power supply making design integration a simple task. Finally, because of the short mission life, an onboard propulsion system was not included in the system sizing. It is assumed that the orbit altitude will be high enough to meet mission requirements without additional station keeping.

Using these requirements, components were selected for each subsystem. The resulting mass, power, and volume requirements are summarized on Table 1 below.

Table 1: Low capability bus

Subsystem	Mass [kg]	Power [W]	Size [cm]
ADC	10.3	18.5	30 x 24 x 12
TTC	2.8	7.4	9.8 x 9.6 x 7.2
NG	0.02	0.8	7.0 x 4.5 x 1.0
CDH	15.2	50	34 x 25 x 20
PM	18.3	70.3	25 x 23 x 21
Structure	21.5	n/a	27 x 40.5 x 71
Propulsion	0	0	0 x 0 x 0
	68.1	**147.0**	**27 x 40.5 x 71**

High Capability Bus

Opposed to the LCB, the high capability bus (HCB) does not represent the maximum capability that is required for small satellite missions. It is merely a more capable bus that is more representative of an ~80% design solution. For ORS, the goal is not a 100% design solution for all scenarios. The capability for each subsystem is summarized below.

Attitude Determination and Control (ADC)

- Attitude knowledge of 0.02° - 0.1°
- Pointing accuracy of 0.05° - 1°
- Slew rate of 0.1 - 0.3 °/s

Telemetry, Tracking, and Command (TTC)

- S-band system for housing keeping
- Ku-band CDL system
- Data rate of 274 Mbps

Navigation and Guidance (NG)

- 12 channel GPS receiver

Command and Data Handling (CDH)

- SPA-U based system
- Legacy system compatibility
- Power management for USB based components

Power Management (PM)

- 1500 W system
- Triple junction deployed solar array
- Lithium-ion batteries
- Peak power tracking

Structure

- Aluminum honeycomb

Propulsion

- No onboard propulsion system

Again using these requirements, components were selected for each subsystem. The resulting mass, power, and volume requirements for the high capability system are summarized on Table 2.

The total power loads summarized on Tables 1 and 2 represent the maximum heat load for the system. Because the majority of the subsystems consist of components that are subject to electrical heat losses rather than mechanical devices, nearly 90% of the power generated by the satellite must be radiated to space by the TCS. Most thermal engineers assume 100% power dissipation for the hot case to provide additional margin to the design. As for the cold case, the satellite never completely shuts down so the internal heat load is always greater than 0W. The actual value is dependent on the satellite and the mission, but in general the lowest value that can be expected is 50 W.

Table 2: High capability bus

Subsystem	Mass [kg]	Power [W]	Size [cm]
ADC	23.3	49.5	35 x 35 x 22
TTC	10.6	64.4	25 x 25 x 15
NG	0.0	0.8	7.0 x 4.5 x 1.0
CDH	15.2	50	34 x 25 x 20
PM	54.6	253	72 x 23 x 21
Structure	38.6	n/a	52 x 40.5 x 71
Propulsion	0	0	0 x 0 x 0
	142.3	**417.7**	**52 x 40.5 x 71**

EXTERNAL ENVIRONMENT

For most spacecraft, the thermal environment and the external heat loads are determined from the specific orbit for the mission, the orientation of the spacecraft, the surface properties, and the size of the system. From these, the absolute worst hot and cold case conditions are determined. Unfortunately, none of these parameters are clearly defined for ORS missions. Since specific orbits are largely unknown for ORS, the TCS must be adaptable to all low earth orbits. The only constraining assumption that can be made is that the orbit regime is limited to low Earth orbits. For simplicity, only circular orbits were evaluated.

Using these constraints, the worst hot case condition is shown on Fig. 1 and is defined below[3,4].

- Orbit beta angle is 90°.
- Eclipse duration is zero.

- The panel with the largest surface area is always nadir facing.
- The panel with the second largest surface area always faces the sun.
- Solar flux is 1414 W/m^2.
- Earth IR is 275 W/m^2.
- Albedo coefficient is 0.57.
- The side reserved for the payload faces space. That side does not radiate heat to space.

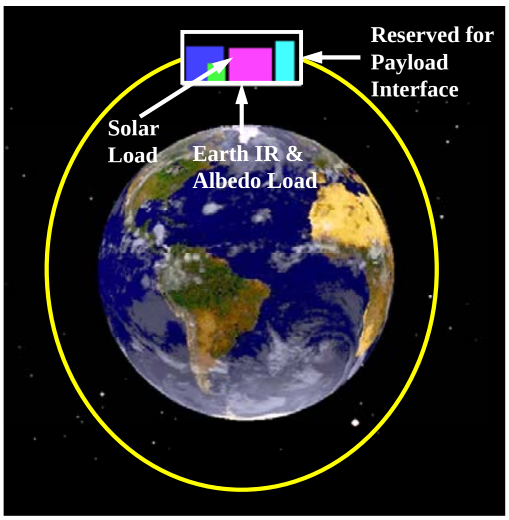

Figure 1: Worst hot case orientation

The worst cold case condition is shown on Fig. 2 and is defined below [3,4].

- Orbit beta angle is 0°.
- Eclipse duration is 43%.
- The panel with the smallest surface area is always anti-nadir.
- Solar flux is 1322 W/m^2.
- The side reserved for the payload is nadir pointing, so there is not an Earth IR or Albedo heat load.

It is important to note that these are theoretical worst case conditions. For example, it is unlikely that both the Albedo and Earth IR maximum heat loads will occur at the same time. The Albedo heat load increases with orbit inclination; whereas, the Earth IR heat load increases with decreasing orbit inclination. The theoretical worst case scenarios were chosen to provide confidence in the design and to add a significant amount of margin for most orbits.

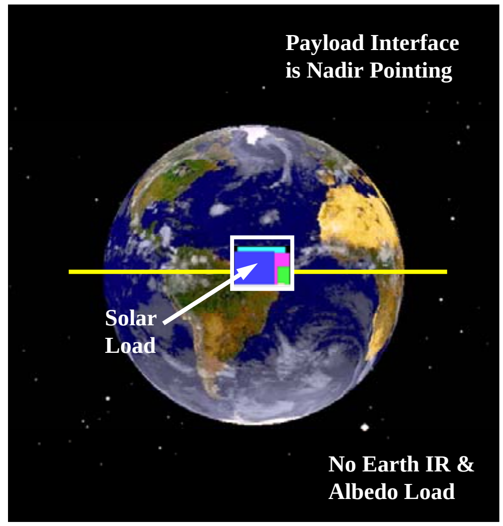

Figure 2: Worst cold case orientation

Finally, it is important to address the transients of the low Earth orbit (LEO) environment. Because of the low altitudes and short orbital periods, the LEO environment is dynamic and creates special difficulties for the thermal engineer. A LEO spacecraft only sees a small portion of the Earth. As it orbits, it is exposed to rapidly changing environmental conditions as it passes over various geographical features and local time zones, which significantly affect Earth IR and albedo heat loads. In addition, eclipse times can vary from nearly half of the orbital period to zero. As a result, the thermal capacitance of the system is important, especially for lightweight components.

The focus of this effort is on the core bus structure and not external component such as solar arrays or antennas. For this reason, orbit averaged values were used because of the large thermal capacitance associated with the bus. To validate this assumption, a first order transient analysis was conducted.

The transient behavior for a radiation-conduction system is determined using the following equation

$$t_{r-c} = \frac{mc}{3\varepsilon\sigma A_{rad}}\left(\frac{1}{T^3} - \frac{1}{T_i^3}\right) \qquad (1)$$

where m is the mass of the system [kg], c is the specific heat [J/kg-K], and T_i is the initial temperature[5]. The equation assumes the temperature of the surroundings is 0 K, which is valid for a first order approximation. Using 875 J/kg-K for the specific heat of aluminum and the modified density of the bus (total mass divided by total volume), the time for the temperature of the low capability bus to change from 303K to 273K is 81 minutes. For the high capability bus, the change occurs in 120 minutes. For a satellite in LEO at an altitude of 200 km and a β angle of 0°, the eclipse time is only 36 minutes. Therefore, orbit averaged values for the external heat loads are acceptable.

TEMPERATURE LIMITS

Before evaluation of the system can be initiated, there is one remaining topic that must be addressed. The fundamental purpose of the TCS is to maintain components within their acceptable operating and survival temperature limits. These limits are wide ranging and component dependent. To ensure that all of the components are within their operating temperature limits the components with the tightest temperature range were used to define the temperature requirements of the system. For this case, they were the momentum wheel and Lithium ion batteries, which have an operating temperature range from 263 to 313 K.

Because of uncertainties within the TCS, it is common practice to add margin to the temperature constraints. A study of a number of military programs concluded that an 11 K margin was required to provide 95% confidence that flight temperatures would be within limits for a model correlated to thermal balance test data[6]. For uncorrelated models, the uncertainty jumps to 17 K. An informal survey of NASA and commercial satellite programs showed that 5K was the most common margin used[4]. Because the external and internal heat load values chosen for the hot and cold case analyses are already conservative, a 10 K margin will be used even though the model will be uncorrelated.

ENERGY BALANCE

Essentially, the primary task of the thermal engineer is to balance the thermal energy of the satellite to ensure all of the internal components remain within their acceptable temperature limits during the worst hot and cold cases. External and internal heat generation must be properly balanced with the excess heat radiated to space. Figure 3 summarizes the sources and sinks for a satellite in low earth orbit. A simple energy balance analysis between the satellite and the space environment can be used to determine whether or not the satellite has enough surface area to maintain its temperature within acceptable limits for the hot case. In addition, it can be used to size survival heater power to maintain the temperature within acceptable limits for the cold case.

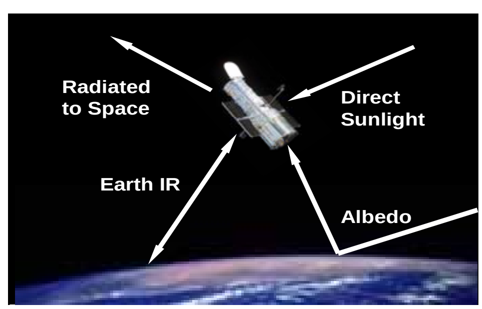

Figure 3: Heat loads for a LEO satellite

The actual temperature of space is 4 K; however, as a first order approximation the temperature of space can be assumed to be 0K. Substituting in expressions for the heat loads, the energy balance equation is[7]:

$$\varepsilon\sigma A_{rad} T_s^4 = \varepsilon A_s F_{s,e} I_{EIR} + \alpha A_{\perp} I_{sun} + \alpha a A_s F_{s,se} I_{sun} + Q_{Internal} \qquad (2)$$

where ε is the emissivity of the spacecraft, σ is the Stefan-Boltzmann constant [W/m^2-K^4], A_{rad} is the radiator surface area [m], T_s is the average temperature of the spacecraft [K], A_s is the surface area [m], $F_{s,e}$ is the view factor between the spacecraft and the Earth, I_{EIR} is the intensity of the Earth IR, α is the surface solar absorptivity, $A_{\perp}$ is the area perpendicular to the sun [m], and I_{sun} is the solar heat flux [W/m^2], a is the Earth albedo coefficient, $F_{s,se}$ is the view factor between the spacecraft and the sunlit Earth, and $Q_{Internal}$ is the internal heat generation [W]. This equation provides a first order approximation of the radiator area need for the hot case and the heater power needed for the cold case.

Energy Balance for the Low Capability Bus

By rearranging Eq. 2 and solving for A_{rad}, the radiator surface area required to keep the satellite below the maximum operating temperature during the hot case condition can be calculated. The cold case temperature is also determined using Eq. 2 by solving for T_s. If the temperature exceeds the lower temperature limit, survival heaters must be used to provide additional heat. Using Eq. 2 to determine the radiator area and the survival heater power for the satellite provides a first order approximation to size the TCS. It also provides a tool to quickly eliminate thermal control schemes and hardware that will not be applicable to the problem.

For the first order approximation of the energy balance, the internal heat load for the hot case, which was summarized on Table 1, is 147.0 W. As for the cold case, the internal heat load was assumed to be 50 W. Next, it was assumed that the surface was painted white, and only five surfaces were available for radiation to space. The remaining surface was reserved as the interface surface for the payload. An emissivity of 0.88 and an absorptivity of 0.22 were used for white paint. The inputs into the energy balance equation are summarized below.

Table 3: Summary of the inputs for the energy balance equation for the LCB

	Hot Case	Cold Case
Eclipse Percent	0	0.43
Solar Constant [W/m^2]	1414	1322
Albedo Coefficient	0.57	0.18
Earth IR [W/m^2]	275	218
Internal Heat [W]	147	50
Temperature Limit [K]	303	273
Area ⊥ to Sun [m^2]	0.192	0.109
Area ⊥ to Earth [m^2]	0.288	0

Using the energy balance equation and the parameters above, the radiator area required to keep the bus below 303 K was 0.76 m^2 and the resulting cold case temperature was 204.6 K. The total available radiator area of the bus was 1.07 m^2. If the surface area was increased to the total available radiator area, the hot case temperature was reduced to 278.3 K, and the cold case temperature was reduced to 187.9 K. To increase the cold case temperature to acceptable levels, a survival heater power of 240 W would be required. A passive thermal control system incorporating survival heaters would satisfy the thermal requirements. However, an active system might be needed because of the large survival heater power requirement.

Energy Balance for the High Capability Bus

For the high capability bus, the internal heat loads for the hot case and cold case were 417.7 W and 50 W, respectively. Again, it was assumed that only five sides of the satellite were available for radiation to space, the surface finish was white paint, and the temperature limits remain unchanged. All of the input values are summarized on Table 4. Following the same process as before, the radiator area required to keep the bus below 303 K was 1.59 m^2; however, the available radiator area was only 1.52 m^2. The result was a hot case temperature of 306.5 K. The cold case temperature was 183.0 K, and the survival heat power was 360 W. Because the system was already above the maximum temperature limit, supplemental radiator area was required.

Table 4: Summary of the inputs for the energy balance equation for the HCB

	Hot Case	Cold Case
Eclipse Percent	0	0.43
Solar Constant [W/m^2]	1414	1322
Albedo Coefficient	0.57	0.18
Earth IR [W/m^2]	275	218
Internal Heat [W]	417.7	50
Temperature Limit [K]	303	273
Area $\perp$ to Sun [m^2]	0.287	0.211
Area $\perp$ to Earth [m^2]	0.392	0

THERMAL CONTROL SYSTEM ARCHITECTURE

Once the internal and external heat loads for both the worst hot and cold cases were determined, the focus of the effort turned to the thermal control system architecture. Since the focus of ORS is to deploy a space-craft within six days of call-up, the primary design drivers of the system are modularity and ease of integration. To achieve the goals of the "six day satellite," the satellite or the subsystem components will have to be on hand for rapid integration and launch; however, their state of pre-integration is still open for debate.

There are three primary options. The first is the more flexible option in which the comp-onents are on hand so that the satellite can be quickly assembled to meet the needs of the mission. The second and faster option would be to have the satellite preassembled so that it is ready for integration to the launch vehicle. The final option is a comb-ination of the two where the subsystems are preassembled and then integrated into the satellite structure based on mission needs. Because this option provides both flexibility and speed to some degree, it was used as the integration strategy. The actual layouts for the LCB and HCB are shown on the figure below.

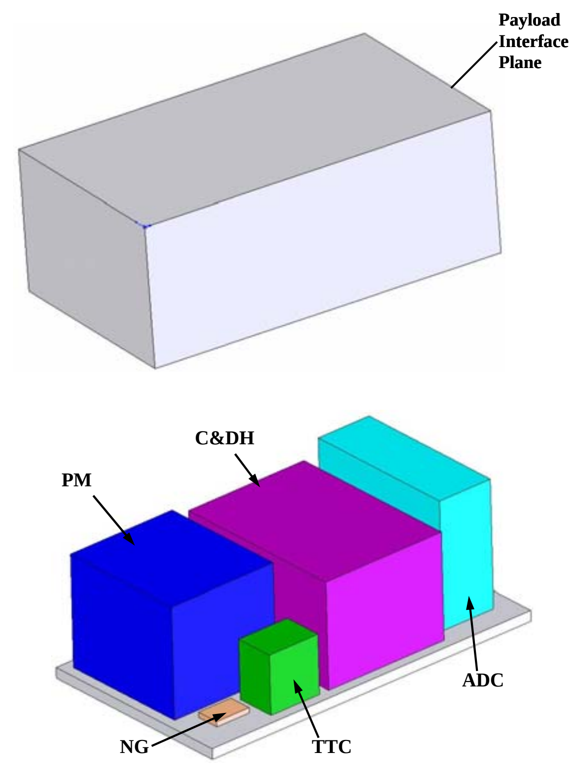

Figure 4: Layout of the LCB

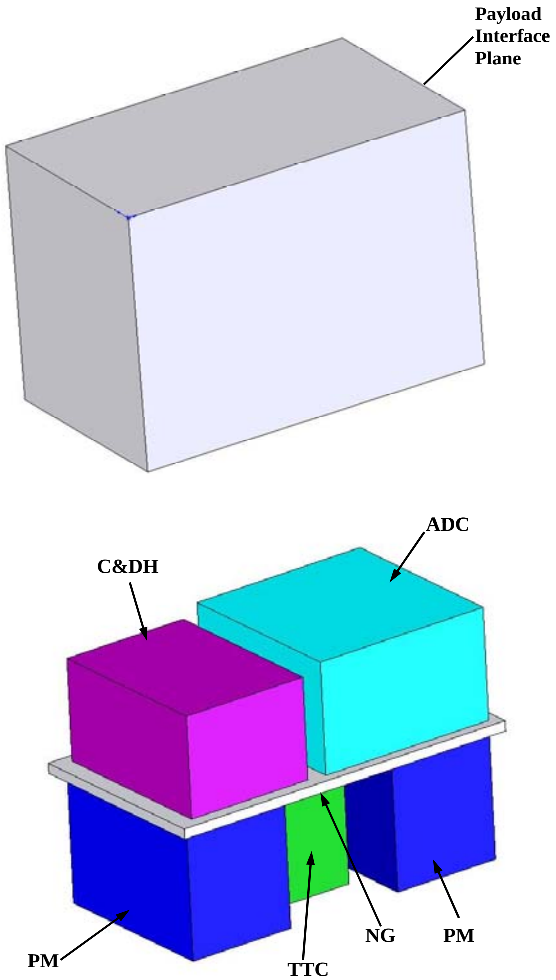

Figure 5: Layout of the HCB

Because the subsystems were housed in separate enclosures, the design of the TCS was split into two parts. The first part, which focused of the design of the overall bus TCS, emphasized the conduction of heat from the subsystems through the bus structure to the exterior of the satellite where it can be radiated to space. Initially, the subsystems were modeled as simple aluminum enclosures with uniform heat loads. After the heat conduction path through the bus was designed, the focus turned to the subsystems, which was the focus of the second part of the design. Finally, the bus model and the subsystem models were integrated, and the final design was analyzed.

Initially three basic TCS architectures were investigated. The first was an isothermal architecture where panels with high thermal conductivity inserts were used to spread heat across the entire satellite. The second was a thermal isolative approach where each subsystem was isolated from one another and mounted to a dedicated radiator area. The final architecture consisted of a variable heat transfer rate, which can be achieved with either a passive heat switch or an active system approach.

<u>Isothermal Architecture</u>

To achieve isothermal conditions, the design incorporated a honeycomb electronics shelf with an APG core to improve the lateral conductivity of the panel. The design was based on k-Technologies' k-Core concept, which uses encapsulated Annealed Pyrolytic Graphite (APG) to spread heat across the panel. The lateral conductivity of APG is on the order of 1700 W/m-K. A schematic of the k-Core concept is shown in below.

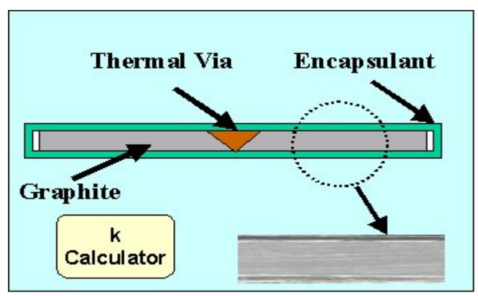

Figure 6: k-Technologies patented k-Core material system[8]

The bus structure and the subsystem enclosures were modeled in Thermal Desktop (TD). The subsystems were modeled as Al-2024 enclosures with wall thicknesses of 1.5875 mm and a thermal conductivity of 185 W/m-K. The edges between the different sides of the enclosures were assumed to be in perfect contact, which is the equivalent of a continuous material around the corners. The conductivity of the interface between each of the subsystem enclosures and the shelf was controlled using surface contact conductors. Conservatively, a joint conductivity of 110 W/m-K was assumed for bare aluminum interfaces[4]. As for the electronics shelf, it was modeled as an aluminum honeycomb panel with a 1mm APG core encapsulated in the face sheets. As a result, the face sheets for the electronics shelf were 2.6 times thicker than the face sheets used for the other panels. The interface conductivity between the face sheets and the core was controlled using surface contact conductors. The other panels were modeled as two face sheets with a contactor to control the conductivity of the honeycomb core. Initially, a honeycomb core transverse conductance of 250 W/m^2-K was used. The panels were also assumed to be in perfect contact with one another.

As for the boundary conditions, the internal heat loads for each subsystem were evenly distributed over all six surfaces of the enclosure. The external loads were applied using surface heat loads. The solar loads for the hot and cold cases were 312 W/m^2 and 166 W/m^2, respectively. The combined Earth IR and A+lbedo load for the hot case was 414 W/m^2. White paint with an ε of 0.88 and an α of 0.22 was used for the exterior of the satellite. RadCAD was used to calculate the radiation exchange factors with space. Radiation within the bus was included in the calculations. The interior surfaces were painted black to enhance radiative heat transfer.

Because of the low internal heat density of the LCB, the design was fairly simple. The ADC, TTC, and NG subsystems could be maintained within proper temperatures with a bare aluminum-aluminum interface between the enclosure and the electronics shelf. As for the CDH and PM subsystems, an RTV insert was required to increase the contact conduction at the interface. The design of the HCB was more complicated in that three deployable radiators had to be added to achieve proper cooling. The radiator locations are shown on Figure 7 and are 0.35 m by 0.405 m. An adequate thermal design could be achieved for the HCB if the solar and the combined Earth IR and Albedo loads were eliminated through the use of Multi-Layer Insulation (MLI). The use of MLI was not considered practical for ORS operations because of its complicated fabrication process, high touch labor, and fragility. Also, since orbits, missions, orientations, and components are unknown, its pre-application to the structural panels is impractical.

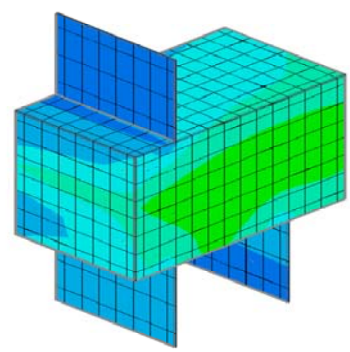

Figure 7: Deployable radiator locations

As for the cold case, the temperatures of all of the components were well below the minimum temperature limit without supplemental survival heater power. To maintain the bus within the baseline temperature limit, an additional heater power of 150 W was needed for the LCB, which was higher than the total power consumption for the hot case. The HCB required 165 W of survival heater power.

The high survival heater power requirements are a result of the drastic change in the external load between the hot and cold cases. The same problem was reported by Barton, where survival heater power was 63% higher than the component operating at full load[9]. As for the worst hot and cold case conditions defined here, it is important to note that it is impossible for both cases to exist for the same orbit. For a more realistic analysis, the worst hot and cold cases were separated by orbit and are outlined below. For each different orbit, the surface properties were tailored and then the heaters were sized.

1a. Worst Case for Hot Orbit: Same as before; results are unchanged
1b. Cold Case for Hot Orbit: Beta angle of 90°, minimum power output, and an orientation with the payload facing the Earth and the smallest adjacent side receiving the solar load
2a. Hot Case for Cold Orbit: Beta angle of 0°, maximum power, and an orientation with largest panels exposed to the solar, Earth IR, and albedo loads
2b. Worst Case for Cold Orbit: Same as before

For case number one for the LCB, the satellite exterior was painted white, and the survival heater power required was reduced to 30 W. For case number two, the exterior was painted green, which increased the solar absorptivity to 0.57. The emissivity was unchanged. The survival heater power needed to maintain the minimum temperature was reduced to 40 W. As for the HCB, the survival power requirements were reduced to 115 and 90 W, respectively.

<u>Thermally Isolative Architecture</u>

To achieve the thermally isolative design, each subsystem was mounted to a different panel. Each panel was then isolated from others with a felt insert at the interface. In this design, the location of the subsystems and the orientation of the satellite play an important role in the design of the TCS, this makes this architecture more difficult to implement. However, since the sub-system properties will not be known ahead of time, the majority of the engineering can be performed ahead of time and a fairly simple design analysis and optimization software tool can be completed once the mission is known.

The results for the thermal isolative architecture were similar to the results for the isothermal architecture in that a large survival heater power level was needed to maintain cold case temperatures. Because the system and component location could be optimized somewhat with this architecture, the heater power level was 25% less. However, this value is still significant compared to the overall power of the buses. This architecture provided an improved thermal performance but will be more difficult to implement under ORS operations.

<u>Variable Heat Transfer Architecture</u>

This type of architecture can be achieved either with a passive thermal switch, an active convection based system, or a variable emissivity radiator. The key is to be able to change the heat transfer rate between the hot and cold cases. The ideal method would be to implement passive conduction based thermal switches. However, thermal switches have not achieved the reliability necessary for space missions. Instead of basing the analysis on a single technology, a more general analysis was conducted to determine the switching requirements for such architectures.

In the variable heat transfer system architecture, the critical design parameter is the heat transfer from the subsystem through the base plate during the cold case in order to maximize the temperature rise across the system, which minimizes the survival heater power requirement. The temperature rise through the interface between the base plate and the enclosure is calculated with the following equation.

$$T_H - T_C = \Delta T = \frac{Q}{AK_J} \qquad (3)$$

where T_H is the temperature on the hot side of the joint [K], T_C is the temperature on the cold side of the joint [K], Q is the heat load [W], A is the contact area [m^2], and K_J is the joint conductivity [W/m^2-K]. Since the two interfaces and the interstitial material are in series, their thermal resistances are added. This is analogous to electrical resistances and the same rules apply. Figure 8 provides a schematic for clarity.

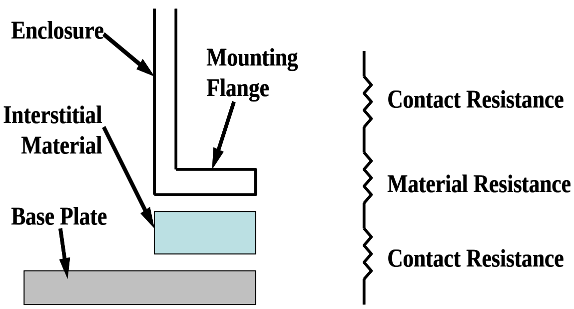

Figure 8: Schematic of the thermal joint between the enclosure and the base plate

To be consistent with the electrical resistance analogy, Eq. 3 is modified to the following form:

$$\Delta T = QR \tag{4}$$

where R is the thermal resistance [K/W]. The total resistance for the joint is:

$$R_{Tot} = \frac{1}{A}\left(\frac{2}{K_{\text{int}}} + \frac{L}{K}\right) \tag{5}$$

The joint conductivity, K_J, is the inverse of the total joint resistance divided by the contact area.

Since the temperature on the hot side of the interface is dependent on the system parameters i.e. the contact area, the internal power dissipation, and the cold side temperature, it is difficult to identify a single joint conductivity that would meet the thermal needs for all potential components and subsystems. A very small joint conductivity on the order of 1 W/m^2-K would probably meet the needs of the majority, but it might not be possible to design a thermal joint with that small of a thermal conductivity. To better gauge the need, the LCB and HCB designs were evaluated. For the LCB, the cold case temperature from the energy balance was 187.9 K. For the HCB, it was 183.0 K. Using the cold case power consumptions and the contact areas for each enclosure, based on the thermal joint above, the joint thermal conductivity required to keep the subsystem temperatures above the lower temperature limit was calculated. The results are presented on Table 5.

Table 5: Joint conductivity required to meet the minimum temperature limit

System	Heat Load [W]	Surface Area [m^2]	Power Density [W/m^2]	K_J [W/m^2-K]
LCB				
ADC	18.5	0.0168	1101.19	12.80
CDH	13.0	0.0236	550.85	6.41
PM	16.2	0.0184	880.43	10.24
TTC	7.4	0.0067	1101.19	12.80
HCB				
ADC	18.5	0.0228	811.40	9.43
CDH	13	0.0236	550.85	6.41
PM	41.2	0.0372	1107.53	12.88
TTC	7.4	0.016	462.50	5.38

To meet the needs of all of the subsystems on the table, a joint conductivity of 5W/m^2K is required; however, this does not take into account the temperature rise from the enclosure to the component and a joint conductivity on the order 10W/m^2-K will probably be acceptable. The design or description of such a joint is beyond the scope of this effort.

For architectures based on thermal switches, the performance of the system is based on the conductance ratio of the system. If the conductance ratio is high enough, then survival heaters can virtually be eliminated. The result would be a very robust system. Conductance ratios on the order of 20:1 to 70:1 are needed for a robust operational system.

There is one disadvantage to this system architecture. The first is that the switching component typically adds an additional thermal interface to the system. For radiator panels that are already operating at their limit, adding the additional interface will cause the components to exceed their operating temperatures during the hot case. As a result, radiator area has to be oversized to ensure proper operation, which will add some mass to the system. However, the advantage of a modular, robust system outweighs the disadvantages when a short turn-around-time becomes more important than mass.

CONCLUSIONS

This effort was an initial investigation into the system architectures that are best suited for ORS. In that process, preliminary thermal requirements were identified base on the range of satellite capabilities presented. From there, three different system architectures were evaluated. Of the three, the variable heat transfer architecture was best suited for ORS because of the wide range of components, missions, and orbits envisioned for ORS. This architecture provided the most robust solution. However, with advances in specific technologies, the other two architectures would also be suitable for ORS missions.

As for a one-size-fits-all system, a design solution is possible based on the variable heat transfer architecture. However, its success is dependent on developing either passive thermal switches or lightweight, low power active systems suitable for small satellites.

As for technology development, there are two other critical technologies that need to be developed. The first technologies that need to be developed are variable emissivity and reconfigurable radiator concepts. Varying the amount of energy radiated to space at any given time enables the isothermal architecture. The bus would be completely insulated with a high conductivity connection to the radiator. The thermal balance of the system would then be controlled at the radiator.

The second is a low cost and robust insulation system that can be rapidly fabricated to replace the MLI currently used on satellite systems. In addition, reconfigurable insulation concepts are of interest. Eliminating external loads from the design equation significantly simplifies the thermal design process. A break through in this area enables the thermally isolative architecture.

The thermal control system poses significant challenges to the goals of ORS. Highly optimized systems will not be feasible on the short time scale dictated for tactical satellites. Instead modular, robust, adaptable systems are required. To meet these challenges, two areas of development are critical. The first is system architecture and design tools. The second is the technologies capable of meeting the requirements dictated by the system architectures.

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