Thermal Analysis of Iodine Satellite (iSAT) from Preliminary Design Review (PDR) to Critical Design Review (CDR)

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The Iodine Satellite (iSAT) is a 12U cubesat with a primary mission to demonstrate the iodine fueled Hall Effect Thruster (HET) propulsion system. The spacecraft (SC) will operate throughout a one year mission in an effort to mature the propulsion system for use in future applications. The benefit of the HET is that it uses a propellant, iodine, which is easy to store and provides a high thrust-to-mass ratio. This paper will describe the thermal analysis and design of the SC between Preliminary Design Review (PDR) and Critical Design Review (CDR). The design of the satellite has undergone many changes due to a variety of challenges, both before PDR and during the time period discussed in this paper. Thermal challenges associated with the system include a high power density, small amounts of available radiative surface area, localized temperature requirements of the propulsion components, and unknown orbital parameters. The thermal control system is implemented to maintain component temperatures within their respective operational limits throughout the mission, while also maintaining propulsion components at the high temperatures needed to allow gaseous iodine propellant to flow. The design includes heaters, insulation, radiators, coatings, and thermal straps. Currently, the maximum temperatures for several components are near to their maximum operation limit, and the battery is close to its minimum operation limit. Mitigation strategies and planned work to solve these challenges will be discussed.

Nomenclature

Al	=	Aluminum
BOL	=	Beginning of Life
°C	=	Degrees Celcius
CAD	=	Computer Aided Design
C&DH	=	Command and Data Handling
DAC	=	Design and Analysis Cycle
DCE	=	Drive Control Electronics
EIB	=	Extra Interfaces Board
EOL	=	End of Life
FC	=	Flight Computer
FEM	=	Finite Element Mesher
FL	=	Fuel Lines
FLUINT	=	Fluid Integrator
GEVS	=	General Environmental Verification Standard
GN&C	=	Guidance, Navigation, and Control
GPS	=	Global Positioning System
HET	=	Hall Effect Thruster
IMU	=	Inertial Measurement Unit
in	=	inches
IR	=	Infrared
iSAT	=	Iodine Satellite

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kg	=	kilogram
LVLH	=	Local Vertical Local Horizon
LWIR	=	Long Wavelength Infrared
m	=	Meter
MIL	=	Military Standard
min	=	minute
MLI	=	Multilayer Insulation
mm	=	millimeter
MT	=	Magnetorquer
MW	=	Momentum Wheels
OSR	=	Optical Solar Radiator
PCB	=	Printed Circuit Board
PDB	=	Power Distribution List
PDR	=	Preliminary Design Review
PFCV	=	Proportional Flow Control Valve
PMB	=	Power Management Board
PPU	=	Power Processing Unit
PT	=	Pressure Transducer
RAAN	=	Right Ascension of the Ascending Node
RADk	=	Radiation Conductor
s	=	second
SC	=	Spacecraft
SINDA	=	Systems Improved Numerical Differencing Analyzer
SS	=	Sun Sensor
ST	=	Star Tracker
TD	=	Thermal Desktop
Temp	=	Temperature
U	=	Unit
W	=	Watt

I. Introduction

The iSAT model discussed in this paper is in the midst of a design analysis cycle (DAC) between Preliminary Design Review (PDR) and Critical Design Review (CDR). Some details of the design and concept of operations remain in flux. This current design has gone through many changes stemming from both iterations between the design and analysis team, and comments/ requests for actions during reviews. Previous designs, not discussed in detail in this paper, can be found in Appendix A.

Included in the thermal model are all powered components, the chassis, and solar panel assemblies. Most secondary fixtures are not explicitly modeled. The only component mounts modeled are the camera and cathode mounts. The mounting interfaces for inner components and hinges between the chassis and solar panel backing are simplified as contactors with no geometry modeled to represent the mass. Component dimensions and locations were measured from the Computer Aided Design (CAD) model and built in the thermal model. Four of the six chassis wall panels were built in a finite element mesher (FEM) and imported into the thermal model to accommodate cutouts or non-rectangular shapes, but all were measured from the same CAD model. Figure 1 shows the current Thermal Desktop (TD) model of the entire satellite, with top and bottom, and SC coordinate system indicated. Components within the enclosure are not visible from this view. The thruster shown is the iodine Hall Effect Thruster, which is comprised of an anode and a cathode. The anode is within the thruster housing, simplified in the TD model as a single solid cylinder, and the cathode is modeled as a separate, cylindrical TD entity

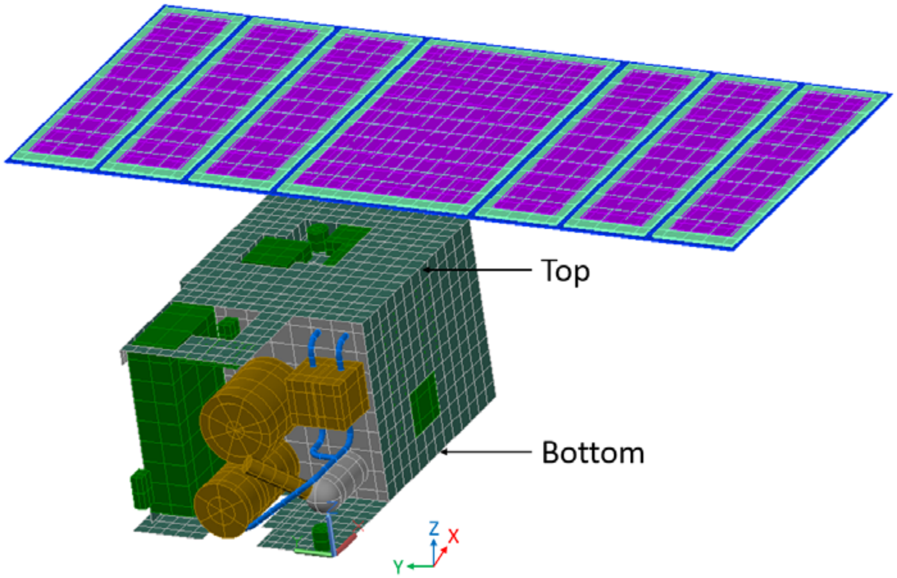

Figure 1: Thermal Model Showing Entire Spacecraft

Figure 2 shows a view of the TD solar array, with the aluminum backing, FR4 substrate, and solar cells labeled.

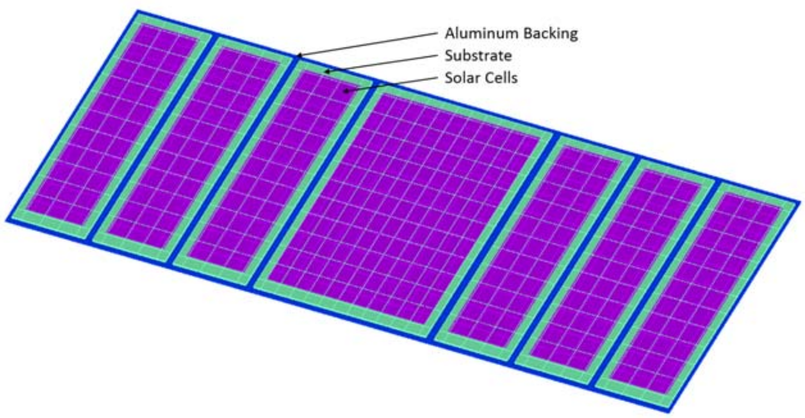

Figure 2: TD Model of Solar Array

Figure 3 shows the outside of the model with the solar array assembly hidden to better view the outer components, which are labeled.

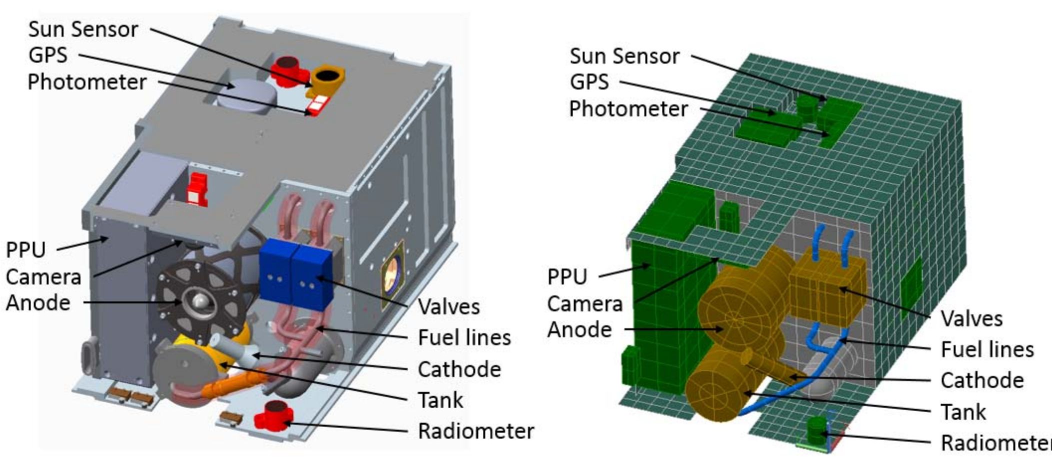

Figure 3. CAD Model (left) and Thermal Model (right) Showing Outer iSAT Components

Figure 4 shows a view of the model revealing the components inside the enclosure. Hidden components include the solar array assembly, the top of the chassis, sun sensor, GPS antenna, and top photometer. The card stack, from top to bottom, includes the Input / Output (I/O), Auxiliary, Flight Computer (FC), Power Distribution, Power Management, and PPU Switch Board. Surface mounted components are thermally connected to the satellite structure via contact conductors calculated from the physical mounting conditions (bolt size, number of bolts, torque, and surface area).

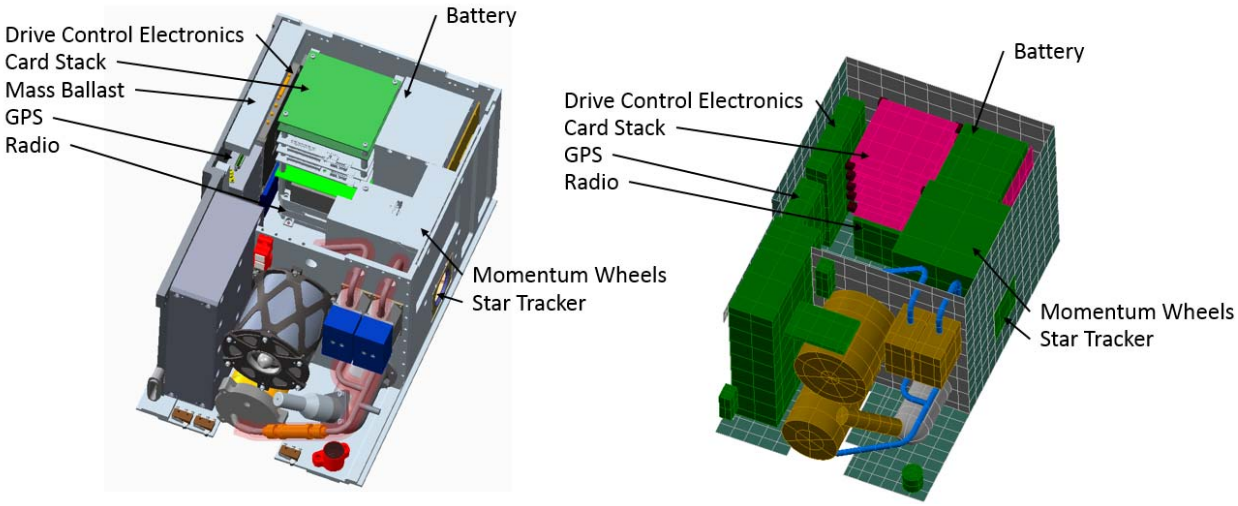

Figure 4. CAD Model (left) and Thermal Model (right) of Showing Inner iSAT Components

Additional views of the components within the spacecraft are included in Appendix B.

II. Model Details

The thermal model was built in Thermal Desktop[1] (TD) and analyzed with SINDA/FLUINT[2]. Four of the six chassis walls were built and meshed using FEMAP[3], and then imported into TD and integrated into the model.

A. Components

The thermal analysis model of iSAT was built to perform analyses to assess component temperatures throughout the key phases of the mission. The components individually modeled for each subsystem are:

- Power
  - Power Management Board (PMB)
  - Power Distribution Board (PDB)
  - PPU Switch Board
  - Lithium-Ion Battery
  - Solar Cells
- Guidance, Navigation and Control (GN&C)
  - Reaction Wheels
  - Star Tracker
  - Inertial Measurement Unit (IMU)
  - Magnetic Torquers (MT)
  - Global Positioning System (GPS) Receiver
  - GPS Antenna
- Propulsion:
  - Thruster
    - Anode
    - Cathode
    - Cathode Heater (internal)

- Thruster Camera
  - Thruster Camera Heater (internal)
- 2 Proportional Flow Control Valves (PFCV)
  - PFCV Heaters (internal)
  - PFCV Insulation
- Power Processing Unit (PPU)
- Solid Iodine Fuel (in fuel tank)
- Fuel Tank
  - Fuel Tank Heater (external)
  - Fuel Tank Insulation
- Fuel Lines (FL)
  - FL Heaters (external)
  - FL Insulation

- Communications
  - Transceiver
  - S-Band Antenna
- Command and Data Handling (C&DH)
  - Flight Computer (FC)
  - Auxiliary Board (AB)
  - Input / Output Board (I/O)

- Payload
  - Iodine HET Diagnostics: Photometers
    - Top Panel
    - Thruster Panel
  - Iodine HET Diagnostics: Radiometers
    - Top Panel
    - Bottom Panel
    - Thruster Panel
- Structure
  - Chassis
  - Solar Panel Backing
  - Solar Panel RF4 Substrate
  - Cathode Mount
  - Separation Switch

The avionics circuit boards (I/O, auxiliary, flight computer, power management, power distribution, and PPU switch) were each modeled separately as surfaces. The Tank was modeled as a cylindrical shell surface with two circle-shaped surfaces as the top and bottom, creating a closed, hollow cylinder. All other components were built as a solid brick with a pre-designated bulk material and outside coating. Each component has a density multiplier applied to represent the correct mass. The thermophysical and optical properties are included in Appendix D.

B. Heat Loads

Nodal uniform heat dissipations are applied for solid and surface heat loads, except for the thruster, tank and fuel lines. The thruster heat load is applied to a ring of nodes within the solid cylinder of the thruster to represent the heated anode. The tank and fuel line heater are thermostatically controlled on / off heaters. A tank heater is applied to the bottom portion of the tank, where the fuel will be contacting the inside wall. The fuel line heater is applied in three separate locations. These heater locations are shown in Appendix E.

The components needing heaters to maintain temperature requirements are listed in Table 1. The maximum and minimum temperature set points of each thermostatically controlled heater were selected based on each components' temperature requirements, which exist to ensure the sublimation of the solid iodine in the tank and the flow of gaseous iodine through the fuel lines. The tank must remain above 90°C and the FLs and PFCVs must remain above 120°C so that the iodine fuel does not solidify and clog the propulsion system. The power of each heater was selected based on the power budget and what is needed to heat the fuel lines and tank to the required temperatures within 50 minutes. The fuel lines are heated in three zones. Zone 1 includes all FLs outside of the chassis, zone 2 is the FL section inside the chassis to the thruster anode, and zone 3 is the FL section inside the chassis to the cathode. The portion of the fuel line inside the cathode mount does not have a heater because heat transfer from the cathode is adequate to maintain its temperature above the requirement.

Table 1: Heater Parameters

Component	Heater Temperature Set Points		Power [W]
	Min [°C]	Max [°C]	
Tank	90	95	8
Fuel lines Zone 1	120	125	2
Fuel lines Zone 2	120	125	1
Fuel lines Zone 3	120	125	1
PFCVs (each)	120	125	3

An additional heater is built in to the cathode, and is turned on for 5 minutes prior to thrust at 120W. The parameters of this heater are not dictated by thermal analysis, but are required for the successful operation of the HET. The cathode heater was modeled as a time dependent heat load, not a thermostatically controlled heater.

C. Heat Dissipation Timeline

The heat dissipation due to component operation with respect to time is a key component in the thermal analysis. The heat dissipation details are largely defined by component specifications and the spacecraft operational timeline. It is assumed that the battery dissipates 5% of its power output while charging and 9% while discharging, based on typical power losses of lithium-ion batteries. Appendix C contains the heat dissipation timeline used in the current model. This timeline begins with propulsion event preparations (the fuel lines, PFCVs, and tank heating), then the thruster fires, and the SC enters a nominal charge state after firing.

The orientation timeline is also documented in Appendix C. The orientation of the SC is either Local Vertical Local Horizon (LVLH), or Solar Inertial. LVLH orients the satellite so that the bottom of the chassis, opposite facing of the solar cells, is always parallel to the surface of the earth, and the thrust axis is aligned with the velocity vector. The solar inertial alignment orients the spacecraft so that the solar panels are always sun facing. Figure 5 shows a diagram of these two orientations, with the thruster plume shown as the blue triangle (although the thruster is not necessarily firing in the shown orientations). The SC turns to LVLH while thrusting and during communications relays, and remains in solar inertial alignment at all other times to charge the battery.

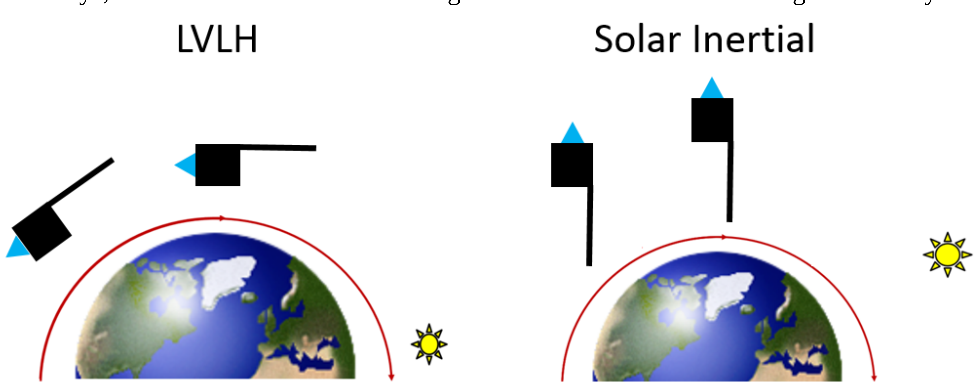

Figure 5: LVLH (Left) and Solar Inertial (Right) Orientation

III. Environments

Exact orbital parameters and altitude remain unknown, because the launch vehicle is unknown. Therefore, the orbital environment has been bracketed by hot and cold cases. The Hot Case Environment and Cold Case Environment have different orbital parameters defined to create the hottest and coldest orbits, respectively. The hot case orbital parameters result in a full sun orbit, whereas the cold case orbital parameters create an orbit in which the

SC is in the earth's eclipse for 37%. Table 2 summarizes the differences between the two cases, and Figure 6 shows an image of the two different orbits. The orientation timeline is the same for each case.

Table 2: Hot and Cold Orbital Environments

Orbital Parameter	Hot Environment	Cold Environment
Altitude [km]	300	570
Orbit Inclination [°]	98	98
Right Ascension of Ascending Node (RAAN) [°]	296	27
Argument of Periapsis [°]	270	270
Calculated Beta Angle [°]	-88	0
Solar Flux[4] [W/m^2]	1414	1322
Albedo[4]	0.30	0.48
Infrared (IR) Planetshine[4] [W/m^2]	240	218

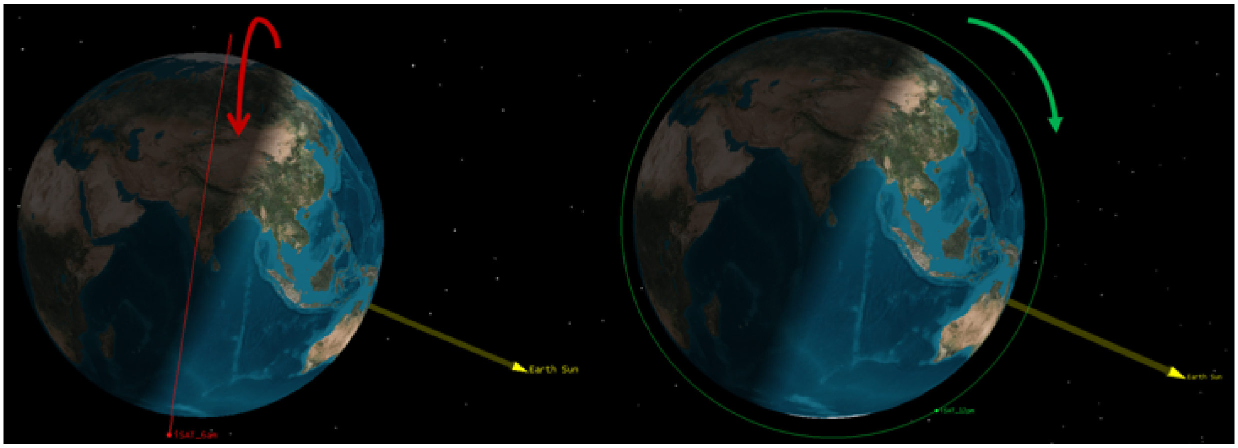

Figure 6: Hot Environment Orbit (Left) and Cold Environment Orbit (Right)

Furthermore, the hot case uses the EOL optical properties for the silver Teflon tape, whereas the cold case uses the BOL optical properties. These are listed with the model materials in Appendix D.

IV. Analysis Results

A hot case and cold case were both analyzed. The cases differ only by environment, the heat dissipation and orientation timelines are the same. This is so because although several components do turn off periodically, the satellite never enters a truly non-operation state. Additional scenarios to be analyzed are discussed later as forward work.

Table 4 and Table 5 list the components modeled, their survivability and operational temperature limits and the maximum and minimum temperatures predicted by analysis for the hot and cold environment cases, respectively. An asterisk next to a survivability or operational temperature designates that this parameter is an estimate and has not been verified. If a survivability or operational temperature parameter is blank, that value is unknown. A color key is shown by Table 3. A margin of only 5°C is used because of the Class D status of this mission. The color key is applied based on a comparison between the predicted and operational temperature, not the survival temperature, because almost all components are operating at least at a minimal capacity for the majority of the time. If a predicted temperature is not highlighted, it is within the temperature range of the component.

Table 3. Results Table Color Key

Red	Above Operational Max
Orange	Within 5 °C of Operational Max
Green	Within 5 °C of Operational Min
Blue	Below Operational Min

Table 4: Results of Hot Environment Case after 1 Orbit

		Survivability [C]		Operational [C]		Predicted [C]	
Subsystem	Component Name	Max	Min	Max	Min	Max	Min
Power	PMB	60	-20	60	-20	44	14
Power	PDB	60	-20	60	-20	46	14
GNC	Momentum Wheels Assembly	70	-30	60	-20	23	12
GNC	Momentum Wheels DCE	70	-30	60	-20	22	16
GNC	Star Tracker	70	-30	65	-30	22	11
GNC	IMU	85	-40	85	-40	21	9
GNC	GPS	85	-40	85	-40	16	7
GNC	GPS Antenna	85	-55	85	-55	19	11
GNC	Sun Sensor	85	-40	75	-25	21	11
GNC	Magnetometer	125	-55	80	-40	17	9
Comm	Transceiver	110	-55	85	-40	15	11
GNC	S-band Antenna	100	-65	100	-65	18	4
C&DH	FC	70	-20	60	-40	54	13
C&DH	AB	85	-20*	85	-20*	42	13
C&DH	I/O	85	-20*	85	-20*	41	14
Propulsion	PPU Switch Card	85*	-20*	85*	-20*	19	13
Propulsion	Camera	85	-24	70	-5	16	10
Propulsion	Thruster	200		200		106	-6
Propulsion	PFCVs	150	12	150	125	127	-12
Propulsion	PPU	125	-20*	125	-20*	19	8
Propulsion	Cathode	450*		450*		300	13
Propulsion	Iodine Fuel	150*		150*	90	95	29
Propulsion	Tank	150*		150*	90	96	-37
Propulsion	Fuel Lines	150*		150*	120	187	20
Propulsion	Fuel Line section to Cathode	150*		150*	120	240	14
Power	Battery	50	-20	40	10	19	13
Power	Solar Cells	110		100		84	-37
Payload	Radiometer Top	150	-60	120	-50	8	-2
Payload	Radiometer Thruster Plate	150	-60	120	-50	20	8
Payload	Radiometer Bottom	150	-60	120	-50	15	4
Payload	Photometers	150	-60	120	-50	14	4
Structure	Separation Switch					12	10
Structure	Solar Panel FR4	110		100		81	-37
Structure	Solar Panel Al Support	110		100		74	-36
Structure	Chathode Mount	120*		120*		10	260
Structure	Chassis	120*		120*		5	34

Table 5: Results of Cold Environment Case after 1 Orbit

		Survivability [C]		Operational [C]		Predicted [C]	
Subsystem	**Component Name**	**Max**	**Min**	**Max**	**Min**	**Max**	**Min**
Power	PMB	60	-20	60	-20	33	2
Power	PDB	60	-20	60	-20	36	2
GNC	Momentum Wheels Assembly	70	-30	60	-20	10	-1
GNC	Momentum Wheels DCE	70	-30	60	-20	12	5
GNC	Star Tracker	70	-30	65	-30	10	-2
GNC	IMU	85	-40	85	-40	12	-4
GNC	GPS	85	-40	85	-40	6	-4
GNC	GPS Antenna	85	-55	85	-55	7	-2
GNC	Sun Sensor	85	-40	75	-25	8	-2
GNC	Magnetometer	125	-55	80	-40	6	-2
Comm	Transceiver	110	-55	85	-40	4	0
GNC	S-band Antenna	100	-65	100	-65	8	-6
C&DH	FC	70	-20	60	-40	44	2
C&DH	AB	85	-20*	85	-20*	32	1
C&DH	I/O	85	-20*	85	-20*	31	2
Propulsion	PPU Switch Card	85*	-20*	85*	-20*	10	-5
Propulsion	Camera	85	-24	70	-5	2	-3
Propulsion	Thruster	200		200		95	-21
Propulsion	PFCVs	150	12	150	125	126	20
Propulsion	PPU	125	-20*	125	-20*	10	-5
Propulsion	Cathode	450*		450*		294	-44
Propulsion	Iodine Fuel	150*		150*	90	95	5
Propulsion	Tank	150*		150*	90	95	5
Propulsion	Fuel Lines	150*		150*	120	198	5
Propulsion	Fuel Line section to Cathode	150*		150*	120	230	0
Power	Battery	50	-20	40	10	9	1
Power	Solar Cells	110		100		80	-61
Payload	Radiometer Top	150	-60	120	-50	3	-8
Payload	Radiometer Thruster Plate	150	-60	120	-50	10	-5
Payload	Radiometer Bottom	150	-60	120	-50	-3	-10
Payload	Photometers	150	-60	120	-50	2	-8
Structure	Separation Switch					1	-2
Structure	Solar Panel FR4	110		100		77	-61
Structure	Solar Panel Al Support	110		100		72	-61
Structure	Chathode Mount	120*		120*		249	-5
Structure	Chassis	120*		120*		26	-6

V. Discussion

The current thermal analysis predicts there to be several components outside of their operational temperature range and a few components within 5°C of their operational temperature limit. After analyzing the results, it is believed that the reasons for these unfavorable temperatures are known, and updates can be made to the design and model to address these issues. There are several additional aspects of the design not represented in the model that must also be considered.

A. Out-of-Range Components

The iodine fuel, fuel tank, PFCVs, and fuel lines all reach temperatures below their low operational limits, but these values are not highlighted in the above tables because this does not occur while the components are operating. These components only operate during propulsion preparation heating, and during thruster firing. They are not operating during charging.

In both the hot and cold environment case, the fuel lines exceed their assumed temperature limit of 150°C. This is because the three heater zones in the model are each controlled by the minimum temperature in that zone. If any part of the fuel line falls below the heater set point, the entire line is heated, and some parts of the line are heated more than necessary. This could be adjusted by changing the heater placement so that only the coldest part of the line is directly heated and the rest of the section is heated through conduction. The heater sensor also needs to be updated to use a single node as the sensor location, rather than the minimum temperature. Furthermore, the 150degC limitation is based on the upper limit of the valves. The fuel lines and heaters can survive a higher temperature, but the exact limit is not yet determined.

The section of fuel line that is within the cathode mount, connecting the fuel line inside the chassis to the cathode, reaches a higher temperature than the rest of the fuel lines because it is attached to the cathode, which is heated with a 120W heat load for 5 minutes prior to thrusting. This temperature is acceptable, because this section of the line is not near the valves, and there is a thermal break between the fuel line inside the cathode mount and the fuel line inside the chassis. The temperature limit of the fuel lines needs to be updated based on heater and insulation selection, rather than basing the temperature limit on the PFCV's limits. The heaters and insulation to be used in flight have not yet been finalized.

Another component of concern is the battery. In the hot environment case, the battery reaches a temperature only 3°C away from its lower limit, and in the cold environment case, its temperature decreases below that limit. In this model, the battery is modeled as being mounted directly to the chassis with nine #6 bolts, with neither insulation nor a heater. Preliminary analysis has shown that mounting the battery with thermally isolating washers of G10 Fiberglass with a thickness of 0.062in, will decrease heat transfer to the chassis enough for the battery to remain within its operational range as long as the spacecraft is operating nominally. A survival heater may be necessary for colder cases (safe mode), or unforeseeable circumstances, when less components are on and the battery is dissipating less heat.

Additionally, the flight computer and other circuit boards do not yet reach their temperature limits, but each is represented by a board level model, with no individual components modeled. The components could reach a higher temperature than what is being predicted. The component temperatures will need to be monitored during testing. The flight computer currently utilizes a thermal strap, which comes standard with the board and cannot be altered, for cooling and will also need to be reevaluated after updates.

The camera, which is positioned over the thruster, and is required to take at least one image of the thruster plume, nears its cold temperature limit in the cold case. This is not a concern because the camera has a built in heater that will turn on when the camera reaches below its operational limit of -5°C. This heater was not modeled.

B. Additional Considerations

There are several additional considerations that must be acknowledged while analyzing these results. Firstly, these results show temperatures from one orbits during which the thruster firing sequence occurs at the start, the thruster fires, and the SC enters a charge mode for the remaining time. In reality, assuming iSAT is orbiting the earth in an orbit with maximum eclipse (worst case cold), it is possible that the spacecraft will need 20 orbits after thrusting to charge the battery enough to be able to begin the propulsion sequence again. After firing, the spacecraft will be in charge mode, in which all non-essential components are turned off in order to conserve power. During the longer charge time, the temperature of all components will drop until they reach a steady cycle of heating and cooling caused by entering and exiting eclipse. Additional components may drop below their minimum operational temperature limit during this time. If there are unexpected issues during operation, the spacecraft will enter into a safe mode, which could have an even colder environment with more components turned off. Safe mode operation has not yet been fully defined, but it will be important to create a steady state safe mode model, to ensure that no components reach lower limits during this mode.

Another important factor is that the fuel lines must be thermally isolated from the thruster, so that the thruster does not act as a heat sink while the heaters are bringing the temperatures up to the required value. Similarly, the tank must be thermally isolated from the chassis wall to which it is mounted. This thermal isolation is crucial because it allows less heater power to be used and prevents sections of the line from being over-heated while others are under-heated.

VI. Forward Work

Many design changes have been made since the start of the iSAT project, due to a variety of challenges including volume constraints, power margin, ability to meet performance requirements of various subsystems, and thermal limits. There are several design changes that are currently in-work, and have not yet been incorporated into the thermal model.

A. Configuration Changes

Model changes included in an upcoming design update include the following:

- The solar panel size and configuration will be updated. The solar panels will be reconfigured to include 3 same-sized panels (the size of the current center panel), which unfold from one another in a single direction, away from the chassis. This change will be made to be able to increase the thickness of the aluminum mounting panel of the solar panels, and still maintain the same number of solar cells.
- The parameters of the PFCVs will be changed to match a different valve that will be used because of a test failure of the current model valve.
- The battery size and location will be updated with slight modifications due to battery cell spacing requirements.
- The locations of several components will shift as a result of the battery updates, star tracker, and star tracker mount.
- The heat dissipation timeline will also be updated due to power constraints and to increase power margin. Currently, the power timeline is being scrubbed to turn off any components that do not need to be on during specific periods of time in order to decrease time needed to charge the SC enough to fire the thruster.
- Fuel line, tank and valve heaters will be updated to remain on after thrust during a period of time to ensure that no iodine is deposited to clog the fuel lines.

B. Changes Due to Thermal Analysis

Updates needed that will result from additional thermal analysis to address current challenges include the following:

- The battery must be thermally isolated with non-thermally conductive washers or standoffs. If this does not thermally isolate the battery enough, a survival heater will need to be added.
- Heater placement, insulation, and outer insulation layer will be finely tuned to decrease temperature gradient on fuel lines.
- Hot and cold environment cases will be analyzed during a longer time period of at least 20 orbits without thrusting. This analysis is crucial because current power analysis estimates that the SC will need to charge for about 20 orbits in a worst case cold environment with maximum eclipse. In the current analysis, the orbit begins with the thruster sequence, and the SC component temperatures generally begin to decrease after the thruster fires, and so temperatures of some components may drop below their minimum operational temperature limits.
- A Safe Mode steady state analysis will also be created. In Safe Mode, all non-essential components will be turned off, and so this scenario could cause colder temperatures than previously modeled. The definition of Safe Mode is still in work.
- Updates must be made to the estimated temperature limits used in Tables 4 and 5.

C. Increased Model Fidelity

A high fidelity thruster model has also been created using Thermal Desktop and imported into the integrated spacecraft model. A model with the low fidelity thruster replaced with the high fidelity version was analyzed and compared to the same model with the thruster fidelity being the only difference. This comparison showed that the thruster model change affected the other component temperatures by only a 1 to 2°C difference. Because of the small difference, next iterations of the design will keep the low fidelity thruster because model run time is much faster. The high fidelity thruster model also showed that the thruster was not in danger of exceeding its maximum operational temperatures with its current power distribution and requirements. The high fidelity model of the propulsion system will be used for final CDR analysis.

D. Testing

Temperature cycle and model correlation testing will occur within a thermal vacuum chamber. The General Environmental Verification Standard (GEVS)[5] will be used as the iSAT test standard. The temperature cycling will start with the SC dropping to a set cold temperature, then it will reach a steady state and perform a set of day in the life operations, the temperature will be raised to a set hot temperature, then it will reach a steady state and perform a set of day in the life operations, and repeat. The model correlation testing will be performed specifically so that the model can be adjusted to match results. During this test designated sets of components will be turned on and the SC will be allowed to reach steady state.

VII. Conclusion

Many cubesats and smallsats face very similar challenges due to power, heat, and volume constraints. What makes iSAT unique to many cubesats is its very high power density. During operation, the thruster needs 200W to successfully fire, resulting in a very high peak power. Most cubesats have solar panels all around the sides of the spacecraft to generate power from the sun. iSAT does not because the outer surface of the SC is needed as a thermal radiator, with silver Teflon tape covering all available spaces to minimize solar loading. Another challenge is the thermal control of the propulsion components that must be controlled to specific temperatures. The temperatures of the tank, fuel lines and valves must each be maintained above a specific value, without heating up the rest of the cubesat while minimizing heater power. Proper thermal isolation and insulation are crucial to attain a compatible design. Like the majority of cubesats, an additional challenge is the volume constraint. Almost one third of the satellite is taken up by the propulsion system. Other voluminous components include the battery (due to the high power density) and the 6-board card stack. The components are very closely packed, which attributes to a challenging assembly process. Furthermore, the potential orbit of a cubesat is often uncertain, since it is not a primary payload and cannot dictate launch vehicle parameters. This is the case for iSAT.

The thermal design and analysis of iSAT has been challenging, but many design and concept of operations changes have been made to deal with problems as they arose. Several components have been removed as a result of volume and power constraints. Currently, the thermal analysis shows that the fuel lines reach temperatures above their maximum operational limit, and the battery reaches temperatures below its minimum operational limit. There is, however, a path forward to address both issues. The future design and analysis of the SC will involve many updates due to mission changes, power limitations, volume limits, and design updates related to thermal concerns. To reach a full confidence level, the fidelity of several components will be increased, and the time period of analysis will also be lengthened. The spacecraft design, component locations, and thermal systems will continue to change and evolve as the design matures even further.

Appendix

A. Previous Design and Thermal Model Images

The initial study concept for iSAT was a 6U cubesat with a deployable thruster and deployable solar panels, shown in Figure 7.

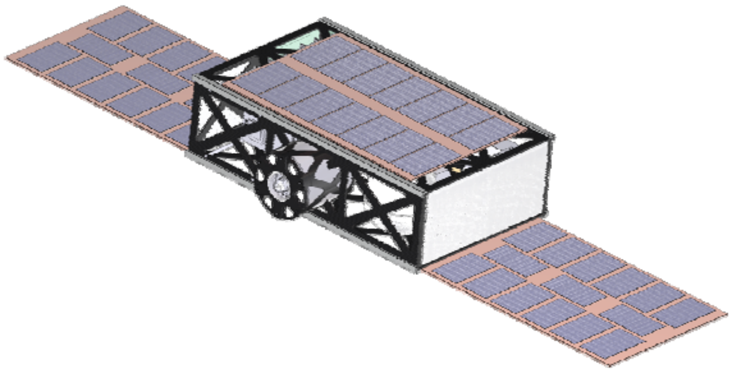

Figure 7: Initial Study Concept

The first detailed thermal model built was created after the concept had grown to a 12U and still had a deployable thruster and three spaces for payloads, shown in Figure 8.

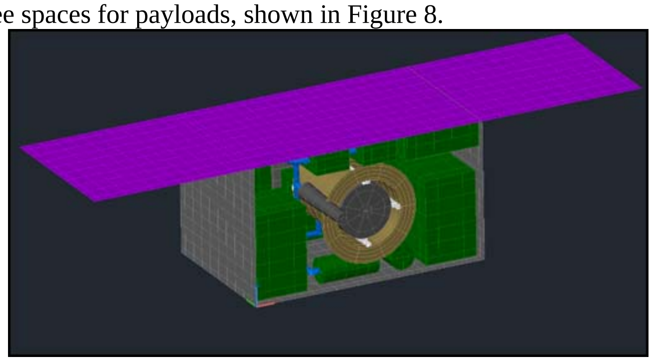

Figure 8: Initial Thermal Model

It was realized that the thruster and cathode reached too high of temperatures to be kept inside the spacecraft at times, and that the propulsion system could be arranged so that a deployment mechanism was not needed. This led to design to 'DAC0', shown in Figure 9.

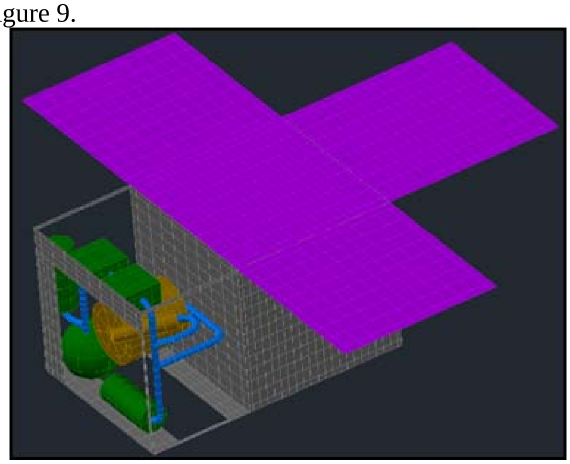

Figure 9: DAC0

The large PPU did not completely fit within the enclosed area of the spacecraft, and so it was moved so that about one third of it stuck out into the propulsion section, outside the enclosed portion of the chassis. This is shown in Figure 10.

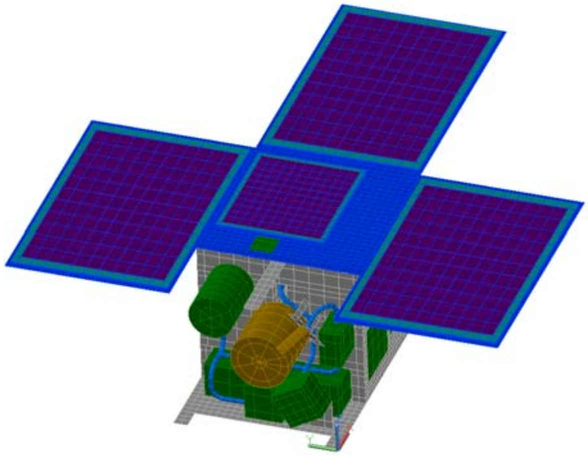

Figure 10: DAC1 / PDR Model

The solar panel on the body of the chassis caused too much heat to be absorbed by the components inside, and the top section of the chassis was needed for the sun sensor, radiometer, and photometer placement. Also, the solar arrays were changed to a main panel plus winglets design. This design is shown in Figure 11.

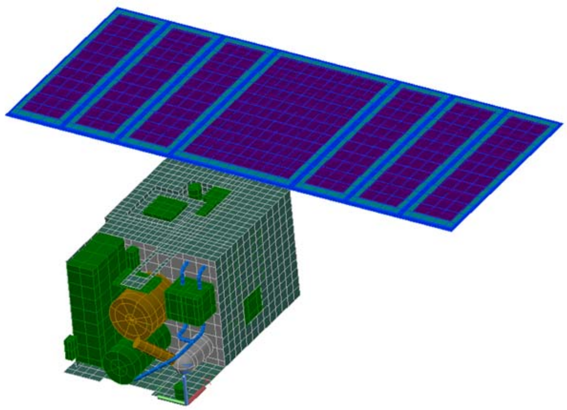

Figure 11: DAC2 / Post-PDR Model

Additional changes were made in DAC3, including removing a section of the bottom panel to make room for the size of the tank, and adding supports to hold the camera above the thruster, as shown in Figure 12.

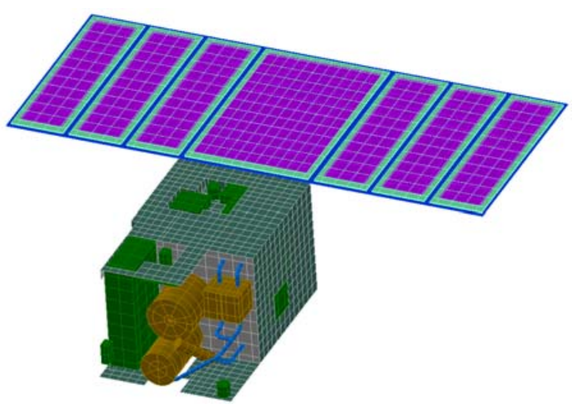

Figure 12: DAC3 Model

B. Additional Images of the Current Model

Figure 13 shows the “front” of iSAT, which is the face with all of the propulsion components in view.

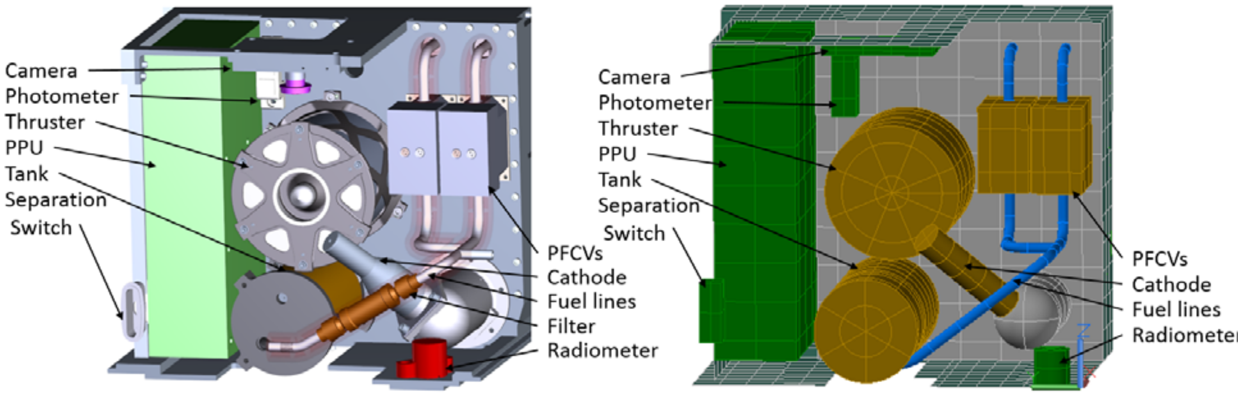

Figure 13: Front View of CAD (left) and Thermal Model (right)

Figure 14 shows a detailed image of the top of the SC, with additional components inside the chassis enclosure labeled.

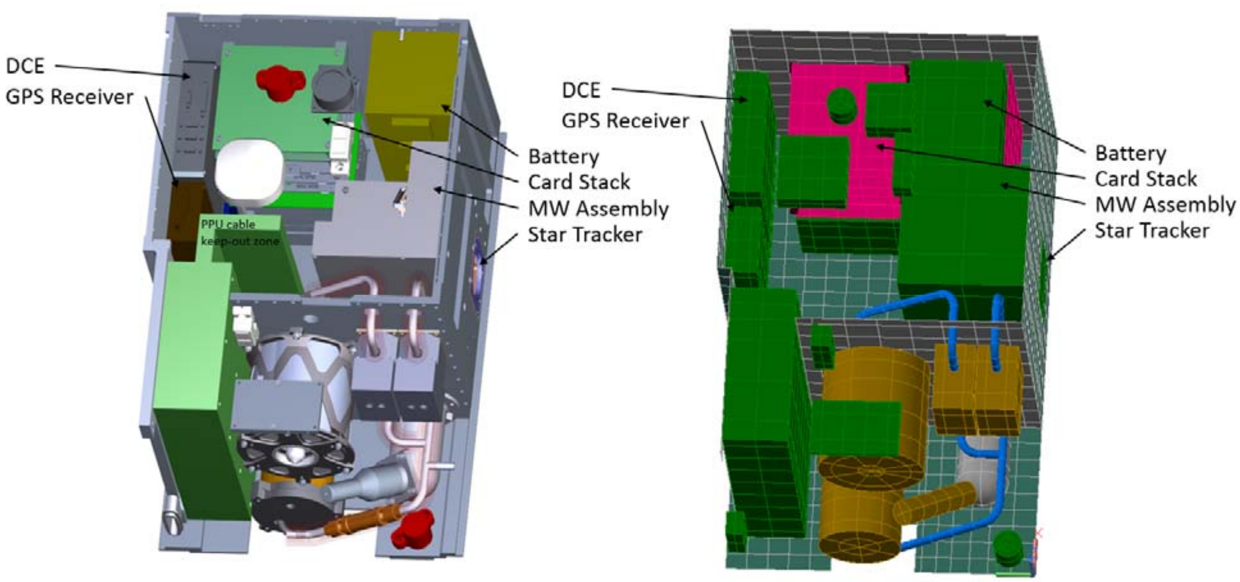

Figure 14: Top View of CAD (left) and Thermal Model (right)

C. Heat Dissipation and Orientation Timeline of Current Model

In Figure 12, the tank and fuel lines are shown to have power on during thruster preparation and thruster firing. This value indicates that the heaters are on, but are being thermostatically controlled, not heating the components with a constant heat load.

Table 6: Heat Dissipation Timeline

Note	Orientation	Phase Length [min]	Time [min]	Time [s]	Thruster	Cathode	PPU
		0	0	0	0	0	0
PPU on	SI	1	1	60	0	0	8.62
Tank Heater on	SI	14	15	900	0	0	8.62
Line & Valve Heaters on	SI	21	36	2160	0	0	8.62
Prop dependent components on	SI	9	45	2700	0	0	8.62
Cathode Heater on	SI	5	50	3000	0	0	8.62
Thrust	LVLH	5	55	3300	68.377	65.377	57.32
Recovery / Prop dependent components on / Change to SI / MT on	SI	4	59	3540	0	0	0
Recovery / Prop dependent components on	SI	10	69	4140	0	0	0
Recovery / Eclipes begins / Change to LVLH	LVLH	4	73	4380	0	0	0
Recovery / Prop dependent components off	LVLH	14	87	5220	0	0	0
Recovery ends	LVLH	8	95	5700	0	0	0
Change to SI / MT off	SI	4	99	5940	0	0	0
Eclipse Ends / Enter Charging	SI	96.06	195.06	11703.86	0	0	0

Time [s]	Heaters: Tank	Heaters: Fuel lines	Heaters: Valves	Heaters: Cathode	DCE	Magnetometer	GPS	IMU	Star Tracker	Sun Sensor	Reaction Wheels
0	0	0	0	0	1.667	1.511	-0.987	0.22	2.222	0.083	1.797
60	0	0	0	0	1.667	1.511	0.013	0.22	2.222	0.083	1.797
900	8	0	0	0	1.667	1.511	0.013	0.22	2.222	0.083	1.797
2160	8	4	6	0	1.667	1.511	0.013	0.22	2.222	0.083	1.797
2700	8	4	6	0	1.667	1.511	2.047	0.22	2.222	0.083	1.797
3000	8	4	6	120	1.667	1.511	2.047	0.22	2.222	0.083	5.392
3300	8	4	6	0	1.667	1.511	2.047	0.22	2.222	0.083	1.797
3540	0	0	0	0	1.667	1.511	2.047	0.22	2.222	0.083	5.392
4140	0	0	0	0	1.667	1.511	2.047	0.22	2.222	0.083	1.797
4380	0	0	0	0	1.667	1.511	0.013	0.22	2.222	0.083	5.392
5220	0	0	0	0	1.667	1.511	0.013	0.22	2.222	0.083	1.797
5700	0	0	0	0	1.667	1.511	0.013	0.22	2.222	0.083	1.797
5940	0	0	0	0	1.667	1.511	0.013	0.22	2.222	0.083	5.392
11703.86	0	0	0	0	1.667	1.511	0.013	0.22	2.222	0.083	1.797

Time [s]	Magnetorquer	Transceiver	Photometers	Radiometer	I/O Board	Auxiliary Board	FC	Thruster Camera	Cortex 130s	PPU Switch	Battery
0	0	0	0	0	1.111	2.222	7.259	1.111	7.604	0	1.358947
60	0	0	0	0	1.111	2.222	7.259	1.111	7.604	0	1.865263
900	0	0	0	0	1.111	2.222	7.259	1.111	7.604	0	2.286316
2160	0	0	0	0	1.111	2.222	7.259	1.111	7.604	0	2.812632
2700	0	0	0.054	0.081	1.111	2.222	7.259	1.111	7.604	0	2.926789
3000	0	0	0.054	0.081	1.111	2.222	7.259	1.111	7.604	0	9.431789
3300	0	0	0.054	0.081	1.111	2.222	7.259	1.111	7.604	2	12.63489
3540	7.28	0	0.054	0.081	1.111	2.222	7.259	1.111	7.604	0	2.098105
4140	7.28	0	0.054	0.081	1.111	2.222	7.259	1.111	7.604	0	1.908895
4380	7.28	0	0	0	1.111	2.222	7.259	1.111	7.604	0	1.983947
5220	7.28	0	0	0	1.111	2.222	7.259	1.111	7.604	0	1.794737
5700	7.28	0	0	0	1.111	2.222	7.259	1.111	7.604	0	1.794737
5940	0	0	0	0	1.111	2.222	7.259	1.111	7.604	0	1.600789
11703.86	0	4.5/0.5 alt	0	0	1.111	2.222	7.259	1.111	7.604	0	2.652527

D. Component TD Model Materials

In the following tables, if the material name is followed by an asterisks, the material properties are temperature dependent and the values shown in the tables are the properties of that material at 20°C.

Table 7. Thermophysical Properties of Components

Subsystem	Component	Bulk Material	Conductivity [W/m/°C]	Density [kg/m^3]	Heat Capacity [J/kg/°C]
Power	PMB	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
Power	PDB	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
GNC	Momentum Wheels Assemb	Al 6061-T6*	151.517	2711.93	872.864
GNC	Momentum Wheels DCE	Al 6061-T6*	151.517	2711.93	872.864
GNC	Star Tracker	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
GNC	IMU	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
GNC	GPS Receiver	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
GNC	GPS Antenna	Al 6061-T6*	151.517	2711.93	872.864
GNC	Sun Sensor	Al 6061-T6*	151.517	2711.93	872.864
GNC	Magnetometer	Al 5052	142.059	2684.69	921.096
Comm	Radio Transceiver	Al 6061-T6*	151.517	2711.93	872.864
GNC	S-band Antenna	Duroid RT 6002	0.600032	2100	921.096
C&DH	FC	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
C&DH	AB	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
C&DH	I/O	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
Propulsion	PPU Switch Card	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
Propulsion	Camera	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
Propulsion	Thruster anode	Carbon Phenolic*	1.3657	1432.9	1085.73
Propulsion	PFCVs	Hastelloy C*	8.73413	8938.3	386.131
Propulsion	PFCVs Insulation	MLI01	20.7688	27679.9	4186.8
Propulsion	PPU	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
Propulsion	Cathode	Graphite	112.152	2162.64	711.999
Propulsion	Fuel	Iodine	0.449	4940	429
Propulsion	Tank	Hastelloy C*	8.73413	8938.3	386.131
Propulsion	Tank Insulation	MLI01	20.7688	27679.9	4186.8
Propulsion	Fuel Lines	Hastelloy C*	8.73413	8938.3	386.131
Propulsion	Fuel Lines Insulation	MLI01	20.7688	27679.9	4186.8
Power	Battery	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
Power	Solar Cells	PCB (4.6% Cu by Volume)	18.04	2259.77	1544.11
Payload	Radiometers	Al 6061-T6*	151.517	2711.93	872.864
Payload	Photometers	Al 6061-T6*	151.517	2711.93	872.864
Structure	Separation Switch	Al 6061-T6*	151.517	2711.93	872.864
Structure	Solar Panel FR4	FR4	0.3	1850	600
Structure	Solar Panel Al Support	Al 7075-T73	155.766	2823.35	958.777
Structure	Cathode Mount	Al 7075-T6	151.517	2711.93	872.864
Structure	Chassis	Al 7075-T73	155.766	2823.35	958.777

Table 8. Optical Properties of Components

Subsystem	Component	Optical Coating	Emissivity	Absorptivity
Power	PMB	PCB	0.93	1
Power	PDB	PCB	0.93	1
GNC	Momentum Wheels Assembly	Black Anodized	0.88	0.88
GNC	Momentum Wheels DCE	Black Anodized	0.88	0.88
GNC	Star Tracker	Al 6061-T6	0.3	0.8
GNC	IMU	Al 6061-T6	0.3	0.8
GNC	GPS Receiver	Al 6061-T6	0.3	0.8
GNC	GPS Antenna	Polyethylene Film 2 layers	0.183	0.075
GNC	Sun Sensor	Gold	0.023	0.299
GNC	Magnetometer	AL 6061-T6	0.3	0.8
Comm	Radio Transceiver	Alodine finished Aluinum	0.12	0.45
GNC	S-band Antenna: Inside	Antenna Inside	0.82	0.39
GNC	S-band Antenna: Outside	Antenna Outside	0.8	0.8
C&DH	FC	PCB	0.93	1
C&DH	AB	PCB	0.93	1
C&DH	I/O	PCB	0.93	1
Propulsion	PPU Switch Card	PCB	0.93	1
Propulsion	Camera	Al 6061-T6	0.3	0.8
Propulsion	Thruster Anode Face	Ceramic	0.5	0.2
Propulsion	Thruster other faces	Gold	0.023	0.299
Propulsion	PFCVs Insulation	Aluminum on Black Kapton	0.03	0.12
Propulsion	PPU	Black Anodized	0.88	0.88
Propulsion	Cathode	Graphite	0.85	0.88
Propulsion	Fuel	Solid Iodine	0.5	0.2
Propulsion	Tank Inside	Hastelloy	0.11	0.39
Propulsion	Tank Insulation	Aluminum on Black Kapton	0.03	0.12
Propulsion	Fuel Lines inside	Hastelloy	0.11	0.39
Propulsion	Fuel Lines Insulation	Aluminum on Black Kapton	0.03	0.12
Power	Battery	AL 6061-T6	0.3	0.8
Power	Solar Cells	Inactive Panel	0.84	0.92
Payload	Radiometers	Z-93 White Paint	0.92	0.17
Payload	Photometers	Z-93 White Paint	0.92	0.17
Structure	Separation Switch	Al 6061-T6	0.3	0.8
Structure	Solar Panel FR4	FR4	0.88	0.88
Structure	Solar Panel Al Support: Top	Black Anodized	0.88	0.88
Structure	Solar Panel Al Support: Bottom BOL	Silver Teflon Tape 10mil BOL	0.87	0.09
Structure	Solar Panel Al Support: Bottom EOL	Silver Teflon Tape 10mil EOL, 13mo.	0.83	0.13
Structure	Cathode Mount BOL	Silver Teflon Tape 10mil BOL	0.87	0.09
Structure	Cathode Mount EOL	Silver Teflon Tape 10mil EOL, 13mo.	0.83	0.13
Structure	Chassis: Outside BOL	Silver Teflon Tape 10mil BOL	0.87	0.09
Structure	Chassis: Outside EOL	Silver Teflon Tape 10mil EOL, 13mo.	0.83	0.13
Structure	Chassis: Inside	Black Anodized	0.88	0.88

E. Heater and Heat Load Locations

The thruster nodes to which heat is applied, to represent the anode in the low fidelity thruster model, are shown in Figure 15. The red circle, crossing through eight nodes, indicates these.

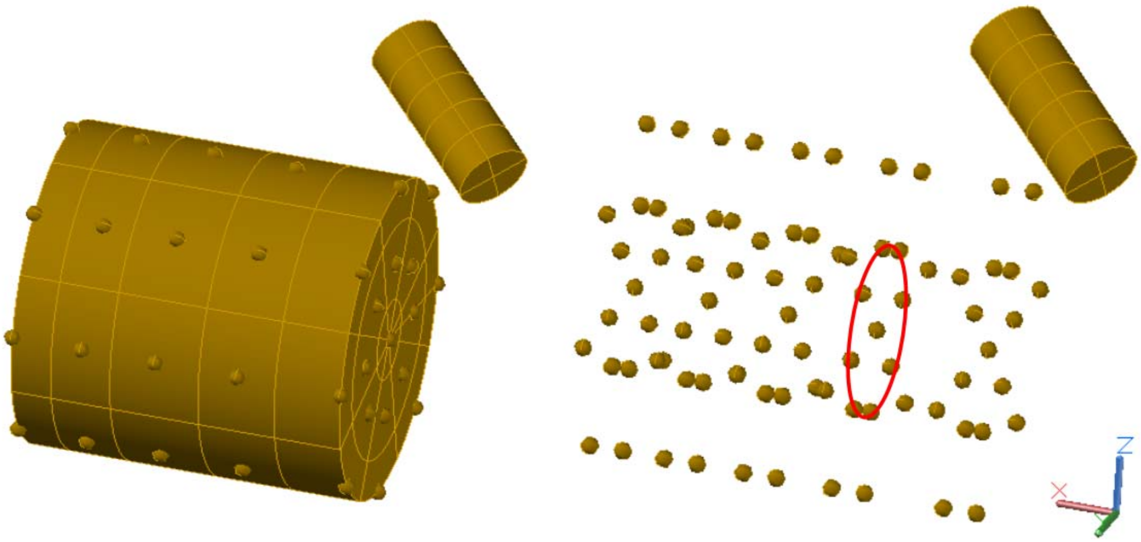

Figure 15: Thruster Location of Power Applied

Figure 16 indicates the fuel line heater zones. Each zone is shown in a different color and are controlled by three separate heaters with three separate sensor points. The sensor points read from the minimum temperature of each zone.

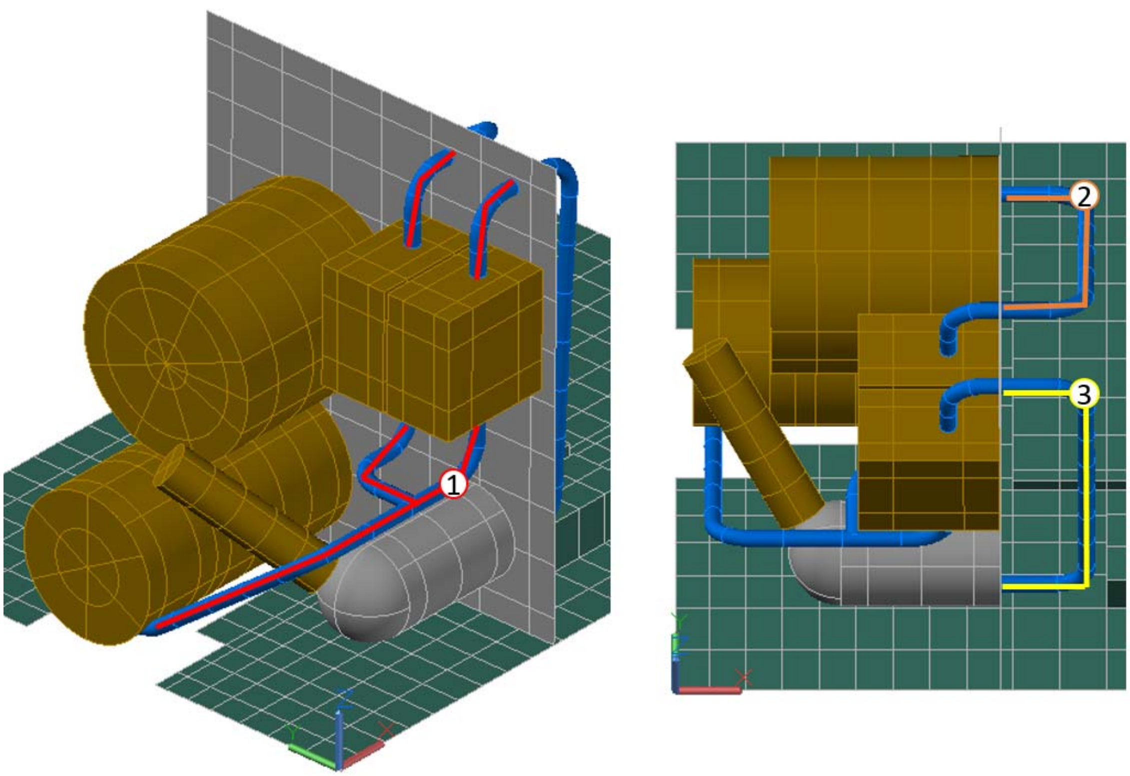

Figure 16: Fuel Line Heater Zones

The red-colored part of the tank in Figure 17 indicates the area to which the tank heater is applied. It is only applied to a section of the tank, because the fuel will be pushed down into this section of the tank by a spring-forced pushed plate.

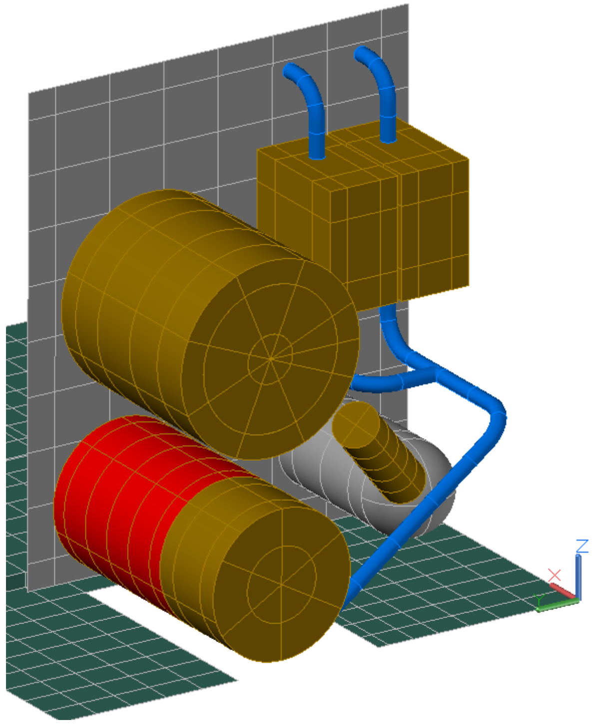

Figure 17: Tank Heated Surface (Highlighted in Red)

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