High-Fidelity Spacecraft Thermal Modeling: Synthesis of STK, SPENVIS, MATLAB, Simulink, and Thermal Desktop using Model-based Systems Engineering

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ABSTRACT

Verification and Validation (V&V) by analysis for required spacecraft Heater Wattage (HW) and Radiator Area (RA) is an iterative procedure highly dependent on spacecraft surface area, absorptivity, emissivity, orbital position, orbital attitude, and operational heat generation. The Alabama Burst Energetics eXplorer (ABEX) mission adopts a Model-Based Systems Engineering (MBSE) approach to analysis wherein model strengths and weaknesses are considered synergistically and integrated using the Systems Modeling Language (SysML) to create a System of Models (SoM). In this work, a procedure for comprehensive spacecraft thermal modeling is detailed using MBSE-centric Modeling and Simulation (M&S) practices including a SysML model as a central source of data truth. Because the analytical models in Systems Tool Kit (STK), MATLAB, Simulink, Thermal Desktop, and the Space Environment Information System (SPENVIS) source input data originally from the SysML model, input data pedigree V&V is only required in SysML. In the analytical models, STK simulates spacecraft modes of operation and communication profiles to export transient spacecraft position and velocity state vectors, solar position state vectors, Earth position state vectors, and unit vectors orthogonal to each spacecraft face, among non-thermal data. An orbital model in SPENVIS produces corpuscular radiation integral flux data for the determination of Charged Particle Heating (CPH), and the MATLAB model imports the STK and SPENVIS data. In MATLAB, heat fluxes from solar emission, Earth emission, Earth albedo, CPH, and Free Molecular Heating (FMH) are calculated and converted to absorbed heat values; radiation surface reflectivity is calculated using specular, spectral Fresnel relationships accounting for complex, spectral refractive indices of both the spacecraft surface coating material and base layer material, surface coating material thickness, and radiation Angle of Incidence (AOI). The MATLAB model utilizes an isothermal energy balance to output a low-fidelity HW and RA value required to stay above and below component operational temperature bounds, respectively. In Simulink, component thermal capacitances are distributed in a thermal resistance network with each discrete spacecraft component considered isothermal; absorbed heat and advanced reflectivity calculations are also recalculated per component. An array of HW and RA values is generated between zero and twice the value provided by the MATLAB isothermal model to create a matrix of potential HW and RA combinations. The Simulink model determines an operational envelope of viable HW and RA combinations for user-defined heater and radiator locations; acceptable HW and RA combinations are those that result in component temperatures within operational boundaries. The HW and RA combinations at the edges of the Simulink-derived operational envelope are provided to a three-dimensional, geometry-specific Thermal Desktop model wherein high-fidelity HW and RA values can be analyzed specific to mounting interface considerations. In this SoM progression from MATLAB to Simulink to Thermal Desktop driven by data inputs from STK and SPENVIS with a central source of truth for all models based in SysML, uncertainty and risk regarding thermal control analysis results are systematically mitigated.

NOMENCLATURE

A	Area	[m^2]
c	Specific Heat	[J/kg-K]
E	Energy	[J]
f_{shadow}	Shadow Fraction	[-]
h_{alt}	Satellite Altitude	[km]
H	Height	[m]
L	Length	[m]
m	Mass	[kg]
P	Power	[W]
Q	Heat	[W]
$Q^{"}$	Heat Flux	[W/m^2]
r	Surface Roughness	[μm]
R	Resistance	[K/W]
t	Time	[s]
T	Temperature	[K]
W	Width	[m]
v	Velocity	[km/s]
$\bar{\hat{x}}_{cs,X/Y/Z+/-}$	Unit Vector Orthogonal to Spacecraft Face	
$\bar{x}_{e-cs}$	Earth to Satellite Vector	[km]
$\bar{x}_{s-cs}$	Sun to Satellite Vector	[km]
α	Absorptivity	[-]
ε	Emissivity	[-]
η	Efficiency	[-]
θ	Angle	[rad]
ξ	Solar Zenith angle	[rad]
ρ	Reflectivity	[-]
σ_{sbc}	Stefan-Boltzmann Constant	[W/m^2-K^4]
φ	Integral Particle Flux	[1/cm^2-s]

Recurring Subscripts

alb	Denotes Radiative Albedo
AOI	Denotes per Angle of Incidence
array	Denotes Array
battery	Denotes Battery
c	Denotes per Component
cell	Denotes Solar Cell
cons	Denotes Consumed
cont	Denotes Contact
cs	Denotes Whole-Satellite (CubeSat)
diode+line	Denotes Power Transport Circuitry
dis	Denotes Dissipated
dist	Denotes Distribution
E	Denotes per Energy Level
Earth	Denotes Earth as Source
ems	Denotes Radiative Emission
EPS	Denotes Electrical Power System
f	Denotes per Face
GCR	Denotes Galactic Cosmic Rays
gen	Denotes Generated Value
HW	Denotes Heater Wattage
in	Denotes Inward Directionality
incident	Denotes Rectilinearly Oncoming
max	Denotes Hot TES Value
min	Denotes Cold TES Value
out	Denotes Outward Directionality
proj	Denotes Projection
rad	Denotes Radiator
SEP	Denotes Solar Energetic Particles
solar	Denotes Sun as Source
supp	Denotes Supplied
surr	Denotes Surroundings
sys	Denotes System Variable
TE	Denotes Trapped Electrons
temp	Denotes Temperature
TES	Denotes per Thermal Environment State
total	Denotes Total
TP	Denotes Trapped Protons
VAB	Denotes Van Allen Belt
λ	Denotes per Wavelength

INTRODUCTION

All models are wrong, but some are useful[1]. Many definitions for the term model exist, which should be consolidated under an ontological definition[2], but the most applicable definition for engineers currently arises from the Department of Defense Instruction 5000.59[3].

> Model: a physical, mathematical, or otherwise logical representation of a system, entity, phenomenon, or process.

For spacecraft engineers, a useful system model is one that is descriptive, analytical, contains unambiguous semantics, and can integrate information from various technical domains or peripheral models. One of the four pillars of MBSE[4,5] is a modeling language wherein syntax defines how an expression is structured to be machine-readable and semantics define what an expression means when it is structured that way. Many systems engineers use SysML to generate useful system models, and the descriptive, analytical, and unambiguous properties of SysML provide a foundation for rigorous thermal modeling. Importantly, models built in SysML can be executable, which leads to the definition of simulation as, "a method for implementing a model." [6] The present work details how models in SysML, STK, SPENVIS, MATLAB, Simulink, and Thermal Desktop can be organized in a SoM using MBSE principles to evaluate thermal Technical Performance Measures (TPM). A visual overview is provided in Figure 1.

Problem Description

Space is cold, radioactive, and electromagnetically active[7,8]. The space vacuum temperature is considered 2.7 K[9] whereas the surface of the Sun is considered 5,780 K[10], so a given spacecraft thermal environment is highly dependent on the intended mission operational environment. The Parker Solar Probe, setting records for near solar proximity[11], would have dramatically differing thermal control mechanics than the Voyager missions, setting records for far solar proximity[12]. While some missions require cryogenic coolers or radioisotope thermoelectric generators, many missions can be thermally evaluated in terms of power required to heat the spacecraft during its coldest Thermal Environment State (TES) and the RA required to reject heat during the hottest TES. HW and RA comprise, for most Low Earth Orbit (LEO) spacecraft, the two most important thermal TPM. Evaluating these TPMs begins with a characterization of the thermal operational environment, translation of the thermal environment to absorbed heat for the size and optical properties of a given spacecraft, and consideration of ohmic heating from spacecraft electrical operation[10]. A thermal energy balance can be determined using these inputs, and, for an isothermal spacecraft model, engineers can assume RA and HW to calculate temperature, assume temperature and RA to calculate HW, or assume HW and temperature to calculate RA. Calculating temperature directly from an isothermal model yields negligible insight into spacecraft design considerations, but setting temperature as a boundary condition can inform TPM baselines.

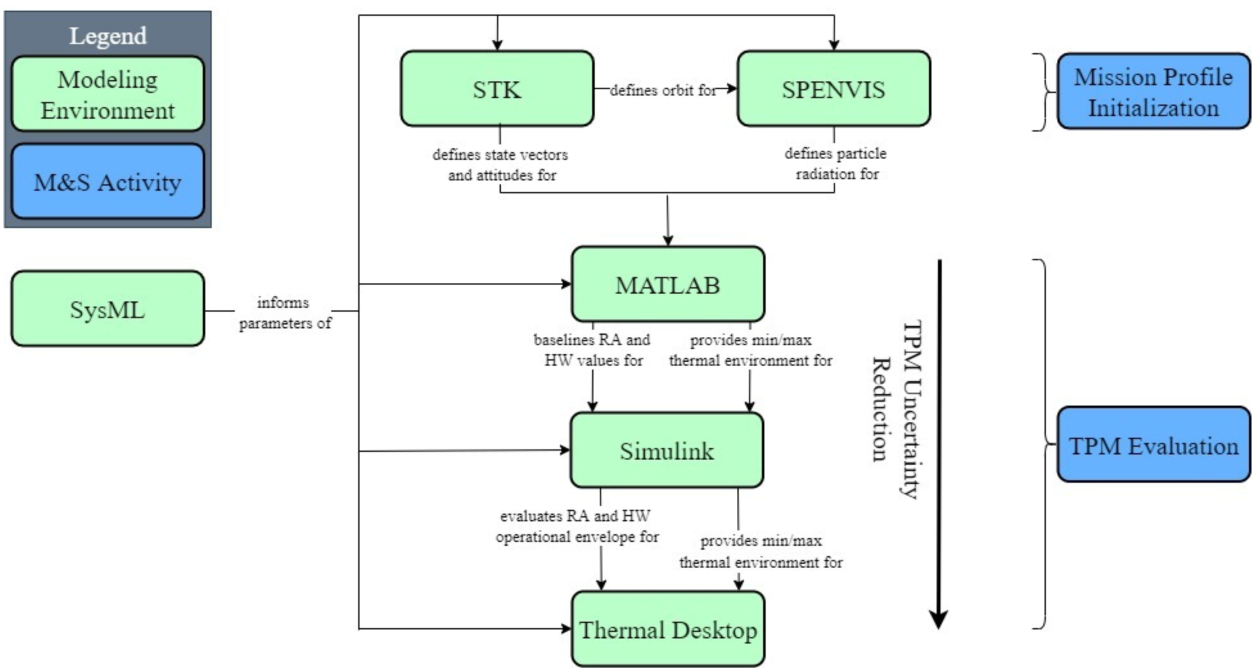

Figure 1: System of Models Organization and Process

Isothermal models are useful starting points for thermal engineers; they provide sanity checks for mission concept reviews. However, the lumped capacitance assumption in isothermal models is not sufficient to characterize HW and RA TPMs because spacecraft are not isothermal. Solar arrays can exceed 100°C[13] with avionics remaining between 0°C and 40°C, and time constants for spacecraft components are on the order of minutes to hours, not days[10]. The location of heaters impacts their ability to maintain component temperatures within operational and extreme bounds; the thermal conductivity of the physical path between a radiator and solar array impacts the amount of heat rejected. Finally, an isothermal model may assume deployable solar arrays, which can exceed 100°C in the Sun and less than -80°C in eclipse conditions, are the same temperature as all other spacecraft components, drastically impacting isothermal results for required RA and HW. Isothermal models can inform higher fidelity models, but they have limitations. When transitioning from a lower fidelity, possibly isothermal model to a higher fidelity, non-isothermal model, V&V must be performed on all models to ensure consistency between the models. National Aeronautics and Space Administration (NASA)-STD-7009A: Standard for Models and Simulations[14] provides a framework for performing V&V for M&S, stating,

> The primary purpose of this NASA Technical Standard is to reduce the risks associated with M&S-influenced decisions by ensuring the complete communication of the credibility of M&S results.

Some may argue within reason the concept of simulation applies to verification by test, but for the purposes of this work, M&S will refer solely to verification by analysis. Verification by analysis is a predicted compliance of a design to imposed requirements, and it is primarily used when accurate analysis is possible, testing is not cost-effective, or verification by inspection is inadequate[15]. While analysis is a verification method and M&S is an analytical method, the model itself, including how the model interfaces with other models, must be verified and validated for the requirement compliance effort.

NASA-STD-7009A is executed first by determining the criticality of the M&S effort, or if failure to perform V&V on the M&S effort will probably result in mission failure. The standard provides guidance for criticality assessment, and, if the M&S effort is deemed mission critical, a M&S credibility assessment is performed including eight factors: data pedigree, verification, validation, input pedigree, uncertainty characterization, results robustness, M&S history, and M&S process/product management. Literature has been written on each of these factors independently[5,6,7,15]; the focus of this work is limited to data pedigree, input pedigree, verification, and M&S management. Contextual qualifications for these factors from NASA-STD-7009A are provided[14].

> Data Pedigree: Is the pedigree (and quality) of the data used to develop the model adequate or acceptable?
>
> Input Pedigree: Is the pedigree (and quality) of the data used to setup and run the model adequate or acceptable?
>
> Verification: Were the models implemented correctly, per their requirements/specifications?
>
> M&S Management: How well managed were the M&S processes and products?
>
> Uncertainty Characterization: Is the uncertainty in the current M&S results appropriately characterized? What are the sources of uncertainty in the results and how are they propagated through to the results of the analysis?

Each of these factors is ranked on a scale from 1-4 with 4 being the highest level. A high data pedigree ranking signifies the data used to develop the model had a high degree of traceability and source confidence whereas a high input pedigree ranking signifies the data used to develop the model was used in the execution or simulation of the model. A high verification ranking signifies model mathematical rigor was of high quality. M&S management refers to process definition, process control, continuous improvement, and change management, but here it should also include model cohesion. Just as there are Systems of Systems (SoS) wherein independent systems with independent authorities are managed under a single SoS, managing multiple models of varying fidelity using SysML should be considered a SoM with independent model authorities. Within that SoM, rankings for some M&S V&V factors, such as verification and M&S management, should remain constant during SoM execution, whereas rankings for other M&S V&V factors, such as data pedigree, input pedigree, and uncertainty characterization, should increase. Uncertainty in HW and RA value determination would decrease in the progression from an isothermal to non-isothermal model, and the decrease in uncertainty over time is a hallmark of a well-defined TPM[16]. Because model parameters such as surface areas, thermal conductivities, and optical properties used in multiple models exist in the SysML model as a central source of truth, data pedigree and input pedigree must only be evaluated in SysML such that other models are importing data from SysML correctly.

The ABEX Mission

ABEX is used as a case study for the present thermal modeling methodology, and mission architecture characterization is warranted. ABEX is the 12U flagship mission of the Alabama CubeSat Initiative and is the largest collegiate satellite program in the world with 80+ students and 15+ faculty collaborating on a single satellite at a given time. The ABEX program is unique in that its workforce is comprised of individuals at seven colleges and universities around the state of Alabama[17]; its astrophysics mission is to study the low energy, prompt emission of Gamma-ray Bursts in both gamma and X-ray spectra. To reduce atmospheric and Van Allen Belt (VAB)-induced noise in both gamma and X-ray detectors, ABEX originally pursued a highly elliptical orbit with an apogee of 60,000 km and a perigee of 300 km. Data collection would occur at apogee, and downlink would occur at perigee. A full Space Radiation Environment (SRE) characterization was performed in Halvorson et al.[18] for this orbit profile, which forms the basis of absorbed heat calculations in the present work, but after speaking with engineers at Goddard Space Flight Center working on the Geostationary Transfer Orbit Satellite (GTOSat), it was clear that passive deorbit concerns, specifically the inability to predict a probability of staying in orbit more than 6 months and less than 25 years over 90%, would preclude manifesting a spacecraft with a similar orbital profile on a commercial launch vehicle through the CubeSat Launch Initiative (CSLI). ABEX has since de-scoped to a strictly LEO mission and altered the science payload to compensate for increased sensor noise. The mission architecture is graphically illustrated in Figure 2, and thermal modeling results shown in the present work are specific to a Sun-Synchronous Orbit (SSO) in which early mission phases do not experience eclipse. Mission life is estimated at 12 months; thermal results are provided for phase 1 notionally in 2025.

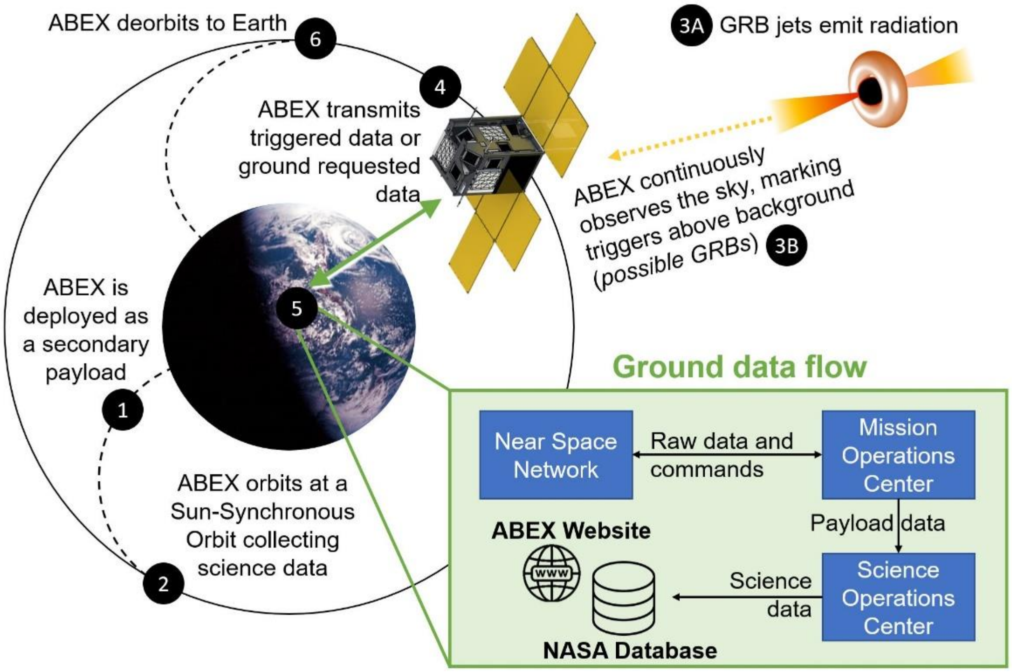

Figure 2: ABEX Mission Architecture

EXISTING SPACECRAFT THERMAL MODELS

ABEX is, of course, not the first program to perform thermal modeling and publish the methodology publicly or commercially. Various MATLAB thermal modeling codes include the Princeton Satellite Systems CubeSat Toolbox (PSSCT)[19], SatTherm[20], and the Adaptive Thermal Modeling Tool (ATMT)[21]. STK can also perform thermal modeling, but MATLAB model validation is more rigorously performed in Thermal Desktop from Cullimore & Ring Technologies, SINDA/FLUINT from MSC Software, or Thermica from Airbus.

MATLAB Thermal Modeling Tools

PSSCT is a thorough analysis tool that can be used for a variety of satellite design elements, but it is a first order analysis tool. The PSSCT Isothermal CubeSat Model utilizes a set data structure to manage variables and external conditions, and thus many thermal environment parameters are restricted to be constants. In his characterization of ATMT[21], Anger notes,

> The main limitation of the [PSSCT] is that the thermal algorithms are meant to provide a first order estimate of the temperatures. Smallsat thermal modeling beyond the concept study phase requires the ability to easily increase the number of components while modifying their geometry, spatial location, and operating parameters through time.

The results of PSSCT would not receive a high uncertainty characterization ranking in NASA-STD-7009A because it makes too many assumptions that simplify the solution and lacks flexibility in execution. The derivation of absorbed heat on a given spacecraft face or from a certain heat source is not possible in PSSCT, and projected areas receiving incident heat flux are not functions of vectors normal to spacecraft faces representing spacecraft attitude. In the ABEX methodology, absorbed heat is calculated on all faces from all sources individually as a function of projected area and spacecraft attitude and summed for a total absorbed heat value with all constituent results made available. Additionally, surface absorptivity is a function of both wavelength and AOI in both the ABEX MATLAB and Simulink models when calculated from complex, spectral refractive indices.

ATMT is a robust toolbox for spacecraft thermal analysis that includes an input solver that iteratively evaluates spacecraft attitude changes, a resistive network of spacecraft components, and internal conduction for a non-isothermal spacecraft model. However, ATMT requires a foundation of object-oriented components input through a custom geometry module, which makes ATMT less user-friendly than importing a Standard for the Exchange of Product Data (STEP) file as is possible in Thermal Desktop. ATMT, like PSSCT, assumes diffuse-grey surfaces wherein spectral emissivity and absorptivity are independent of wavelength and direction, a limitation the ABEX methodology overcomes.

Cumulatively, the weaknesses of PSSTC, SatTherm, and ATMT can be described as a lack of holistic perspective and failure of identity establishment resulting in the absence of widespread adoption. Engineers do not ubiquitously use these tools because the tools execute a broad aspect of M&S V&V efforts weakly instead of executing a narrow aspect of M&S V&V efforts strongly. ATMT is useful and customizable, but in its pursuit of becoming a stronger MATLAB thermal model it becomes a weaker three-dimensional thermal solver. A MATLAB thermal model should have strictly defined TPMs as outputs, and those outputs should have explicitly defined V&V factor rankings akin to those proposed by NASA-STD-7009A with the intent for rankings to increase or remain constant in subsequent thermal models. MATLAB thermal models do not replace three-dimensional models, but three-dimensional modeling efforts without first order validation may waste valuable modeling time due to a lack of an informed baseline. The solution to weak MATLAB models, meaning overly simplistic, difficult to use, or requiring custom Application Programming Interfaces, is a planned understanding of their purpose and limitations with clear transferals of TPM authority to higher-fidelity models for TPM uncertainty reduction.

Multi-Tool Thermal Modeling Efforts

Many characterizations of spacecraft modeling in Thermal Desktop exist[22,23,24], including the inability of Thermal Desktop to model complex geometries effectively. Multi-tool thermal modeling efforts are those that utilize lower-fidelity models to inform higher-fidelity models, such as the use of isothermal MATLAB models to inform Thermal Desktop models. Kovác and Józsa analyzed the SMOG-1 PocketQube using both a resistive network and finite element model created in ANSYS Workbench[25]. Although applying significant thermal assumptions such as Kirchoff's so-called law of thermal radiation, which is not a valid assumption in space conditions with low thermal time constants, this approach represents an accessible SoM archetype usable by university programs. Reyes et al. created a low-complexity SoM including MATLAB, Simulink, and Thermal Desktop models, the same modeling environments as the present work, to inform coating selection of the 1U CIIIASat with viable results[26]. An excellent characterization of multi-model synthesis in thermal analysis was performed by Stohlman of NASA Langley for radiative thermal and nonlinear stress analysis regarding the Near Earth Asteroid Scout (NEA Scout) mission[27]. Modeling environment interaction figures were presented, but the work stops short of describing a planned SoM existing before the work began, a missing link the present work provides.

ABEX THERMAL MODELING METHODOLOGY

The ABEX SoM methodology is predicated on engineers understanding the limitations of each modeling environment. No model attempts to be all-encompassing, and models of successively higher fidelity validate lower-fidelity models. SysML is used as a central source of truth for spacecraft component properties where lower-fidelity models are not providing thermal component properties or pre-calculated boundary conditions. The full organization of the thermal SoM is graphically represented in Figures 3-5. Temporal dependence is not illustrated in parameter variable nomenclature, but all dynamic variables are calculated as a function of time.

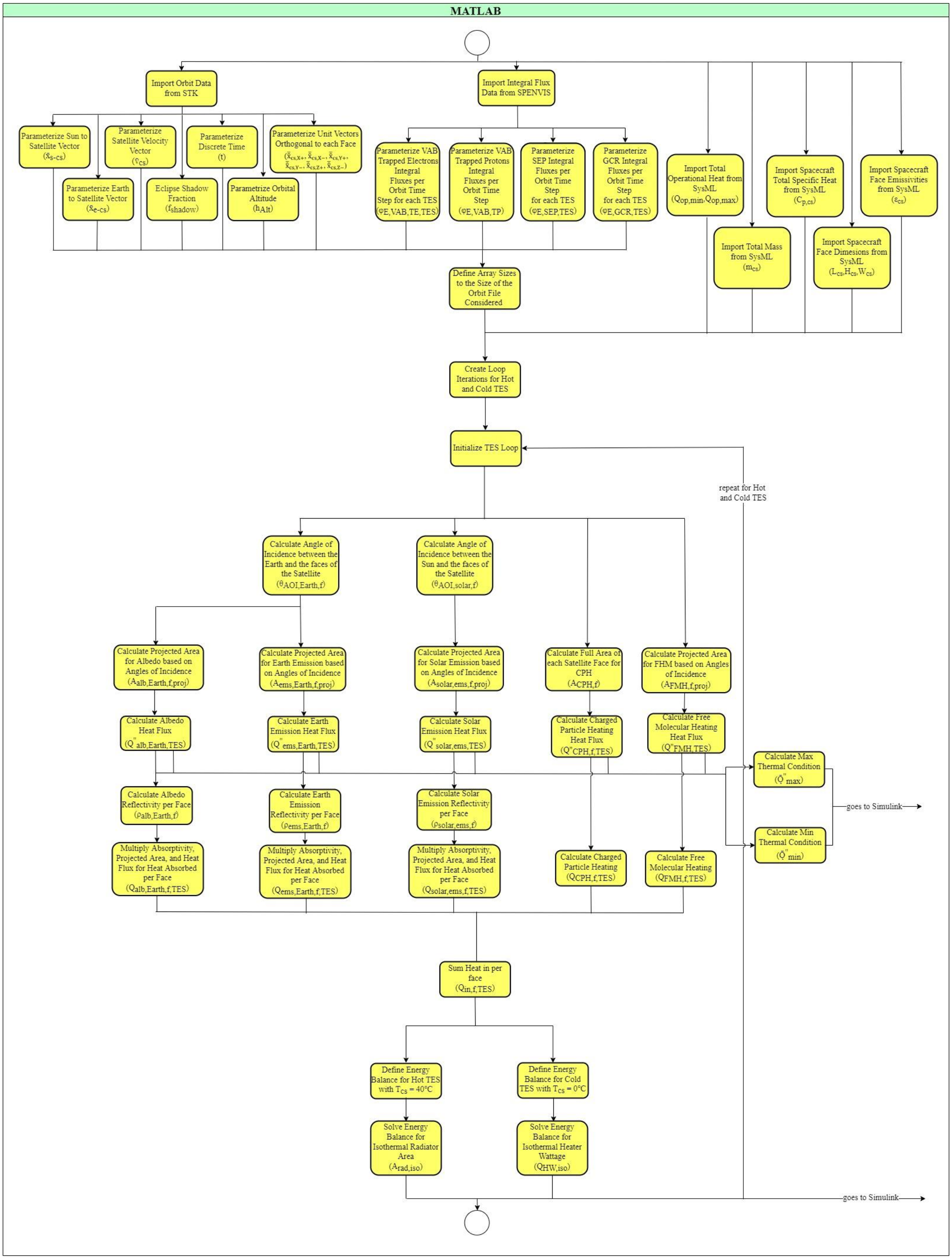

Figure 3: ABEX Thermal System of Models Methodology for MATLAB

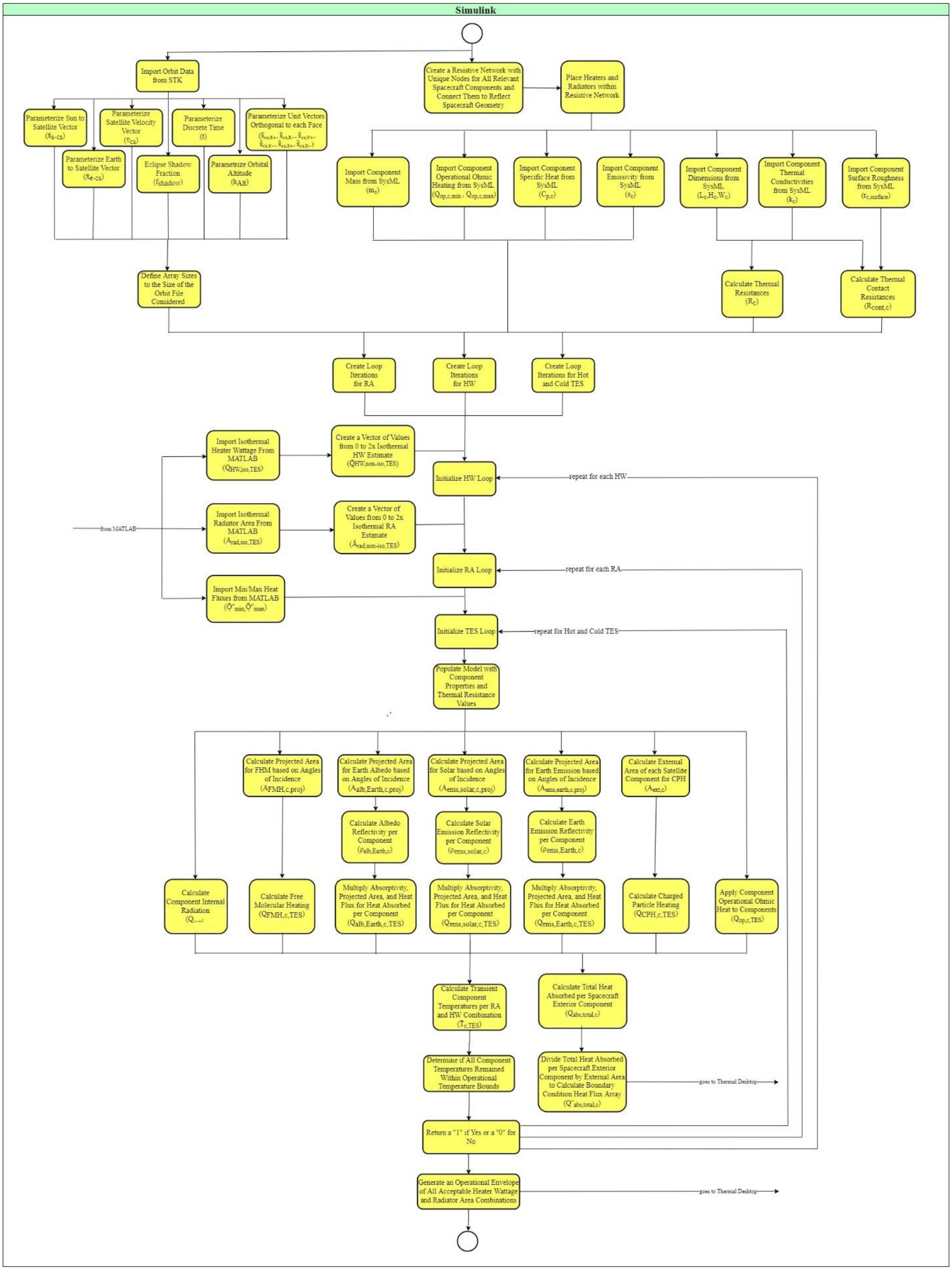

Figure 4: ABEX Thermal System of Models Methodology for Simulink

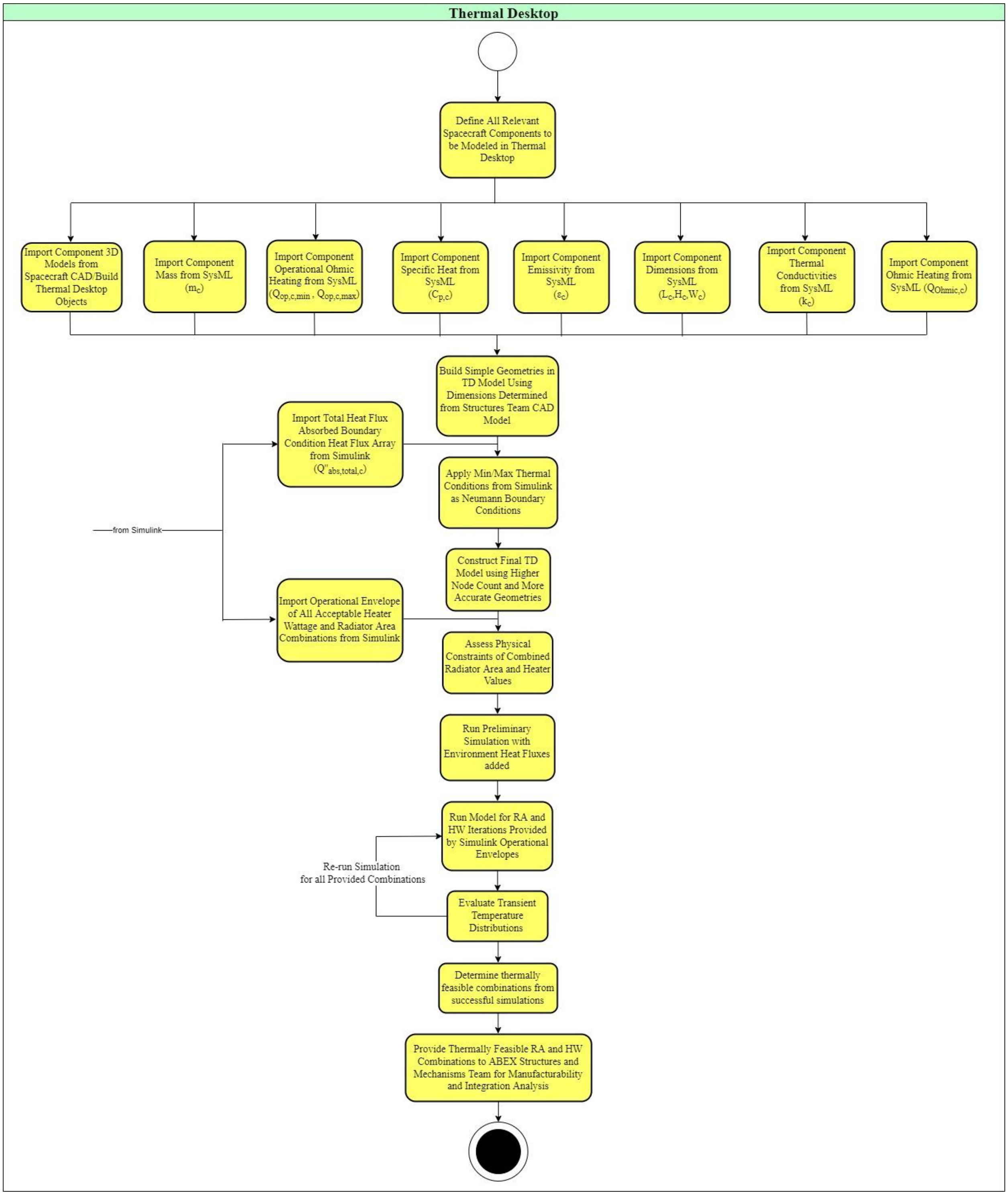

Figure 5: ABEX Thermal System of Models Methodology for Thermal Desktop

A common thermal analysis technique is to evaluate both the first thermal environment the spacecraft will encounter, usually described as Beginning of Life (BOL), and the last thermal environment the spacecraft will encounter, usually described as End of Life (EOL). The difference between BOL and EOL is commonly represented as a degradation in optical properties and component efficiencies. BOL and EOL conditions are evaluated in the most benign thermal environment and the most difficult thermal environments, both hot and cold. This approach is termed, “Best, Worst, First, Last[10].” As a systems engineering principle, the ABEX program makes a definitive distinction between modes and states. To ABEX, a mode is an abstract configuration, condition, or process that occurs with or without a corresponding physical state in a component, subsystem, or system at a given time. A mode is a non-tangible, non-physical concept. A state is defined as a physical mechanical configuration, environmental condition, operational condition, or other physical condition that either happens to or is initiated by a component, subsystem, or system at a given time. For ABEX thermal analysis, the concept of TES is utilized to describe hottest and coldest thermal conditions; mathematical differences between the hot and cold TES are described in Halvorson et al.[18].using the phrases “hot case” and “cold case.” A hot TES features maximum possible heat fluxes, highest confidence particle flux distributions, and ohmic heating during the most power-intensive spacecraft operational mode, the data collection mode. A cold TES commensurately features minimum possible heat fluxes, lowest confidence particle flux distributions, and ohmic heating during the least power-intensive spacecraft operational mode, the idle mode. Parameters that vary with TES are provided the subscript TES.

Systems Tool Kit

The STK model sources orbital element data from SysML to create an orbital profile including spacecraft state vectors and attitude profiles during communication. STK outputs transient state vectors, unit vectors orthogonal to individual spacecraft faces, and spacecraft positional information to a .xlsx file which is read by the MATLAB model. STK can perform other analyses for communications, deorbit profiles, or conjunction, but the purpose of STK here is to produce a .xlsx file of transient orbital parameter data. A set of example STK outputs for a single timestep is provided in Table 1.

Space Environment Information System

Space is radioactive, and particle sources of interest are Solar Energetic Particles (SEP), Galactic Cosmic Rays (GCR), and both protons and electrons trapped in the VAB. SEPs modeled in SPENVIS utilize the methodology and context of Halvorson et al.[18] to evaluate particle integral fluxes from all SRE sources. A more complete characterization of the Earth SRE is provided by Nöldeke[8]. The SPENVIS integral flux data generation process is described in Figure 6. Confidence intervals in particle flux distributions are used to vary TES conditions, with cold TES evaluated at mean integral flux levels and hot TES evaluated at +2σ above mean. SPENVIS models AE8 for VAB trapped electrons, SAPPHIRE for SEP protons, and International Organization for Standardization (ISO) 15390 for GCR ions feature the ability vary confidence intervals, but AP8 for VAB trapped protons does not have that ability. SPENVIS requires a specific file format for orbital data importing from STK, so a text file in the required format must be generated prior to simulation in SPENVIS.

Table 1: Example Output Data from STK. Coordinates are in the Earth-Centered Inertial Frame

Month	**Day**	**Hour**	**Minute**	**Seconds**
8	1	18	5	0
Earth to Satellite Vector, $\bar{x}_{e-cs}$	**x [km]**	**y [km]**	**z [km]**	
	5246.823431	4045.226004	2191.20662	
Sun To Satellite Vector, $\bar{x}_{s-cs}$	**x [km]**	**y [km]**	**z [km]**	**Velocity [km/sec] ,** v_{cs}
	-96340786.2	107657752.5	46671603.71	7.557865
Faces of Sat	**x-direction**	**y-direction**	**z-direction**	**Latitude [deg]**
X+, $\hat{\bar{x}}_{cs,X+}$	-0.204981	0.229063	-0.951584	18.521
X-, $\hat{\bar{x}}_{cs,X-}$	0.204981	-0.229063	0.951584	**Longitude [deg]**
Y+, $\hat{\bar{x}}_{cs,Y+}$	-0.745194	-0.666848	0	176.138
Y-, $\hat{\bar{x}}_{cs,Y-}$	0.745194	0.666848	0	**Altitude [km],** h_{alt}
Z+, $\hat{\bar{x}}_{cs,Z+}$	-0.634562	0.709115	0.307387	602.142683
Z-, $\hat{\bar{x}}_{cs,Z-}$	0.634562	-0.709115	-0.307387	

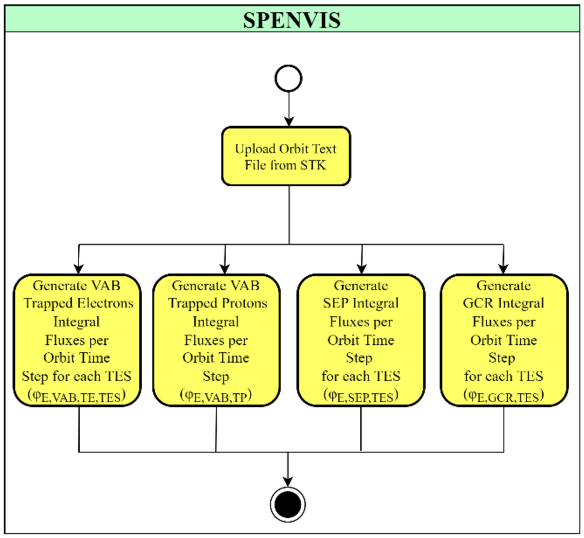

Figure 6: SPENVIS Data Generation Process

While it is shown that CPH has a functionality negligible effect on thermal TPMs, the same particle fluxes from SPENVIS used to calculate CPH can also be used to evaluate Total Ionizing Dose, Non-Ionizing Energy Loss (NIEL), surface charging, deep dielectric charging, and Single Event Effect (SEE) rates, which are outside the scope of this work.

MATLAB

As shown in Figure 3, the MATLAB model imports data from STK, SPENVIS, and SysML, simulates the ABEX thermal environment for the mission life, converts incident heat fluxes to absorbed heat, converts energy deposited by particles into CPH and FMH, and provides a baseline RA and HW calculation to Simulink as the maximum required values calculated over the entire simulation. To convert incident heat fluxes to absorbed heat, the MATLAB model calculates specular, spectral absorptivities for each surface material using either vendor-provided spectral reflectivity profiles or specular, spectral Fresnel relationships for opaque surfaces, but the heat absorbed calculation is per spacecraft face. The Simulink model performs the same absorbed heat calculations as the MATLAB model, but absorbed heats are calculated per externally-facing component instead of per face resulting in location-specific temperature changes for components with differing surface optical properties. A brief summary of thermal sources in space is warranted.

There are three primary sources of electromagnetic radiation for LEO spacecraft dependent on TES: radiative solar emission $Q''_{ems,solar,TES}$, radiative Earth emission $Q''_{ems,Earth,TES}$, and Earth albedo $Q''_{alb,Earth,TES}$. Radiative lunar emission is not considered here. Solar emission is the largest thermal contributor followed by Earth albedo and then Earth emission, though albedo and emission are highly dependent on latitude, altitude, and solar zenith angle ξ. CPH occurs as particle radiation encounters the spacecraft structural elements and deposits energy, and FMH occurs as the spacecraft encounters atmospheric particles as it orbits the Earth. Calculation of heat flux values for all sources are detailed in Halvorson et al.[18]. Heat is generated within the spacecraft due to ohmic heating when spacecraft components are operated, and, if needed, heat is intentionally provided to the system through dedicated resistive heaters to maintain component temperatures above lower temperature bounds. Heat sources are depicted on ABEX in Figure 7.

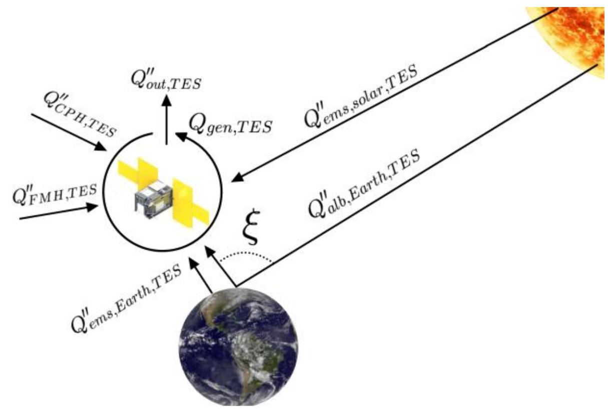

Figure 7: Heat Sources in Space

With these sources in mind, the spacecraft thermal analysis starts with an energy balance in Eq. (1). In this equation, $\dot{E}_{in}$ represents the rate of energy entering the system, $\dot{E}_{gen}$ represents the rate of energy generated within the system, $\dot{E}_{out}$ represents the rate of energy leaving the system, and $\frac{dE_{sys}}{dt}$ represents the change in system energy over time, all in units of W.

$$\Sigma\dot{E}_{in} + \Sigma\dot{E}_{gen} - \dot{E}_{out} = \frac{dE_{sys}}{dt} \qquad (1)$$

The heat entering the system per face per TES from external sources is comprised of absorbed heat from solar emission $Q_{ems,solar,f,TES}$, Earth emission $Q_{ems,Earth,f,TES}$, Earth albedo $Q_{alb,Earth,f,TES}$, FMH $Q_{FMH,f,TES}$, and CPH $Q_{CPH,f,TES}$ represented in Eq. (2) in units of W.

$$\dot{E}_{in} = Q_{in,f,TES} = Q_{ems,solar,f,TES} + Q_{ems,Earth,f,TES} + Q_{alb,Earth,f,TES} + Q_{FMH,f,TES} + Q_{CPH,f,TES} \qquad (2)$$

Absorbed heat from solar emission on a given face $Q_{ems,solar,f,TES}$ is calculated as the product of solar emission heat flux $Q''_{ems,solar,TES}$, projected area of the face solar emission is incident on $A_{ems,solar,f,proj}$, one less the shadow fraction f_{shadow} and spectral, specular, electromagnetic absorptivity of solar radiation on the face $\alpha_{ems,solar,f}$. If solar arrays are converting energy to electricity, the total absorbed energy rate is divided into heat and power wherein $Q_{ems,solar,f,total,TES}$ represents the total absorbed energy rate from solar radiation per face, $Q_{ems,solar,f,TES}$ is the heat absorbed from solar radiation, and $P_{solar,f,TES}$ is the absorbed power in the solar panels. All absorbed heat and power is dependent on TES; $Q_{ems,solar,f,total,TES}$ is equal to $Q_{ems,solar,f,TES}$ if no solar panels are present on a face or panels are not generating electricity. If solar power conversion is present, $P_{solar,f,TES}$ is calculated by Eq. (3) with η_{cell} representing photovoltaic cell efficiency.

$$P_{solar,f,TES} = Q''_{ems,solar,TES} \cdot A_{ems,solar,f,proj} \cdot \alpha_{ems.solar,f} \cdot \eta_{cell} \cdot (1 - f_{shadow}) \tag{3}$$

Eq. (3) assumes the entire face is comprised of solar cells, which may or may not be true per spacecraft. Separating the area term into multiple terms with and without cells clears assumption-based errors. Absorbed heat from solar emission on a power-generating surface is therefore calculated by Eq. (4).

$$Q_{ems,solar,f,TES} = Q''_{ems,solar,TES} \cdot A_{ems,solar,f,proj} \cdot \alpha_{ems,solar,f} \cdot (1 - \eta_{cell}) \cdot (1 - f_{shadow}) \tag{4}$$

For cold TES, the spacecraft is often assumed to be in eclipse conditions without photovoltaic power generation. For hot TES, the spacecraft batteries are often assumed to be fully charged, so arrays are assumed to not generate electricity. Eqs. (3-4) therefore become formalities in practice. The AOI of incident radiation is calculated in Eq. (5) per face wherein $\theta_{AOI,f}$ represents the angle of incident radiation, $\bar{\hat{x}}_{cs,f}$ represents the unit vector normal to a given satellite face exemplified in Table 1, and $\bar{\hat{x}}_{incident}$ represents the unit vector of the incident radiation. AOI for Earth emission and albedo are considered equal, which represents a source of error.

$$\theta_{AOI,f} = \cos^{-1}\left(\frac{\bar{\hat{x}}_{cs,f} \cdot \bar{\hat{x}}_{incident}}{\left|\left|\bar{\hat{x}}_{cs,f}\right|\right| \cdot \left|\left|\bar{\hat{x}}_{incident}\right|\right|}\right) \tag{5}$$

The projected area is calculated in Eq. (6) as the product of the area of the spacecraft face, represented dimensionally in terms of face length and width by $L_f \cdot w_f$, multiplied by the cosine of the AOI.

$$A_{ems,solar,f,proj} = \left(L_f \cdot W_f\right) \cdot cos\left(\theta_{AOI,ems,solar,f}\right) \tag{6}$$

Face or component reflectivity as a function of wavelength and AOI can be calculated by Fresnel relations or the integration of vendor-provided spectral reflectivities; both are outside the scope of this work but are discussed in in section 3.3.5.1 of Halvorson[13]. Absorptivity is substantially different for metals, flat absorbers, or solar absorbers, but generally absorptivity increases with AOI to a maximum and then decreases to zero when parallel with the surface. Because absorptivity is a function of wavelength, absorptivities for solar and Earth emission are dissimilar. Earth albedo is assumed to have the same wavelengths as solar emission, though practically some energy would be lost during the reflection of solar radiation off the Earth atmosphere. Following the calculation trends of Eqs. (4-6) for solar emission, absorbed heat from radiative Earth emission per spacecraft face is calculated without power generation considerations in Eq. (7) as the product of Earth emission heat flux $Q''_{ems,Earth,f,TES}$, projected area of the face Earth emission is incident on $A_{ems,Earth,f,proj}$, and spectral, specular, electromagnetic absorptivity of infrared Earth radiation on the face $\alpha_{ems,Earth,f}$.

$$Q_{ems,Earth,f,TES} = Q''_{ems,Earth,f,TES} \cdot A_{ems,Earth,f,proj} \cdot \alpha_{ems,Earth,f} \tag{7}$$

Absorbed heat from the Earth albedo is calculated similarly in Eq. (8) as the addition product of the cosine of the solar zenith angle ξ. AOI calculations for both Earth emission and Earth albedo follow the format of Eq. (5) with new $\bar{\hat{x}}_{incident}$ values provided by STK; calculations for projected area per face follow the format of Eq. (6) with the resulting $\theta_{AOI,f}$ from Eq. (5).

$$Q_{alb,Earth,f,TES} = Q''_{alb,Earth,f,TES} \cdot A_{alb,Earth,f,proj} \cdot \alpha_{alb,Earth,f} \cdot cos(\xi) \tag{8}$$

Calculation of solar zenith angle ξ is provided by Eq. (9) wherein $\bar{x}_{e-cs}$ represents the vector from the Earth to the satellite and $\bar{x}_{e-s}$ represents the vector from the Earth to the Sun.

$$\xi = cos^{-1}\left(\frac{\bar{x}_{e-cs} \cdot \bar{x}_{e-s}}{||\bar{x}_{e-cs}|| \cdot ||\bar{x}_{e-s}||}\right) \tag{9}$$

Absorbed FMH is calculated in Eq. (10) by multiplying the FMH flux by the projected area orthogonal to the velocity vector; conversion of CPH flux into absorbed heat is calculated similarly in Eq. (11) except the area parameter is simply the area of the face because CPH is considered isotropic. FMH and CPH heat flux calculation is detailed in Halvorson et al.[18].

$$Q_{FMH,f,TES} = Q''_{FMH,TES} \cdot A_{FMH,f,proj} \tag{10}$$

$$Q_{CPH,f,TES} = Q''_{CPH,TES} \cdot A_{CPH,f} \tag{11}$$

Internal heat generation consists of both ohmic heating from nominal component operation and intentional heating from heater operation $Q_{HW,TES}$. Operational ohmic heating includes heat dissipated by the voltage converters of the Electrical Power System (EPS) $Q_{dis,EPS,TES}$, heat dissipated by all components due to electrical operation $Q_{dis,c,total,TES}$, and heat dissipated by the battery due to charging and discharging efficiencies $Q_{dis,batt,TES}$. Total heat generation is described in terms of Eq. (1) in Eq. (12).

$$\Sigma\dot{E}_{gen} = Q_{gen,TES} = Q_{dis,EPS,TES} + Q_{dis,c,total,TES} + Q_{dis,battery,TES} + Q_{HW,TES} \tag{12}$$

Cumulative component power consumption $P_{cons,c,total,TES}$ is determined in Eq. (13) as the summation of power consumption by each individual component $P_{cons,c,TES}$. A hot TES for component power consumption occurs during the software mode consuming the most power, and the low TES occurs during a low power consumption mode such as idle.

$$P_{cons,c,total,TES} = \sum_{c=1}^{c_{max}} P_{cons,c,TES} \tag{13}$$

The power supplied by the EPS $P_{supp,EPS,TES}$ is calculated by Eq. (14) as the total power consumed by all spacecraft components except for the EPS divided by the EPS efficiency $\eta_{EPS,TES}$. EPS efficiency is a product of path-dependent EPS component efficiencies such as voltage converters and maximum power point trackers and can be assumed to vary by TES due to converter temperature dependencies.

$$P_{supp,EPS,TES} = \frac{P_{cons,c,total,TES}}{\eta_{EPS,TES}} \tag{14}$$

Heat dissipated by the EPS is calculated by Eq. (15) as the power supplied by the EPS less the total power consumed by spacecraft components; it represents ohmic heating due to EPS power conversion efficiency.

$$Q_{dis,EPS,TES} = P_{supp,EPS,TES} - P_{cons,c,total,TES} \tag{15}$$

The heat dissipated by all components $Q_{dis,\mathrm{c,total,TES}}$ is a function of power consumption by each component $P_{cons,c,TES}$ and the power conversion efficiency of each component η_c.

$$Q_{dis,\mathrm{c,total,TES}} = \sum_{c=1}^{c_{max}} P_{cons,c,TES} \cdot (1 - \eta_c) \tag{16}$$

The generated power of the solar array $P_{gen,\mathrm{power},TES}$ is calculated in Eq. (17) as the product of the solar power converted to electricity $P_{solar,f,TES}$, the EPS diode and line loss efficiency $\eta_{diode+line,TES}$, the power distribution efficiency η_{dist}, and the solar cell efficiency dependence on temperature $\eta_{temp,TES}$. Additional margins can be applied here if desired for power modeling purposes.

$$P_{gen,power,TES} = P_{solarf,TES} \cdot \eta_{temp,TES} \cdot \eta_{diode+line,TES} \cdot \eta_{dist} \tag{17}$$

Net spacecraft power $Q_{net,TES}$ is a function of the power generated from the solar array $Q_{gen,\mathrm{power},TES}$ and power supplied by the EPS $Q_{supp,EPS,TES}$ in Eq. (18).

$$P_{net,TES} = P_{gen,\mathrm{power},TES} - P_{supp,EPS,TES} \tag{18}$$

The heat dissipated by the battery $Q_{dis,battery,TES}$ is calculated in Eq. (19) as the product of net spacecraft power $P_{net,TES}$ and the quantity one less the battery power conversion efficiency $\eta_{battery}$. Battery conversion efficiency was set to 0.95 during battery charging, or when $P_{net,TES}$ is positive, and 0.91 during battery discharging, or when $P_{net,TES}$ is negative, based on unpublished discussion conclusions with thermal engineers at Marshall Space Flight Center.

$$Q_{dis,battery,TES} = P_{net,TES} \cdot (1 - \eta_{battery}) \tag{19}$$

$$\begin{cases} if\ P_{net,TES} > 1, & \eta_{battery} = 0.95 \\ if\ P_{net,TES} < 1, & \eta_{battery} = 0.91 \end{cases}$$

Thus, Eq. (12) can be rewritten as Eq. (20).

$$Q_{gen,TES} = \frac{P_{cons,c,total,TES}}{\eta_{EPS,TES}} - P_{cons,c,total,TES} + \sum_{c=1}^{c_{max}} P_{cons,c,TES} \cdot (1 - \eta_c) + (P_{gen,\mathrm{power},TES} - P_{supp,EPS,TES}) \cdot (1 - \eta_{battery}) + Q_{HW,TES} \tag{20}$$

The transient energy term $\frac{dE_{sys}}{dt}$ in Eq. (2) can be written for an isothermal CubeSat as Eq. (21) with spacecraft mass m_{cs} and specific heat $c_{p,cs}$ included.

$$\frac{dE_{sys}}{dt} = m_{cs} \cdot c_{p,cs} \cdot \frac{dT_{cs,TES}}{dt} \tag{21}$$

In the cold TES, spacecraft temperature $T_{cs,TES}$ is set equal to 0°C, and an energy balance is established with an assumed radiator area to calculate $Q_{HW,TES}$. In the hot TES, $T_{cs,TES}$ is set equal to 40°C, and $Q_{HW,TES}$ is set equal to 0 W to calculate a radiator area. Because the temperature does not change, Eq. (21) is always equal to zero. The rate of energy leaving the spacecraft per face $\dot{E}_{out,f}$ begins with Eq. (22) wherein A_f is the total external area of a satellite face, σ_{sbc} is the Stefan-Boltzmann constant, ε_f is the emissivity of a given face, and T_{surr} is 2.7 K, the temperature of empty space[28].

$$\dot{E}_{out,f} = Q_{out,f,TES} = A_f \cdot \sigma_{sbc} \cdot \varepsilon_f \cdot (T_{CS,TES} - T_{surr}) \tag{22}$$

To solve for radiator area, Eq. (22) can be broken into multiple terms representing distinct areas with distinct emissivities, one for radiator area per face $A_{rad,f}$ and one for all other external area per face $A_{no\,rad,f}$, such that the face area is equal to the area without a radiator and an area with a radiator. The radiator is not assumed deployable here. Just as not all faces have solar cells necessitating Eqs. (3-4), not all faces will have radiators.

$$A_f = A_{rad,f} + A_{no\ rad,f} \tag{23}$$

A final consideration for isothermal energy balances is that deployable solar arrays would be considered the same temperature as the spacecraft in all TES conditions. Because MATLAB values are provided to both Simulink and Thermal Desktop for further, higher-order analysis, it is recommended to remove solar array area $A_{array,f}$ from the energy out term, specifically from $A_{no\ rad,f}$, as shown in Eq. (24). In the Simulink methodology described in the following section, an array from zero to twice the isothermal RA or HW result is evaluated in Simulink. If this brute-force method was replaced with a root-finding method, excessively large isothermal RA or HW results caused by isothermal, deployable solar arrays would not result in Simulink analysis difficulties.

$$Q_{out,f,TES} = \left[A_{rad,f} \cdot \varepsilon_{rad} + \left(A_{no\ rad,f} \cdot \varepsilon_{no\ rad} - A_{array,f} \cdot \varepsilon_{array}\right)\right] \cdot \sigma_{sbc} \cdot \left(T_{CS,TES} - T_{surr}\right) \tag{24}$$

Examples of mathematical inputs corresponding to the same time step as Table 1 are provided in Tables 2 and 3.

Table 2: Face-Dependent Parameter Values for the First Timestep

Parameter	+Z face	-Z face	+Y face	-Y face	+X face	-X face
$\alpha_{ems.solar,f}$ [-]	0.81518	0.1764	0.14465	0.14465	0.14112	0.14112
$\alpha_{ems,Earth,f}$ [-]	0.83335	0.1764	0.35593	0.35260	0.35260	0.35260
$\alpha_{alb,Earth,f}$ [-]	0.7879	0.1785	0.16891	0.14465	0.14465	0.14465
$\theta_{AOI,ems,solar,f}$ [°]	0	180	90	90	90	90
$\theta_{AOI,ems,Earth,f}$ [°]	162.11	17.89	72.11	107.88	90	90
$\theta_{AOI,alb,Earth,f}$ [°]	162.11	17.89	72.11	107.88	90	90
$A_{ems,solar,f,proj}$ [m^2]	0.5077	0	0	0	0	0
$A_{ems,Earth,f,proj}$ [m^2]	0	0.0294	0.0119	0	0	0
$A_{alb,Earth,f,proj}$ [m^2]	0	0.0294	0.0119	0	0	0
ε_f [-]	0.7756	0.7756	0.1976	0.1976	0.039	0.039

Table 3: TES-Dependent Parameter Values, Cold TES

Parameter	Value	Unit
$Q''_{ems,solar,TES}$	1322	W/m^2
$Q''_{ems,Earth,TES}$	184.59	W/ m^2
$Q''_{alb,Earth,TES}$	312.86	W/ m^2
$Q_{cons,c,total,TES}$	23.556	W
$\eta_{EPS,TES}$	0.75	-
$Q_{dis,EPS,TES}$	7.852	W
$Q_{dis,c,\mathrm{total,TES}}$	3.648	W
m_{cs}	25 kg	kg
$c_{p,cs}$	887	J/kg-K
$A_{rad,f}$	0.02	m^2

It is clear from the preceding equations that energy entering and leaving the system must be organized per face for a given TES and summed for a holistic isothermal spacecraft energy balance. All calculated values are calculated per timestep, and maximum values of isothermal HW and RA are provided to Simulink for non-isothermal analysis. Eq. (25) is offered for the calculation of HW with an assumed spacecraft temperature and radiator area, but spacecraft-specific considerations for solar array, shadow fraction, and radiator placement must be implemented for accurate calculation.

$$Q_{HW,TES} = \sum_{f=1}^{6} [Q''_{ems,solar,TES} \cdot (L_f \cdot W_f) \cdot \frac{\bar{\hat{x}}_{cs,f} \cdot \bar{\hat{x}}_{ems,solar}}{\left|\left|\bar{\hat{x}}_{cs,f}\right|\right| \cdot \left|\left|\bar{\hat{x}}_{ems,solar}\right|\right|} \cdot \alpha_{ems.solar,f} + Q''_{ems,Earth,TES} \cdot (L_f \cdot W_f) \cdot \frac{\bar{\hat{x}}_{cs,f} \cdot \bar{\hat{x}}_{ems,Earth}}{\left|\left|\bar{\hat{x}}_{cs,f}\right|\right| \cdot \left|\left|\bar{\hat{x}}_{ems,Earth}\right|\right|} \cdot \alpha_{ems.Earth,f} + Q''_{alb,Earth,TES} \cdot (L_f \cdot W_f) \cdot \frac{\bar{\hat{x}}_{cs,f} \cdot \bar{\hat{x}}_{ems,Earth}}{\left|\left|\bar{\hat{x}}_{cs,f}\right|\right| \cdot \left|\left|\bar{\hat{x}}_{ems,Earth}\right|\right|} \cdot \alpha_{alb.Earth,f} \cdot \frac{\bar{x}_{e-cs} \cdot \bar{x}_{e-s}}{||\bar{x}_{e-cs}|| \cdot ||\bar{x}_{e-s}||} + Q''_{FMH,TES} \cdot A_{FMH,f,proj}$$

$$+Q^{"}_{CPH,TES} \cdot A_{CPH,f}] - [\frac{P_{cons,c,total,TES}}{\eta_{EPS,TES}} - P_{cons,c,total,TES}$$

$$+ \sum_{c=1}^{c_{max}} P_{cons,c,TES} \cdot (1-\eta_c) + (P_{gen,\text{power},TES} - P_{supp,EPS,TES})$$

$$\cdot (1-\eta_{battery})] - \sum_{f=1}^{6} \{[A_{rad,f} \cdot \varepsilon_{rad} +$$

$$(A_{no\ rad,f} \cdot \varepsilon_{no\ rad} - A_{array,f} \cdot \varepsilon_{array})] \cdot \sigma_{sbc} \cdot (T_{CS,TES} - T_{surr})\}$$

(25)

The energy balance to calculate radiator area is defined similarly to Eq. (25), but spacecraft-specific considerations for radiator placement will alter equations per face. The maximum calculated isothermal HW over all timesteps was ~47 W without solar array area included in the outbound radiation term. With array area included, required HW was 172 W, an unrealistic value.

Simulink

As shown in Figure 4, the Simulink model imports dimensional data, masses, specific heats, thermal conductivities, and emissivities from the SysML model, imports the same time-dependent STK data as the MATLAB model, calculates thermal resistances and contact resistances for the resistive network, imports from MATLAB the maximum and minimum thermal conditions, a characterization of hot and cold TES heat fluxes per face, and applies that data to components, or nodes, within a thermal resistive network. For external-facing components, spectral, specular absorptivities and projected areas are calculated per component for incident radiative fluxes, so each component has its own $\mathrm{Q_{in}}$ calculation. The Simulink model evaluates combinations of RA and HW in a brute-force approach wherein each combination is evaluated individually. Arrays are created from the MATLAB isothermal RA and HW values, and a matrix is formed from the product of those two arrays. The arrays for each range from zero to two times the isothermal value from MATLAB. The simulation is executed to calculate the temperature of each node, and each component node is prescribed boundary temperature conditions. An example of temperature boundary conditions is provided in Table 4. If the temperature of a given node exceeds the bounds ascribed to that node, the simulation fails. If no bounds are exceeded for any node, the simulation passes. Simulations that do not violate temperature bounds are considered part of the operational envelope and are viable options for the Thermal Desktop model to consider. A more intelligent method such as a root-finding method would start with the isothermal MATLAB values and iteratively approach an optimum set of RA and HW values; this is considered future work.

Table 4: Simulink Component Temperature Bounds

Component	Lower Bound, [°C]	Upper Bound, [°C]
Gamma Ray Detector	-20	80
X-Ray Detector	0	60
Li-Ion Battery	15	40
PCB	-20	85

It is important to note the results of the Simulink model are highly dependent on the number of heaters, heater locations, and the thermal conductivity between sensitive components and the radiators. Heaters also require control algorithms, which may be simple or highly complex. These design considerations must be included in the Simulink model and considered in a sensitivity analysis. If careful selection of heater number and wattage is not considered, the results may state no combination of HW and RA yield a viable solution for a given operational environment or TES. The combination matrix of RA and HW values must be evaluated for both the hot and cold TES individually and subsequently combined. The operational envelopes deemed acceptable for the cold TES are generally not the same as the envelopes deemed acceptable for the hot TES. Like a Venn diagram, the union of viable solutions for both the hottest and coldest operational envelopes is the operational envelope of RA and HW provided to the Thermal Desktop model for implementation evaluation.

Thermal Desktop

The Thermal Desktop model receives the operational envelope of viable HW and RA combinations from Simulink that provided sufficient thermal control to prevent component temperatures from exceeding their operational temperature bounds for both the hot and cold TES. Whereas it is the function of the Simulink model to evaluate heater placement, number, control algorithm, and maximum wattage per heater, it is the function of the Thermal Desktop model to evaluate the physical interfaces of the viable candidates. Heaters may be patch or cartridge type, and radiators may be offset from a given surface or deployable. There may need to be heat pipes or thermal straps between high-temperature components, such as solar array hinges or voltage converters, and the radiator, or the radiator may need to feature an embedded heat pipe. These are considerations under the purview of the Thermal Desktop model.

As depicted in Figure 5, modelers using Thermal Desktop import thermal properties such as conductivities, specific heats, and emissivities from SysML. Spacecraft CAD in .STEP format may be provided to Thermal Desktop modelers or modelers may create reduced-complexity geometries for components. The Neumann, constant flux boundary conditions provided from Simulink were created after calculating absorbed heat from all sources and dividing by external area. The spectral, specular absorptivities were considered in Simulink, and therefore the absorptivities relating to these provided heat flux boundary conditions are unity. Internal radiation still requires defined absorptivities per component. Modelers using thermal desktop can then begin assessment of physical RA and HW implementation. Once the physical implementation is characterized, a more realistic geometric mesh with a higher node count is generated, and specific heat and area values can be defined for the model. Modelers then simulate the model for the outer boundaries of the operational envelope of RA and HW combinations to determine three-dimensional consequences. A radiator area that is excessively large may result in isolated low-temperature regions, or a heater providing significant heat to a region with low thermal conductivity may overheat at steady-state. It is also in the Thermal Desktop model where BOL versus EOL considerations are evaluated, which may require altering provided Simulink values. A combination of HW and RA may provide sufficient thermal control at BOL but not EOL.

RESULTS

Results of the ABEX thermal SoM are not the final results for the mission itself; ABEX is still considered in early to moderate development. However, substantial results are available and reproducible for other missions using this SoM methodology. While SPENVIS, MATLAB, and Simulink results are robust, Thermal Desktop results require additional modeling work before publication can be considered accurate.

SPENVIS Results for MATLAB

SPENVIS can both characterize particle radiation environments and the effects particles have on spacecraft. Figure 8 depicts the ABEX altitude profile modeled in SPENVIS from STK inputs.

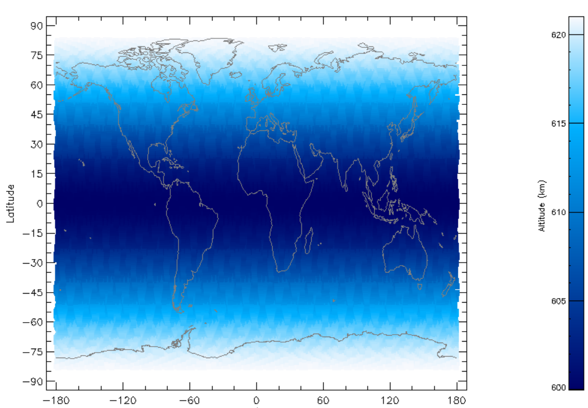

Figure 8: ABEX Altitude Profile

Using this altitude data, integral flux can be plotted for each time step, in this instance every 15 minutes, for any desired particle source or energy level. Figure 9 represents the trapped VAB electrons above 0.5 MeV. Visible is the South Atlantic Anomaly (SAA) and High Latitude Zones (HLZ) where aurorae are generated.

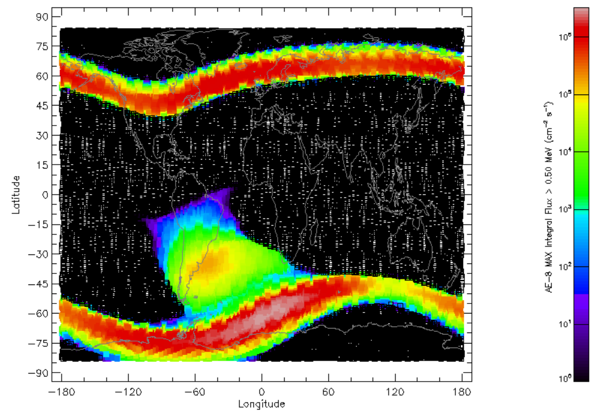

Figure 9: VAB Trapped Electrons Above 0.5 MeV

Figure 10 represents the trapped VAB protons above 1 MeV, which are all within the SAA. These are particles that would cause SEEs, and turning off components during SAA crossings is a viable radiation effect mitigation strategy due to the presence of these deleterious particles. SPENVIS can also generate SEP fluxes for quiet sun conditions and conditions corresponding to Coronal Mass Ejections (CME).

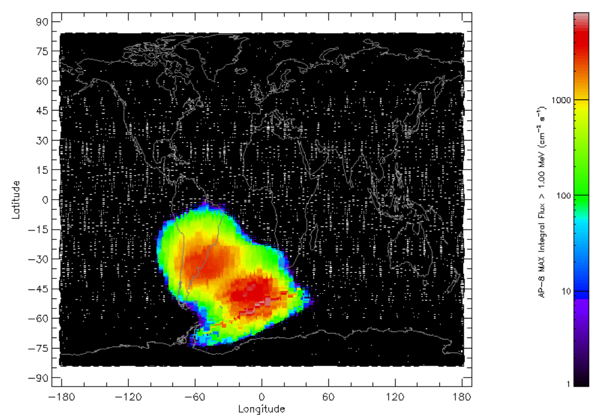

Figure 10: VAB Trapped Protons Above 1 MeV

MATLAB Results for Simulink and Thermal Desktop

Thermal data produced by the MATLAB model warrants an understanding of the spacecraft coordinate frame, depicted in Figure 11. During nominal operations, the solar array on the +Z face tracks the Sun and the +X face roughly tracks the Earth. The MATLAB model can produce absorbed heat value results per face and per heat source. Total heat absorbed on each face is shown in

Figure 12. This information serves as a first-order sanity check; the only difference between absorbed heat calculations in MATLAB and Simulink is that Simulink calculates absorbed heat per component and not per face. Because the non-isothermal Simulink model simulates a variety of radiator areas whereas the MATLAB model only simulates one assumed area during HW calculations, the absorptivity and emissivity of radiator-bearing faces will change in the SoM progression from MATLAB to Simulink.

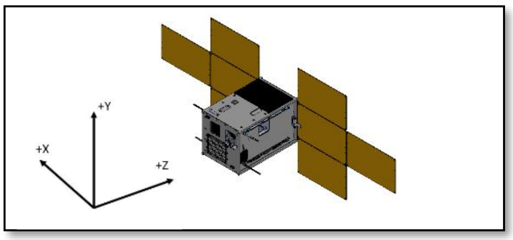

Figure 11: VAB Trapped Protons Above 1 MeV

Apparent in Figure 12 is the substantive difference between heat absorbed by the +Z face, the face with solar panels directed toward the Sun, and all other faces. As discussed, eclipse conditions should be considered the cold TES environment; this analysis is applied directly to the ABEX SSO for early mission operations wherein ABEX does not experience eclipse conditions. Calculated HW and RA for just the first phase is not sufficient for full-mission analysis, and the maximum HW and RA across the entire mission is the value that should be provided to Simulink. Additional results from the isothermal MATLAB model include power generation profiles and transient heat generated within the spacecraft from the EPS, components, batteries, and heaters. If power consumption values are set specific to spacecraft operation and not simply highest and lowest power consumption modes, accurate net power and battery capacity profiles can be modeled, though conditional statements for turning batteries and solar panels on and off when at capacity must be written.

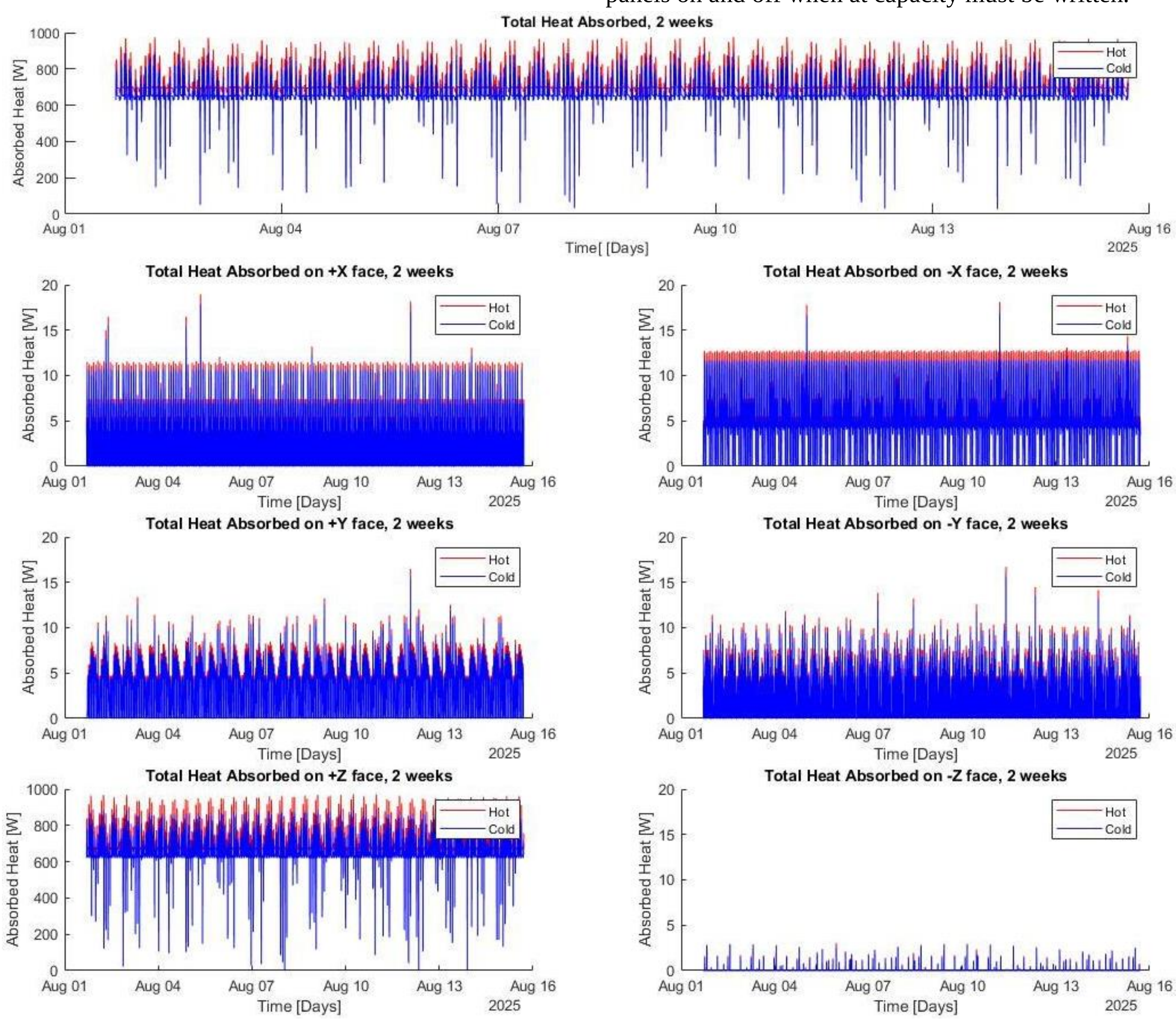

Figure 12: Total Heat Absorbed per Spacecraft Face, 2 Week Timeframe

Simulink Results for Thermal Desktop

Non-isothermal Simulink results characterize transient spacecraft temperatures for varied combinations of RA and maximum HW for both hot and cold TES. RA is considered static for each simulation, but HW for each heater is controlled by a Proportional-Integral-Derivative (PID) feedback controller with a thermal set point above the operational low temperature bound. The isothermal HW provided by the MATLAB model is divided by the number of heaters present in the spacecraft. For this ABEX thermal model, twelve heaters were considered on the detectors and other thermally sensitive electronics. An example temperature profile is provided in Figure 13 for the eight PCBs in the ABEX avionics stack during a hot TES evaluation with 5.49 W HW and 0.023 m^2 RA. Per Table 4, the upper temperature bound for PCBs is 85°C, which is unrealistically high for operational temperatures, but it is readily apparent that the temperatures in Figure 13 do not violate this temperature bound. PCB temperatures are unreasonably high indicating additional electronics thermal management design is warranted, but they are not out of limiting bounds. Table 5 is a snapshot of the operational envelope for the Simulink results wherein a zero signifies temperature bounds were violated and a one signifies temperature bounds were not violated. The bolded border between zeroes and ones delineates the combinations that did and did not violate temperature bounds for spacecraft components. The ones in Table 5 were acceptable combinations for both hot and cold TES, but the zeroes were only acceptable for the hot TES. The HW and RA combination in Figure 13 was non-viable because there was not sufficient HW to sustain component temperatures above the lower bound in the cold TES, so that combination of HW and RA did not merit additional characterization in Thermal Desktop. As expected, required HW to sustain component temperatures above operational bounds increases with increasing RA in Table 5.

Importantly, the Simulink model also provides heat flux boundary conditions to the Thermal Desktop model. The cumulative maximum and minimum heat absorbed by each component from all radiation, which differs in each Simulink simulation for a given RA, is divided by the external area of that component. The resulting heat flux is provided to Thermal Desktop as a boundary condition for each externally facing model component, and the absorptivity for that heat flux is unity because spectral and specular calculations were performed in Simulink. There is then, in this SoM, no need to re-perform specular or spectral absorptivity calculations in Thermal Desktop for external components, and radiation calculations are simplified. Internal radiation must still be evaluated, including selection of surface coatings, but Thermal Desktop allows engineers to select component surface coatings and compute radiative view factors per mesh node. While no Thermal Desktop results were presented in this work, future publications will characterize differences between ABEX SoM results and other modern Thermal Desktop practices.

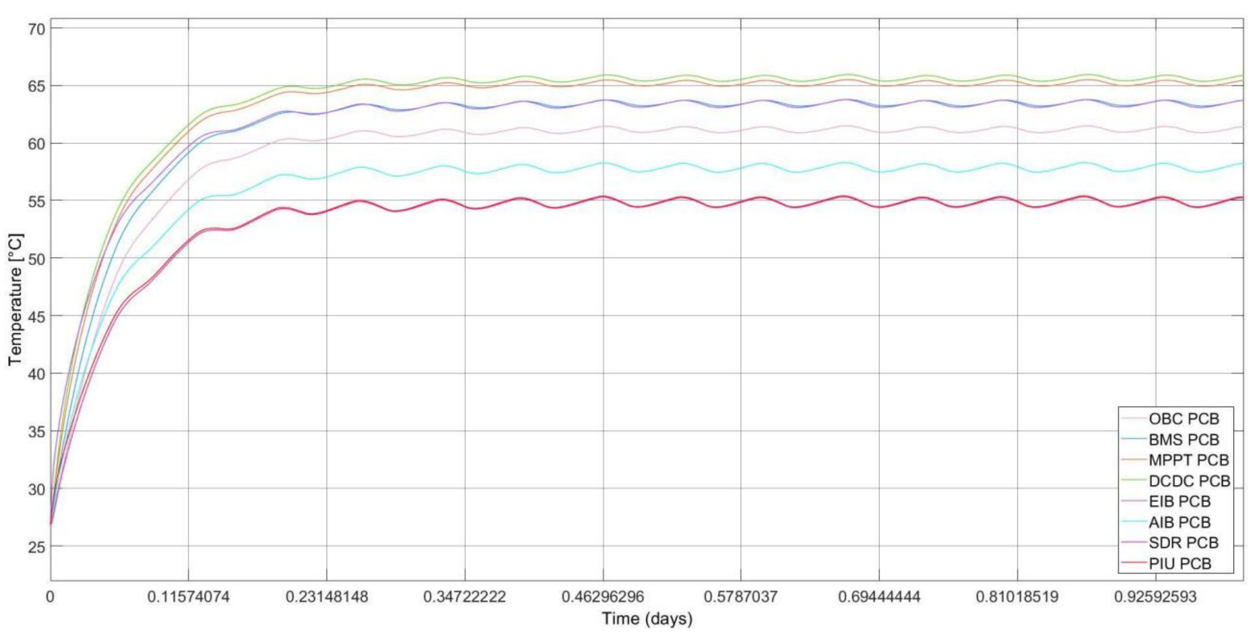

Figure 13: Avionics Stack PCB Transient Temperature Profiles

Table 5: Simulink Heater Wattage and Radiator Area Operational Envelope

	Maximum Heater Wattage per Heater, [W]														
Radiator Area, [m^2]	5.00	5.25	5.37	5.49	5.60	5.72	5.78	5.81	5.84	5.92	6.00	6.25	6.50	6.75	7.00
0.0046	0	0	1	1	1	1	1	1	1	1	1	1	1	1	1
0.0092	0	0	0	1	1	1	1	1	1	1	1	1	1	1	1
0.0138	0	0	0	0	1	1	1	1	1	1	1	1	1	1	1
0.0184	0	0	0	0	0	1	1	1	1	1	1	1	1	1	1
0.0230	0	0	0	0	0	0	1	1	1	1	1	1	1	1	1
0.0245	0	0	0	0	0	0	0	1	1	1	1	1	1	1	1
0.0253	0	0	0	0	0	0	0	0	1	1	1	1	1	1	1
0.0276	0	0	0	0	0	0	0	0	0	1	1	1	1	1	1
0.0307	0	0	0	0	0	0	0	0	0	0	1	1	1	1	1
0.0322	0	0	0	0	0	0	0	0	0	0	1	1	1	1	1
0.0368	0	0	0	0	0	0	0	0	0	0	0	1	1	1	1

CONCLUSION

Planning the scope, outputs, and limitations of thermal models within a MBSE-motivated SoM can provide a clear analysis methodology that can be verified and validated by industry standards such as NASA-STD-7009A. Sourcing model data inputs from a central source of truth such as a SysML model precludes the need for version control efforts within disparate models and affords high rankings for data pedigree and input pedigree. Additional work must be done to characterize the ABEX spacecraft within its operational environment, but the process for executing M&S for TPM evaluation within the ABEX program is thoroughly defined using a synthesis of SysML, STK, SPENVIS, MATLAB, Simulink, and Thermal Desktop.

Acknowledgments

The Alabama Space Grant Consortium is thanked for its fastidious support of ABEX through the Alabama CubeSat Initiative. Stephanie Mauro and Carlos Gomez of NASA Marshall Space Flight Center are thanked for their fastidious support of ABEX students.

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