Thermal Control Development and Integration Plan

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Change Log

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Thermal Control	1.0	Added previously written TQP material	03/27/2022
Thermal Control	1.1	Added Temperature Envelop Table	04/03/2022
Thermal Control	1.2	Revised PBS and Integration Schedule	04/04/2022
Thermal Control	1.3	Removed Radiation Content for the Radiation Analysis Plan	04/05/2022
Thermal Control	1.4	Added Intro and Process Flow Diagram	04/07/2022
Thermal Control	1.5	Revised Introduction and Process Sections	04/13/2022
Thermal Control	1.6	Added Sub-team Modeling Tools and Descriptions	04/15/2022
Thermal Control	1.7	Added Knowledge Point Tables and Revised Structure	04/19/2022
Thermal Control	1.8	Revised Structure	04/20/2022
Thermal Control	1.9	Added New DKMs, Content, and Revised Structure	05/02/2022

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No.	Section	Description	Date

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1 Introduction

1.1 Document Description

The Thermal and Radiation Team "Development and Integration Plan" (DIP) will establish the analytical methods, tools, technology development approach, and the integration flow to mature this subsystem to a Technology Readiness Level (TRL) of 5. This document will also provide reference material aimed at educating on the analysis process and associated system engineering concepts for the material covered within.

1.2 Scope of Applicability

The document will establish the necessary analysis efforts to analytically verify the Thermal and Radiation Team design. It will also cover the technology development efforts up to TRL 5, which will include descriptions of the testing demonstrations and the flight unit integration as applicable. The TRL 6 effort will be covered by the Subsystem Qualification Plan and the flight unit's acceptance testing prior to spacecraft assembly is covered in the Subsystem Acceptance Plan. The system wide integration will be covered in the ABEX Integration Plan. The application of this plan, such as the specific analysis and testing described within, will be published in separate analysis and test reports as needed during the project design analysis cycles.

1.3 Reference Documents

The following project and subsystem documents should be referenced as needed.

Table 1: Relevant Project Documents

Project Document Name	Description of Document
NASA Systems Engineering Handbook	Description of NASA Systems Engineering terms and processes.
NASA SOA 2021	State of the art small spacecraft technology report as of October 2021.
ABEX Space Radiation Environment	Description of the radiation environment that will be encountered by the proposed ABEX orbit.
On the Design, Synthesis, and Radiation Effect Prevention of a 6U Deep Space CubeSat	Report of small spacecraft design, synthesis, and radiation effect prevention as beneficial information for the proposed ABEX spacecraft.
Radiation Analysis Plan	Radiation Development and Integration Plan Supplement to Thermal Control

1.4 Process Material

In order to verify radiator area and patch heater wattage for a small spacecraft through analysis, a higher fidelity approach progressing in MATLAB, Simulink, and Thermal Desktop models can be utilized. The models progress in complexity, respectively, to build towards a more accurate radiator area and patch heater wattage for the spacecraft. Analysis begins with an isothermal MATLAB model informs a non-isothermal Simulink model, which then provides an operational envelop to a spatially dependent Thermal Desktop model.

The MATLAB model first simulates the thermal environment of the ABEX spacecraft during its one-year proposed orbit. The end goal for the MATLAB model is to output isothermal estimates for the Patch Heater Wattage and Radiator Area, which are used as the starting point for the higher fidelity Simulink model. The MATLAB model reads in the spacecraft orbit data provided by the ABEX Orbit Team and iterates through the whole mission, performing several calculations such as calculating the absorbed heats per spacecraft face and operational heat generated to develop a range from the maximum thermal condition to the minimum thermal condition for the estimated Patch Heater Wattage and Radiator Area.

The isothermal required Patch Heater Wattage is a function of several things: the Minimum Thermal Condition, Minimum Component Operational Temperatures, and Satellite Outgoing heat. After acquiring each of these variables, they can be input into an isothermal energy balance in the MATLAB code. The Patch Heater is set to only turn on if the isothermal temperature is less than 0°C. As the analysis becomes more complex, the isothermal assumption will be discarded due to its inaccuracy and a non-isothermal model will be used. The Radiator Area calculation begins much like the patch heater calculation; an isothermal analysis is performed in MATLAB, but also considers the presence of the solar arrays. Like the Patch Heater Wattage, the isothermal Radiator Area estimate will be discarded, and the non-isothermal model will be used in further analysis.

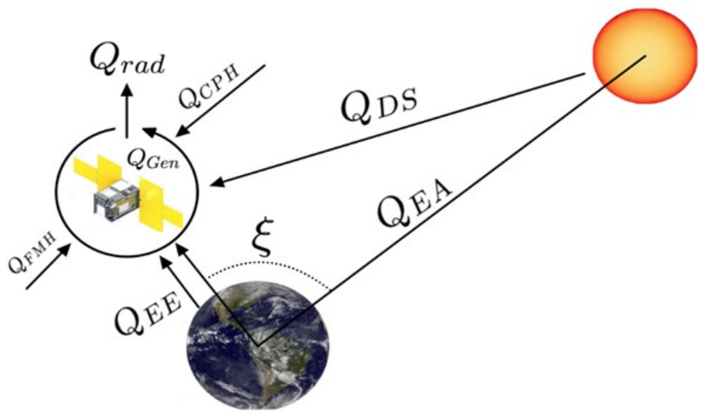

Figure 1: Incoming Heat Sources on the Spacecraft

The MATLAB code analyzes the various heat fluxes incident on the satellite, shown in Figure 1 above. These heat fluxes are translated to absorbed heats per face on the satellite, calculated through the combination of incident flux, projected area, wavelength dependent reflectivity, and an angle of incidence. Using the attitude vectors off the faces of the spacecraft and the position vectors from the spacecraft to the Earth and Sun, vector math calculations are performed to determine which faces receive Earth Albedo, Earth Emission, and Direct Solar radiation at each orbit time step. The MATLAB model then calculates incident radiation for each of these sources. The calculation displays that a face is receiving radiation from a source through the angle of incidence between the normal vector of the face and the incident vector of that source radiation. The incident radiation value is then multiplied by an effective absorptivity determined through a Fresnel relation analysis, and that value is multiplied by the projected area of the affected face to output a single value for the heat absorbed from the incident radiation source on a specific face. This process is repeated on all six faces from for every measured timestep in orbit to generate data. The heat absorbed from Free Molecular Heating (FMH) and Charged Particle Heating (CPH) were calculated through separate relations. For FMH, the face whose attitude vectors align with the spacecraft velocity vectors will be the only face receiving FMH. To calculate CPH absorbed heat, an orbit text file with positional coordinates and time steps is uploaded to the space environment modeling interface tool SPENVIS. With this orbit uploaded, SPENVIS has available models for deriving integral fluxes from Van Allen Belt trapped electron, Van Allen Belt trapped proton, solar energetic particle, and galactic cosmic ray radiation sources as functions of charged electron and proton energy levels in space. A heat flux for each of the four charged particle radiation sources can be calculated from combining its respective integral fluxes, specific powers, and penetration depths with the chosen spacecraft material density. This charged particle heat flux is multiplied by the area of each face to obtain CPH absorbed heat.

The final source of heating, Operational Heat Generation, occurs due to the use of the electrical components of each software mode and their associated efficiencies as well as the heat generated from the battery and solar arrays while neglecting the intentional heat produced by the patch heaters. The heat generated from the software modes of operation is dependent on phase. Once all heat values and thermal management component estimates have been calculated, analysis can begin in Simulink.

The Simulink model is a thermal resistance network whose configuration was created by referencing the current CAD model; thermal resistances, masses, thermal contact resistances, specific heats, and emissivities were either researched or tabulated, then used to define the network. By accepting the radiation profile from MATLAB and SPENVIS in addition to the maximum radiator area and patch heater wattage values, the constructed Simulink model was able to elevate the isothermal model into a non-isothermal, quasi three-dimensional model that provided a higher-fidelity picture of the spacecraft's thermal environment during orbit. Once the network was defined, a brute force analysis was utilized to generate an operational envelop to determine which patch heater wattage and radiator area combinations were viable for testing in the Thermal Desktop Model.

This brute force approach began with the formulation of two vectors ranging from zero to two-times the isothermally calculated maximum patch heater wattage and radiator area values. One hundred equally spaced values populated each vector, therefore initializing 10,000 different combinations. The Simulink model was evaluated for each of these combinations, and if at any time during the orbit a component strayed from its operable temperature range, the index corresponding to the failed combination was populated with a zero. This process was repeated for both the maximum and minimum thermal environmental states, and the results were overlaid. Following elimination of unfeasible combinations, the remaining pairings were provided to Thermal Desktop for further consideration.

Thermal Desktop team receives the matrix of patch heater wattages and radiator areas that are produced from the Simulink model. These are then constructed within the AutoDesk program AutoCAD. The 3D models within AutoCAD are produced from information given by the Structural and Payload sub-teams. The Thermal Desktop model applies thermophysical and optical properties to the spacecraft 3D models. Since Thermal Desktop is a Finite Element Analysis tool, the spacecraft is built at a low level of detail to facilitate obtaining simulation results in reasonable amounts of time. Once the thermophysical and optical properties are assigned to the low-detail 3D model, the physical connections between faces can be input as “connectors”. Finally, the absorbed heats from the MATLAB model can be applied to the Thermal Desktop models varied only by the patch heater wattage and radiator area combinations provided by the Simulink Model. Tests for the hot, cold, and trend thermal environmental states are performed in the Thermal Desktop model, and the ideal combination is confirmed alongside Simulink results.

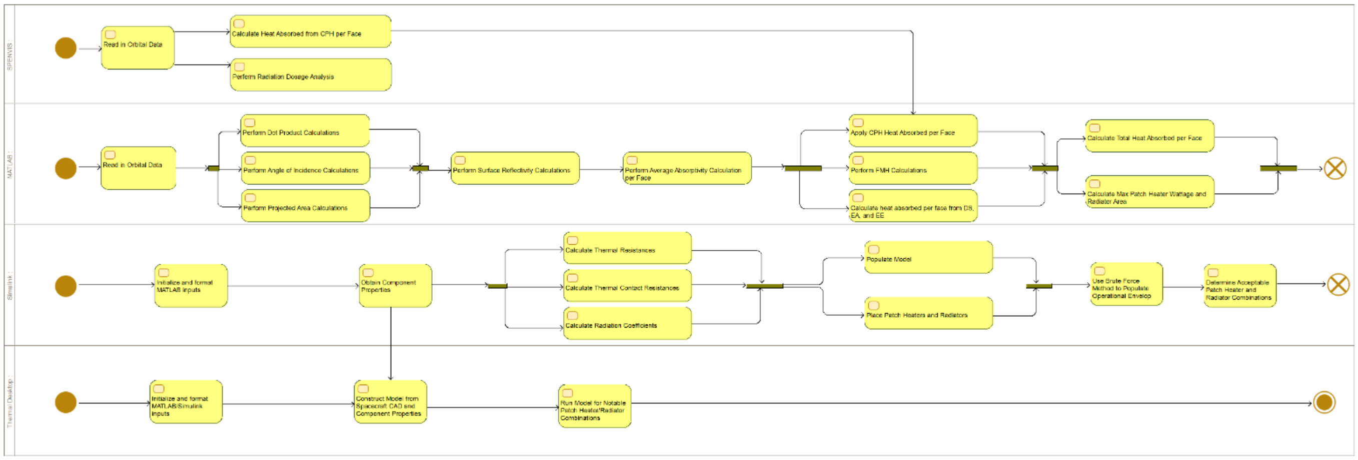

Figure 2: Thermal Control Process Diagram

2 Thermal Control and Radiation Technical Summary

2.1 System Description

The ABEX Thermal Control team determines operational envelopes of acceptable heater wattages and radiator areas to maintain a thermal energy balance in space. The models' input data are self-consistent, meaning all models are using the same input data and can be accurately compared. The result is an amount of heat required to keep the spacecraft warm and an amount of area needed to keep the spacecraft cool at a given surface emissivity. To warm the spacecraft, ABEX has opted to use low-outgassing, surface-mountable patch heaters. Some heaters are mounted to exterior features of spacecraft components while others are mounted directly to electronics, but the purpose remains the same. For the radiator, ABEX is currently evaluating a variety of design options based on first-principle analyses. Knowing the required area and emissivity is sufficient for an energy balance but achieving the required area using a low-mass system requires thought. Radiator design elements facilitating low-mass results include embedded heat pipes, ultrasonic additive manufacturing, thermostructural elements, anisotropic thermal conductivity, and flexible physical interfaces between the Sun-pointing face and the radiators. As ABEX proceeds through PDR and CDR in 2023+, the design will be analyzed for physical integration and thermal functionality, integrated with the spacecraft structural prototypes, and subsystem-level qualification tested in preparation for system-level qualification at Goddard Space Flight Center.

2.2 Product Breakdown Structure

The Product Breakdown Structure (PBS) of the Thermal Control Team is designed to organize the system elements into a hierarchy for work, planning, documentation, and requirements planned around the physical things that a project needs to make. The PBS for ABEX includes a numbering pattern outlining system team name (the Thermal Control and Radiation Team ID number is 7), flight products, development products, and lower-level development products needed to mature the systems technology. 7.3 equipment will get Thermal Control aspects to TRL 4, and 7.2 equipment will get Thermal Control aspects to TRL 5. The PBS will also establish a common terminology for the system for the team and for the project's system engineering effort. The flight products are needed for the final mission, and the development products help raise flight product technology readiness level (but do not end up in the final mission). For additional details on makeup of the PBS, see Appendix F.

Table 2: Product Breakdown Structure of the Thermal Control and Radiation Team

PBS ID	Product Name	Product Description
7	Thermal Control	Thermal Control Team Name
7.1	Flight Hardware	Flight System Elements
7.1.1	Patch Heaters	Spacecraft Heating Element (Active)
7.1.1.1	PH GRD 1	Flight PH on GRD
7.1.1.2	PH GRD 2	Flight PH on GRD
7.1.1.3	PH GRD 3	Flight PH on GRD
7.1.1.4	PH XRD 1	Flight PH on XRD
7.1.1.5	PH XRD 2	Flight PH on XRD
7.1.1.6	PH XRD 3	Flight PH on XRD
7.1.1.7	PH XRD 4	Flight PH on XRD
7.1.1.8	PH XRD 5	Flight PH on XRD
7.1.1.9	PH XRD 6	Flight PH on XRD

7.1.2	Radiators	Spacecraft Cooling Element (Passive)
7.1.2.1	Radiator +Y	Radiator on +Y Face
7.1.2.2	Radiator -Y	Radiator on -Y Face
7.2	Prototype Hardware	All the Technology Prototype Hardware
7.2.1	PH Prototypes	Prototype Spacecraft Heating Elements
7.2.1.1	PH Prototype 1	Prototype PH 1
7.2.1.2	PH Prototype 2	Prototype PH 2
7.2.1.3	PH Prototype 3	Prototype PH 3
7.2.1.4	PH Prototype 4	Prototype PH 4
7.2.1.5	PH Prototype 5	Prototype PH 5
7.2.1.6	PH Prototype 6	Prototype PH 6
7.2.1.7	PH Prototype 7	Prototype PH 7
7.2.1.8	PH Prototype 8	Prototype PH 8
7.2.1.9	PH Prototype 9	Prototype PH 9
7.2.2	Radiator Prototypes	Prototype Spacecraft Cooling Elements
7.2.2.1	Radiator Prototype	Prototype Radiator
7.2.3	Microcontrollers	Integrated Circuit to Govern for an Operation
TBD	PH Prototype Microcontroller Board	Will Fit in Avionics Box, Shared with EPS
7.2.3.1	Radiator Microcontroller	Radiator Integrated Circuit for Testing
7.2.4	Thermocouples	Temp Recording Devices
7.2.4.1	PH Thermocouple	PH Temp Recording Device for Testing
7.2.4.2	Radiator Thermocouple	Radiator Temp Recording Device for Testing
7.2.5	High Temp Kapton Tapes	Tape for Thermocouple Connection
7.2.5.1	PH High Temp Kapton Tape	PH Thermocouple Connection Tape
7.2.5.2	Radiator High Temp Kapton Tape	Radiator Thermocouple Connection Tape
7.2.6	CDAQ Units	Data Acquisition Box to Interact with Lab View
7.2.6.1	PH CDAQ Unit	PH Data Acquisition Box
7.2.6.2	Radiator CDAQ Unit	Radiator Data Acquisition Box
1.3.3	Avionics Box Prototype	Avionics Box Shared with Structures
7.2.7	PH Prototypes for Radiator	Spacecraft Heating Element
7.2.7.1	PH Prototype for Radiator	PH for Radiator Testing (TRL 3-4)
7.3	Development Hardware	Technology for Dev Hardware Before Prototypes
7.3.1	Dev PHs	Dev Spacecraft Heating Elements

7.3.1.1	Dev PH	PH for Dev Purposes
7.3.2	Dev Microcontrollers	Integrated Circuit to Govern for an Operation
TBD	PH Dev Microcontroller Board	Will Fit in Avionics Box, Shared with EPS
7.3.3	Dev Thermocouples	Temp Recording Devices
7.3.3.1	Dev PH Thermocouple	PH Temp Recording Device for Dev
7.3.4	Dev High Temp Kapton Tape	Tape for Thermocouple Connection
7.3.4.1	Dev PH High Temp Kapton Tape	PH Dev Thermocouple Connection Tape
7.3.5	Dev CDAQ Units	Data Acquisition Box to Interact with Lab View
7.3.5.1	Dev PH CDAQ Unit	PH Dev Data Acquisition Box

3 System Analysis Plan

3.1 Overview

The Thermal Control Team aims to combine analysis with testing on relevant components and integrated systems for optimal structural aspects before system integration stages. The goal is not to calculate temperatures at certain locations throughout the orbit, but it is to calculate patch heater wattage and radiator area for keeping satellite components inside their operating temperature ranges for the orbit duration. Changes found late in the integration schedule can lead to unnecessary delays and costs. The most sensitive components are the lithium-ion batteries (with an operating range of -10°C to 20°C) and the science instrument (with an operating range of). Once system integration begins, the thermal team will assist in operational and environmental testing efforts.

3.2 Technical Performance Measures

Technical performance measures (TPMs) are monitored measures for comparing the current actual achievement of the parameters with that anticipated at the current time and on future dates [NASA Systems Engineering Handbook].

For reference on establishing good TPMs, see Appendix B for details.

Table 3: List of TPMs for the subsystem

TPM	Rationale	Units
Minimum/Maximum Thermal Conditions	To correctly define the thermal environment, the lowest and highest heat flows into and out of the system must be found so that a quantitative analysis may be made of the potential highest and lowest temperatures in the satellite. This TPM represents a custom array of absorbed heat per spacecraft face and operational ohmic heat; this calculation can be performed for spacecraft Beginning of Life (BOL) or End of Life (EOL).	W
Required Patch Heater Wattage	The passive heat radiating and conducting away from spacecraft elements must be met with active heat so that components remain within their operational and extreme temperature limits.	W
Required Radiator Area	Spacecraft using photovoltaic generation can feature Sun-pointing faces with high temperatures. Radiators physically interfacing with the high-temperature surfaces can be used to radiate heat into space before the heat negatively affects spacecraft electronics.	m^2

3.3 Models & Simulation Tools

“All models are bad, some are useful.” -George Box

To verify radiator area and wattage heat wattage through analysis, the ABEX Thermal Control Team and the ABEX Radiation Team use model and simulation tools to set up the space radiation environment and analyze the thermal aspects of the satellite geometry.

Table 4: List of Analysis Tools and Models with the TPMs They Calculate.

TPM Calculated	Analysis Tool	Reason For Use
Min/Max Thermal Conditions, Required Patch Heater Wattage, and Required Radiator Area	MATLAB	Large Amount of Calculation Customization for an Isothermal Output Estimate
Min/Max Thermal Conditions	SPENVIS	Access to Models of all Five Radiation Effects of Interest
Required Patch Heater Wattage and Required Radiator Area	Simulink	Thermal Resistance Network and Non-isothermal Outputs
Required Patch Heater Wattage and Required Radiator Area	Thermal Desktop	3D FEA Non-isothermal Thermal Analysis

The SPace ENVironment Information System (SPENVIS) is the European Space Agency's interface to various space radiation environment and effect models. The ABEX Radiation Team, considered part of the ABEX Thermal Control Team, uses SPENVIS to access models for five radiation effects of interest: Charged Particle Heating, Total Ionizing Dose, Non-Ionizing Energy Loss, surface charging, and Single Event Effects. With particle flux outputs from SPENVIS and stopping power values provided by NIST's PSTAR and ESTAR platforms, Charged Particle Heat absorbed per face at every time step is calculated in the MATLAB model to assist in the isothermal estimation of radiator area and patch heater wattage. The other four radiation effects are analyzed by the ABEX Radiation Team to advise on avionics-protecting structural design considerations like radiation shield areal density and solar cell coverglass thickness.

A programming language primarily used for numeric computation while also including numerous extensions and add-ons for specific analyses. The ABEX Thermal Control Team imports STK data for a specific orbit and calculates absorbed heat per spacecraft face due to incident radiation and thermal generation. From these absorbed heats, an isothermal spacecraft model is generated to provide a "first guess" heater wattage and radiator area, though the values are overestimated due to the high area-to-mass ratio of the solar array. MATLAB was chosen for this computation due to the ease of importing data and calculating an orbit-specific thermal radiation environment. The "first guess" values and calculated environment parameters are provided to Simulink as a basis for non-isothermal analysis.

Simulink is an interactive, graphical interface that acts as an extension of MATLAB. Through university-provided toolboxes, a wide array of dynamic systems can be transiently modeled, including thermal analysis. With the Simscape toolbox, the ABEX Thermal Control Team constructed a thermal resistance network of the entire spacecraft to analyze different combinations of heater wattage and radiator areas. The pre-programmed blocks can be oriented to successfully represent the spacecraft while providing a relatively high speed, code-based approach to location-specific analysis as opposed to the Finite Element Analysis in Thermal Desktop. Simulink analyzes a range of heater wattages and radiator areas based on provided isothermal values to create an operational envelope of acceptable heater wattages and radiator areas for both minimum and maximum thermal conditions. The coincidence of hottest and coldest operational envelopes represents a subset of acceptable heater wattages and radiator areas for evaluation in Thermal Desktop with physical interfacing and internal radiative view factors considered.

Thermal Desktop is a three-dimensional (3D) computer-aided design (CAD) environment for thermal and fluid simulation. It functions as an add-on to Autodesk's AutoCAD program and excels as a tool for heat conduction calculations, radiative view factor assessment, and orbital definition utilizing 3D finite element analysis (FEA). With the built-in case set manager, the ABEX Thermal Control Team uses Thermal Desktop to quickly iterate and compare different combinations of heater wattage and radiator area that the Simulink thermal resistance network deems acceptable. The primary advantage of performing this analysis in Thermal Desktop as opposed to Simulink and MATLAB is the 3D FEA solver. Obtaining results in three dimensions, even with errors incurred by FEA grid dependence, allows for a higher-fidelity model and a more informed decision about the selection of radiator area, radiator mounting locations, heater wattage, and heater locations. With the radiator area and heater wattage design verified by MATLAB, Simulink, and Thermal Desktop analysis to be compliant to operational temperature requirements, the design is deemed acceptable for integrated qualification testing.

3.4 Relevant Standards

Design standards relevant to the analysis plan for the Thermal Control and Radiation Team are listed here for reference.

Table 5: Relevant Standards and References for the Technical Analysis.

Resource Name	Resource Title	Brief Description of Resource
NASA-STD_7009A	NASA Technical Standard for Models and Simulations	The standard establishes uniform practices for design, development, and use of models and simulations to ensure essential requirements are met.

3.5 Sensitive Component Temperature Envelops

Simulink and Thermal Desktop monitor the temperatures of critical components during simulation to ensure no operating requirements are violated. The following table contains the component names and any temperature conditions they may have.

Table 6: Sensitive Component Temperature Ranges

Component Name	Number of Component	Lower Temperature Bound [C]	Upper Temperature Bound [C]
Gamma Ray Detector	3	-20	80
X-Ray Detector	6	0	60
Battery	8	15	40
Printed Circuit Board	6	-50	120

Payload Interface Unit	1	-40	85

4 Technical Performance Analysis

The following sections will outline the information for analysis of the Thermal and Radiation Team's TPMs. This contains descriptions of the organization schema, domain knowledge maps, methodology descriptions, and all the relevant equations and assumptions to determine them, described in full. See Appendix C for details of how to make Domain Knowledge Map (DKMs) and Appendix A for the list of acronyms, nomenclature, and terms.

4.1 Domain Categories & Relationships

The technical analysis of TPMs requires an understanding of the tools, equations and sources of information needed to evaluate them. This information is captured in a technical domain schema which classifies the categories of things that an engineer will need to interact with to determine TPMs. For a given TPM the process for determining it is recorded within a DKM that species the specific entities, such as equations, tools, information sources, parameters, etc., organized along the technical domain categories with their respective relationships between each other shown as semantic triplets. The categories used in this analysis domain are described in Table 7: Categories with Their Descriptions and .

Table 7: Categories with Their Descriptions and Color Coding Used

Categories	Descriptions	Color Code
Scalar Parameter	This is a (1x1) scalar value including all rational numbers. All parameters must either be calculated, modeled, or sourced on a DKM.	*Hex#: 42D4F4*
Array Parameter	This is a set of (1xN) scalar values including all rational numbers. Arrays can be captured as data, provided by components, or created from a set of scalars.	*Hex#: FFE119*
Matrix Parameter	This is a set of (MxN) scalar values including all rational numbers. Matrices might be generated by components or used as parts of a design scheme. Matrices can be specific to the defined analysis method.	*Hex#: E6BEFF*
State	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.	*Hex#: A9A9A9*
Source	Sources are either other ABEX teams or source material. If a parameter comes from a team, it's considered "provided by". If a parameter comes from a document such as an Interface Control Document (ICD), NASA standard, textbook, or otherwise, it's considered "sourced from".	*Hex#: E6194B*
Equation	Equations are used to define how parameters are functions of each other; parameters can be inputs to Equations or calculated by Equations. They might be analytical or empirical, but equations should always be specified to have a source.	*Hex#: AAFFC3*
Modeling Environment Parameter	A Modeling Environment Parameter is a scalar value that only exists in the context of a Modeling Environment (e.g., mesh density for FEA). Models only exist as analysis tools for determining parameters that serve the same purpose as Equations.	*Hex#: F032E6*
Modeling Environment Module	Modeling Environment Modules encapsulate any computational modeling scheme designed to output a parameter for analysis purposes. Modeling Environments can have multiple Modeling Environment Modules. MATLAB functions or subroutines are Modeling Environment Modules. FEA platforms like ABAQUS have vibrational and meshing modules that serve as distinct Modeling Environment Modules. The concept of a Modeling Environment Module is intended to help subsystem teams express modeling concepts in DKMs.	*Hex#: 469990*

Modeling Environmen t	A Modeling Environment is the platform or program an engineer, architect, or designer model something in (e.g., MATLAB, ABAQUS, Thermal Desktop, Simulink, etc.). The organized creation, development, and execution of models is performed within a Modeling Environment Module that exists within a Modeling Environment.	*Hex#: F58231*

In addition to the categories, the different possible relationships which exist between these entities are described in Table 8. These relationships are expressed in the DKMs as semantic triplets (see Appendix C for more information on that and example).

Table 8: Relationships Used Between Classes as Expressed by Semantic Triplets in the DKM

Relationship	Usage Description
is a function of	Describes a relationship between an entity and another which defines its output.
is compared to	Describes a relationship that some entity is directly compared to another entity.
is sourced from	Describes a relationship between an entity that will get data or value from another entity.
is calculated by	Describes a relationship between an entity which is a direct result from some equation.
is an input to	Describes a relationship between an entity which is provided a value needed by another entity, such as by a parameter to an equation.
is modeled by	Describes a relationship, such as a parameter or environment, which is determined though a model.
is comprised of	Describes the relationship that some entity is a part of another entity.
is provided by	Describes a relationship between an entity and an ABEX team who provides it.
is varied by	Describes a relationship between an entity and a component that may change that entity.

4.2 Min/Max Thermal Conditions

To establish the thermal conditions that the spacecraft will be in during its orbit, the minimum and maximum thermal conditions will be calculated. These are a conservative estimate of the conditions; therefore, it will provide conditions that are much hotter and much colder than the spacecraft should ever experience. The thermal environment that will be established is comprised of two separate sources of heat, incoming heat from outside sources and internal heat generated from operation of the electrical components. The full breakdown of the calculation of the min and max thermal conditions can be found in Error: Reference source not found and the Domain Knowledge Map for the calculation can be found in Figure 4.

<table>
<tr><td>Heat Dissipated by EPS
4.2.1.1.1</td><td rowspan="3">Operational Heat Generation
4.2.1.1</td><td rowspan="8">Min/Max Thermal Conditions
4.2.1</td></tr>
<tr><td>Heat Dissipated by Electrical Components
4.2.1.1.2</td></tr>
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4.2.1.1.3</td></tr>
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4.2.1.2</td></tr>
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4.2.1.2.2</td></tr>
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4.2.1.2.3</td></tr>
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4.2.1.2.4</td></tr>
<tr><td>Absorbed Charged Particle Heating
4.2.1.2.5</td></tr>
</table>

Figure 3: Min/Max Thermal Conditions Knowledge Point Breakdown

Update DKMs

Knowledge Point inclusion and section numbering needs to match table above

Min/Max Thermal Condition Updated -4/18/22 TE

4.2.1 Min/Max Thermal Conditions DKM

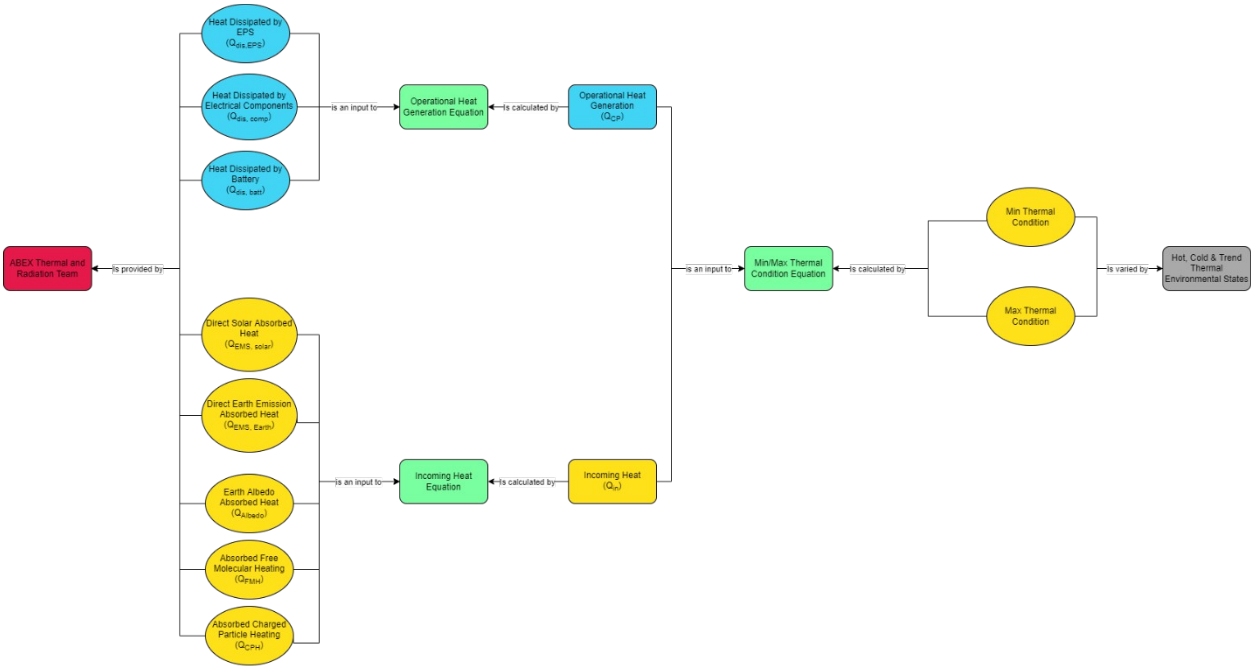


Figure 4: Min/Max Thermal Conditions DKM

4.2.2 Operational Heat Generation

Operational Heat Generation occurs due to the use of the electrical components of each software mode and their associated efficiencies as well as the heat generated from the battery and solar arrays. The heat generated from the software modes of operation is dependent is on time (phase). The DKM for Operational Heat Generation is shown in Figure 5 and is broken down further in the following sections.

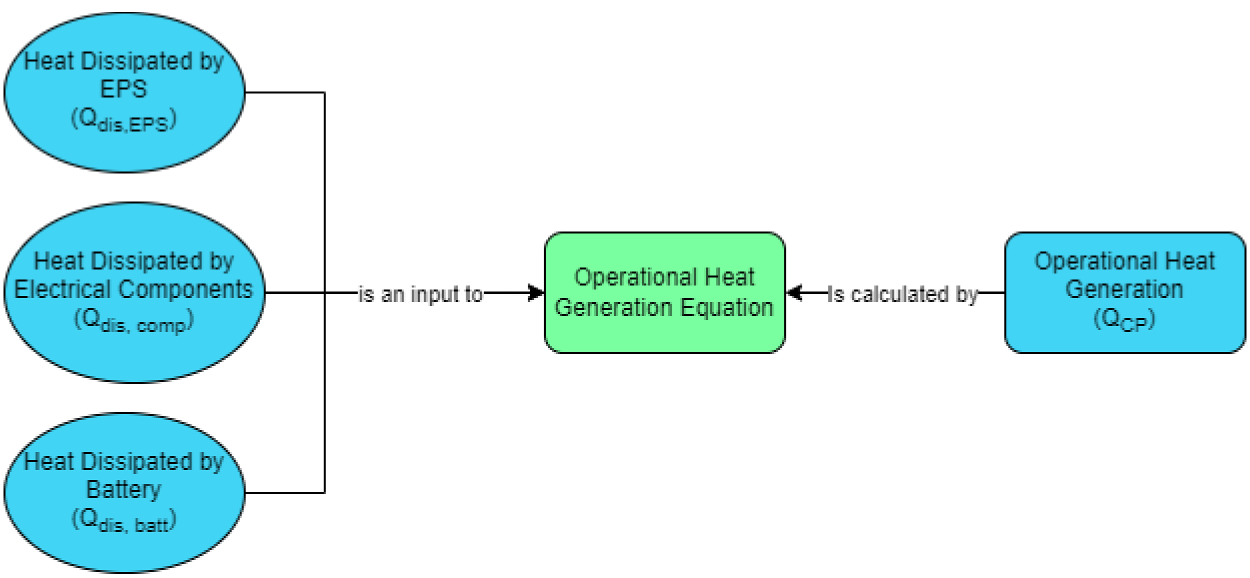


Figure 5: Operational Heat Generation DKM

4.2.2.1 Heat Dissipated by Electrical Power System

Heat dissipated by the Electrical Power System (EPS) is a function of the Component Power and the Component Power Conversion Efficiency; the Component Power Conversion Efficiency is assumed 0.9. The DKM for the Heat Dissipated by Electrical Power System is shown in Figure 6.

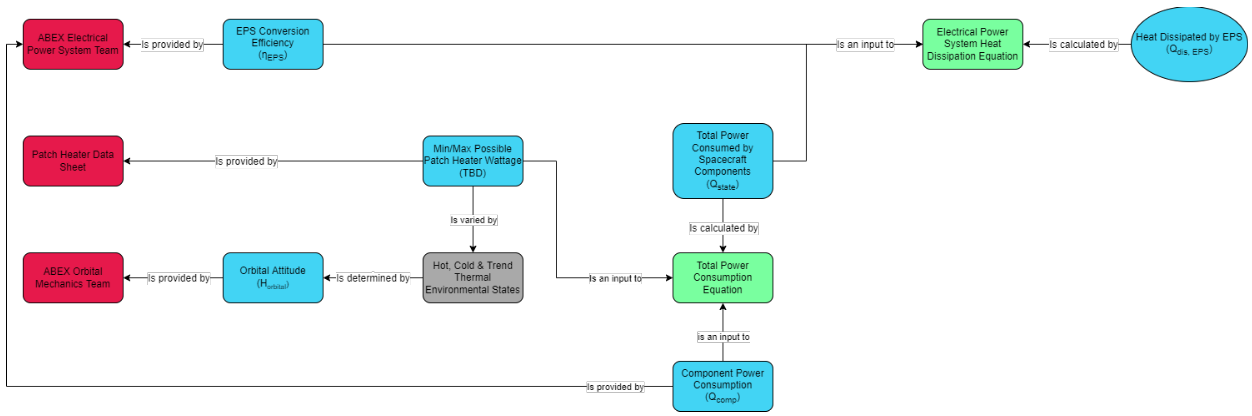


Figure 6: Heat Dissipated by EPS DKM

The Actual Component Power Consumption is a function of each individual component's power consumption, Q_{State}, which is provided by TBD, and the Electrical System Power Conversion Efficiency.

$$Q_{cons,EPS} = \frac{Q_{State}}{\eta_{EPS}} \quad \text{(Eq. 1)}$$

The Heat Dissipated by Electrical Power System is a function of the Actual Component Power Consumption and the Total Power Consumed by Spacecraft Components.

$$Q_{dis,EPS} = Q_{cons,EPS} - Q_{State} \quad \textit{(Eq. 2)}$$

4.2.2.2 Heat Dissipated by Electrical Components

Heat dissipated by the electrical components is a function of the Component Power and the Component Power Conversion Efficiency; the Component Power Conversion Efficiency is assumed 0.9.

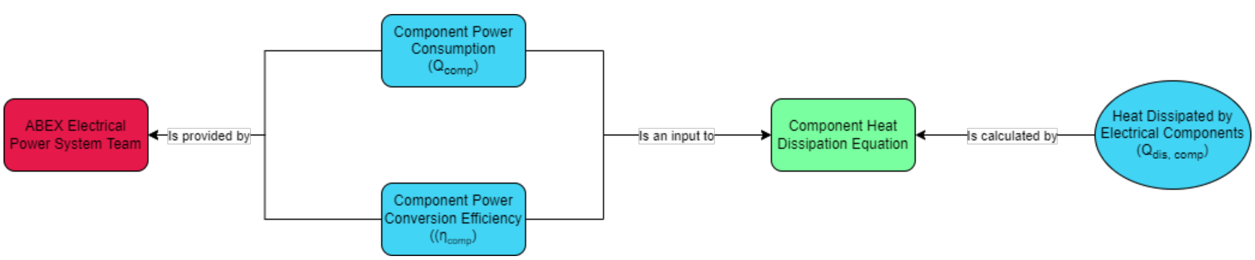


Figure 7: Heat Dissipated by Electrical Components

The Total Power Consumed by Spacecraft Components, Q_{Comp} , a function of the component power consumption, Q_{State}, and the min/max Patch Heater wattage, $Q_{PH\,Min/Max}$. Q_{Comp} will vary with each component due to varying component power requirements. That value is subtracted from the required min and max possible patch heater wattage for each component.

The Total Power Consumed by Spacecraft Components a function of the component power consumption and the min/max Patch Heater wattage. Component power consumption is determined by each individual component's power consumption, Q_{State}. Patch Heater wattage has yet to be determined, however for now, it is assumed to be 23 W for the max case and 0 W for the min case. The Total Power Consumed by Spacecraft Components are calculated in Equation 4.

$$Q_{Comp}=Q_{State}-Q_{PH\,Min/Max} \quad \text{(Eq. 3)}$$

Additionally, the heat dissipated from each component is calculated by multiplying the Component power consumption by one minus the component power conversion efficiency, which is assumed to be 0.9 or 90%.

One minus the component power conversion efficiency allows us to calculate the amount of power that is converted to heat.

$$Q_{dis,comp}=Q_{comp}\left(1-\eta_{comp}\right) \quad \text{(Eq. 4)}$$

4.2.2.3 Heat Dissipated by Battery

The Heat Dissipated by Battery is calculated by multiplying the Net Spacecraft Power by one minus the Battery Conversion Efficiency.

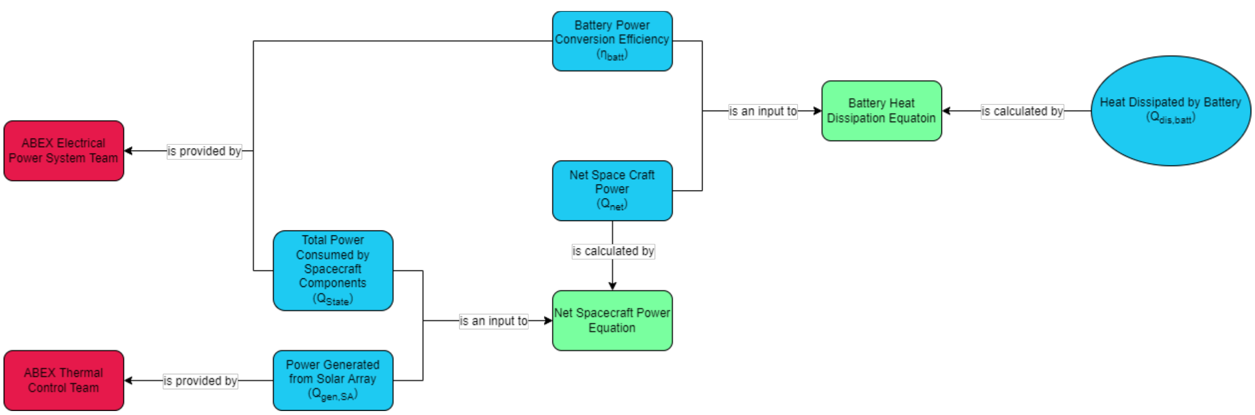


Figure 8: Heat Dissipated by Battery

Net Spacecraft Power is a function of the Power Generated from the Solar Arrays and the actual component Power Consumption, $Q(cons,EPS)$

The Actual Component Power Consumption is a function of the Total Power Consumed by Spacecraft Components and Electrical Power System Power Conversion Efficiency

$$Q_{net}=Q_{gen,SA}-Q_{cons,EPS} \quad \text{(Eq. 5)}$$

The Heat Dissipated by the Battery is a function of the Net Spacecraft Power and the Battery Power Conversion Efficiency.

The Battery Power Conversion Efficiency varies depending on the Net Spacecraft Power value.

If $Q_{net} > 1$

Then,

$$\eta_{bat} = 0.95$$

And

If $Q_{net} < 1$

$$\eta_{bat} = 0.91$$

Where,

$$Q_{net} = Q_{gen,SA} - Q_{cons,EPS} \quad \text{(Eq. 6)}$$

$$Q_{dis,bat} = Q_{net}(1 - \eta_{bat}) \quad \textit{(Eq. 7)}$$

4.2.2.4 Heat Generated by Operation

The total heat generated from the Software Modes of Operation is calculated by summing the heat dissipated from EPS, each component, and the battery.

$$Q_{Op} = Q_{dis,EPS} + Q_{dis,comp} + Q_{dis,bat} \quad \textit{(Eq. 8)}$$

4.2.3 Incoming Heat

There are five types of incoming radiation that the spacecraft will encounter throughout its mission life. These types are Direct Solar Absorbed Heat, Direct Earth Emission Absorbed Heat, Earth Albedo Absorbed Heat, Free Molecular Heating and the DKM for it can be found in Figure 9: Incoming Heat.

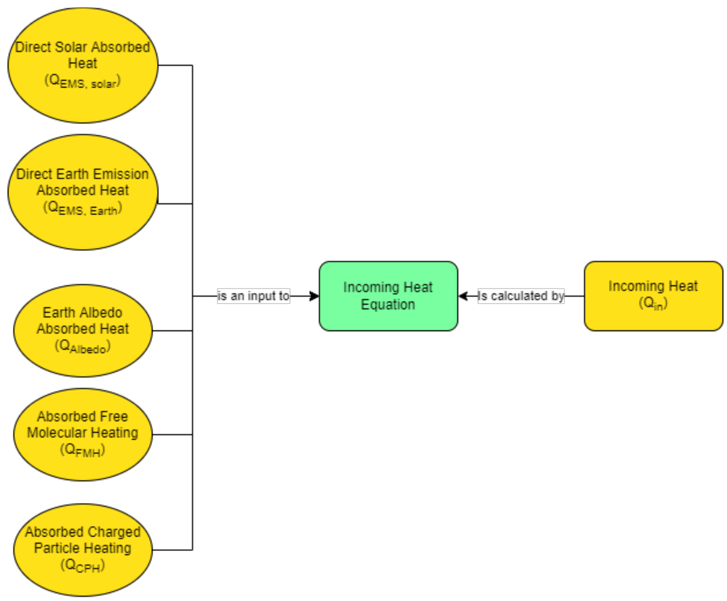


Figure 9: Incoming Heat

4.2.3.1 Direct Solar Absorbed Heat

Direct solar absorbed heat is caused by radiation generated by the sun. Direct solar absorbed heat is dependent on:

- Satellite distance to the Sun
- Eclipse shadowing (when Earth's shadow blocks out the Sun's radiation)
- Spectral Emissive Power of the Sun
- Direct Solar Projected Area
- Direct Solar Angle of Incidence
- Direct Solar Effective Absorptivity

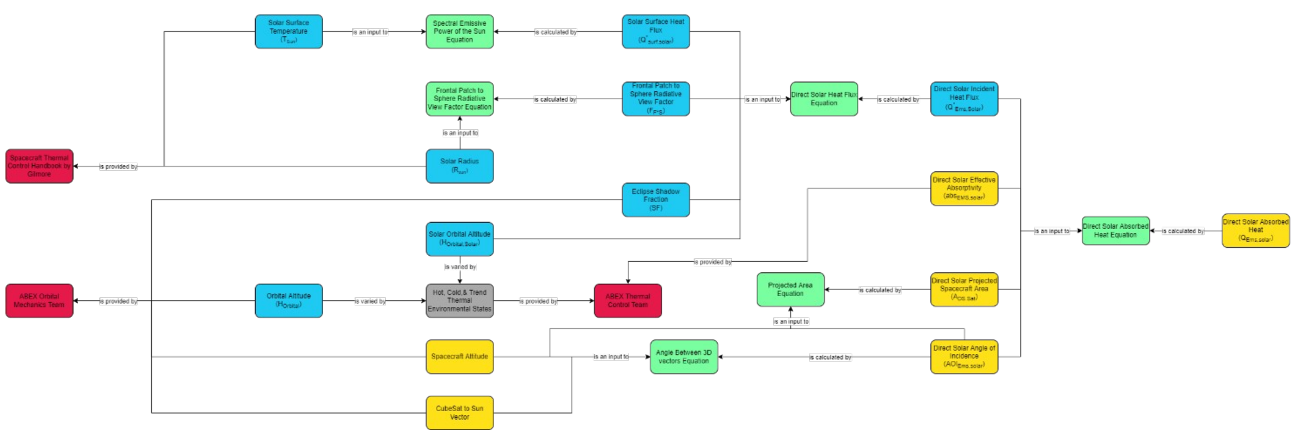


Figure 10: Direct Solar Absorbed Heat

4.2.3.1.1 Shadow Fraction

The beta angle is the angle between the solar vector and its projection onto the orbit plane. A visualization of this angle is shown in Figure 11: Beta Angle.

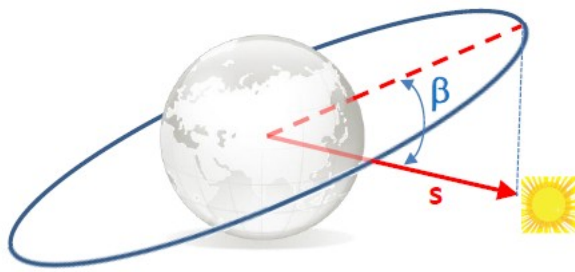


Figure 11: Beta Angle

4.2.3.1.2 Spectral Emissive Power of the Sun

The spectral emissive power is the rate at which radiation is generated on a surface per unit wavelength. The Sun's radiation is present in the form of 3 types of light waves: infrared, visible and ultraviolet. Most of this radiation is made up of visible and infrared light, with only a small percentage being UV light.

The calculation for the spectral emissive power of the Sun is shown in Equation 9, where h_p is Plank's constant, c_0 is the speed of light, n is the refractive index of the medium, which is unity for a vacuum, λ is the wavelength in microns, T is the temperature of the Sun modelled as a blackbody, and k_B is Boltzmann's constant.

$$E_{\lambda b} = \frac{2 \cdot \pi \cdot h_p \cdot c_0^2}{n^{2\cdot} \lambda^{5\cdot} \left[\exp\left(\frac{h_p \cdot c_0}{n \cdot k_B \cdot \lambda \cdot T} \right) - 1 \right]} \qquad \text{(Eq. 9)}$$

A graph of the spectral emissive power of the Sun over wavelength is shown in Error: Reference source not found. The dashed lines show the separation of wavelength types (UV: 0.1 – 0.4 microns, visible: 0.4 – 0.8 microns, IR: 0.8 – 4.5 microns).

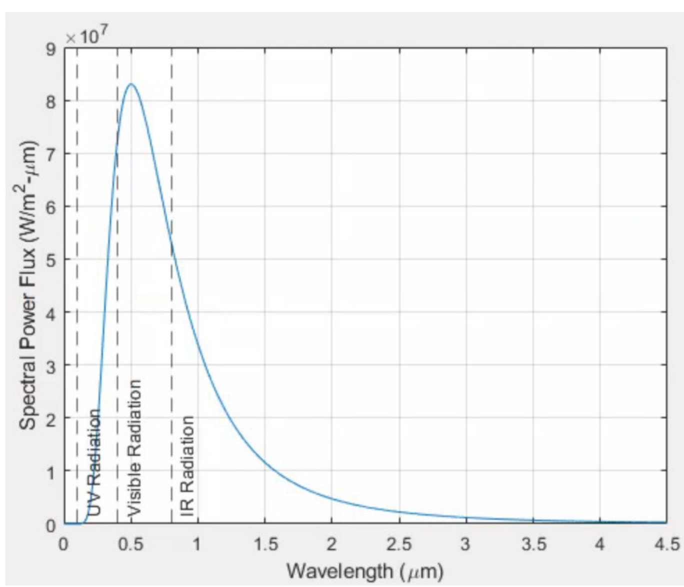


Figure 12: Spectral Emissive Power of the Sun Over Wavelength

From this graph, the wavelength breakdown of the Sun's radiation can be found by calculating the area under the curve in Error: Reference source not found. The Sun emits radiation in the form of mainly visible and IR light. The calculated percentages are shown below:

- UV: 12.4%
- Visible: 46.35%
- IR: 41.25%

To calculate the solar surface heat flux (which is needed for the heat flux calculation), the Spectral Emissive power was integrated over the wavelength range of 0.1 – 4.5 microns, since 99% of the Sun's radiation falls within that range. This integration was done in MATLAB using the Compact Simpsons 1/3 Method, resulting in a solar surface heat flux shown below.

$$Q''_{solar,surf} = \frac{62{,}843{,}984\,W}{m^2}$$

4.2.3.1.3 Direct Solar Heat Flux

The calculation for direct solar heat flux is shown in (Eq. 10, where Q''_{Solar} represents the heat flux from the Sun at a given distance in W/m^2. $Q''_{solar,surf}$ is the heat flux from the Sun at its surface, D_{sun} is the distance from the satellite to the Sun, R_{sun} is the radius of the Sun, and SF is the shadow fraction caused by the Earth's shadow blocking out some or all of the Sun's radiation. Figure 9 shows the full DKM for the calculation of the Direct Solar Heat Flux and also the calculation for the Direct Solar Absorbed Heat. To calculate the Direct Solar Absorbed Heat, the heat flux is applied to the projected area. The projected area will change for each heat source and is dependent mainly on the spacecrafts attitude.

$$Q''_{Solar} = \frac{Q''_{solar,surf}}{\left(\frac{D_{sun}}{R_{sun}}\right)^2} \cdot (1 - SF) \qquad \textit{(Eq. 10)}$$

Note that the direct solar heat flux will trend higher for satellite altitudes near Apogee (closer to the Sun) and is inversely proportional to the shadow fraction.

4.2.3.1.4 Direct Solar Angle of Incidence

Angle of Incidence is described as the angle of the incoming radiation and the normal vector of the face the radiation is incident upon. Given this, the vector components from the satellite to the sun and the attitude vector components of the spacecraft are used to calculate the dot product between them. Since the angle in the dot product is the angle found between the two vectors, it can be reasonably stated that the angle in this case is also the Direct Solar Angle of Incidence. Therefore, the components of the Direct Solar Angle of Incidence can then be calculated by

$$AOI_{DS} = \left[\cos^{-1}\left(\frac{V_{CsS,x} \cdot V_{ScF,x}}{\left|\overrightarrow{V_{CsS}}\right| * \left|\overrightarrow{V_{ScF}}\right|} \right) \cos^{-1}\left(\frac{V_{CsS,y} \cdot V_{ScF,y}}{\left|\overrightarrow{V_{CsS}}\right| * \left|\overrightarrow{V_{ScF}}\right|} \right) \cos^{-1}\left(\frac{V_{CsS,z} \cdot V_{ScF,z}}{\left|\overrightarrow{V_{CsS}}\right| * \left|\overrightarrow{V_{ScF}}\right|} \right) \right] \quad \textit{(Eq. 11)}$$

where $\overrightarrow{V_{CsS}}$ is the vector of the spacecraft to the sun and $\overrightarrow{V_{ScF}}$ is the attitude vector of a spacecraft face.

4.2.3.1.5 Direct Solar Projected Area

The direct solar projected area is the area of a spacecraft face which is receiving radiation. It is dependent on the area of the whole face then the angle of incidence of each component normal to the face that is receiving radiation. It is assumed that the faces of the spacecraft are flat and rectangular. The total projected area for a face given by Direct Solar is given by

$$A_{projected,Ds} = \left(L_{CS} * w_{CS}\right) * \cos\left(\theta_x\right) * \cos\left(\theta_y\right) * \cos\left(\theta_z\right) \quad (Eq.\ 12)$$

where L_{CS} is the length of the spacecraft face, w_{CS} is the width of the spacecraft face, and θ is the component angle of the Direct Solar Angle of Incidence.

4.2.3.1.6 Direct Solar Effective Absorptivity

4.2.3.1.7 Direct Solar Absorbed Heat

Direct Solar Absorbed Heat per face can be calculated by using

$$Q_{Ems,Solar} = Q''_{Solar} * A_{projected,DS} * \left|\text{¿}_{EMS,Solar}\right| \text{¿} \quad (Eq.\ 13)$$

4.2.3.2 Direct Earth Emission Absorbed Heat

Earth emission heating is radiation generated by the Earth, which is exclusively infrared (IR) radiation. Earth emission heating is dependent on:

- Satellite distance to Earth
- Spectral Emissive Power of the Earth
- Direct Earth Emission Projected Area
- Direct Earth Emission Angle of Incidence
- Direct Earth Emission Effective Absorptivity

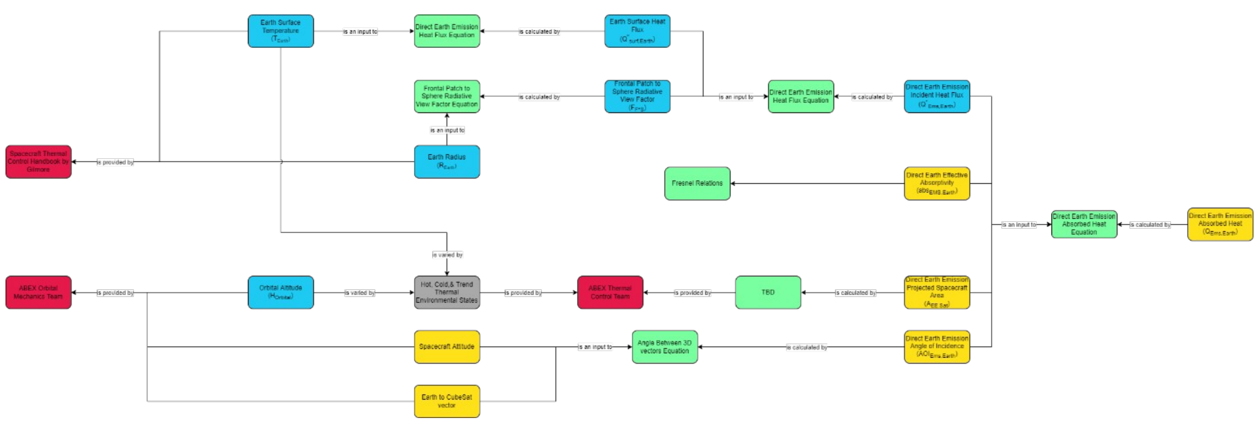


Figure 13: Direct Earth Emission Absorbed Heat

4.2.3.2.1 Spectral Emissive Power of the Earth

The spectral emissive power is the rate at which radiation is generated on a surface per unit wavelength. The Earth's radiation is present only in the form of Infrared (IR) light.

The calculation for the spectral emissive power of the Earth is shown in Equation 18, where h_p is Plank's constant, c_0 is the speed of light, n is the refractive index of the medium, which is unity for a vacuum, λ is the wavelength in microns, T is the temperature of the Earth modelled as a blackbody, and k_B is Boltzmann's constant.

$$E\square_{\lambda b} = \frac{2 \cdot \pi \cdot h_p \cdot c_0^2}{n^2 \cdot \lambda^5 \cdot \left[\exp\left(\frac{h_p \cdot c_0}{n \cdot k_B \cdot \lambda \cdot T}\right) - 1\right]}$$

A graph of the spectral emissive power of the Earth over wavelength is shown in Error: Reference source not found. This graph has a similar shape to the spectral emissive power of the Sun, shown in Error: Reference source not found. However, the radiation emitted by the Earth is at lower wavelengths and is thus purely IR radiation. Since the temperature of the Earth varies for the hot, cold, and trend TES, the emissive power of the Earth will be slightly different for the hot, cold, and trend environmental states.

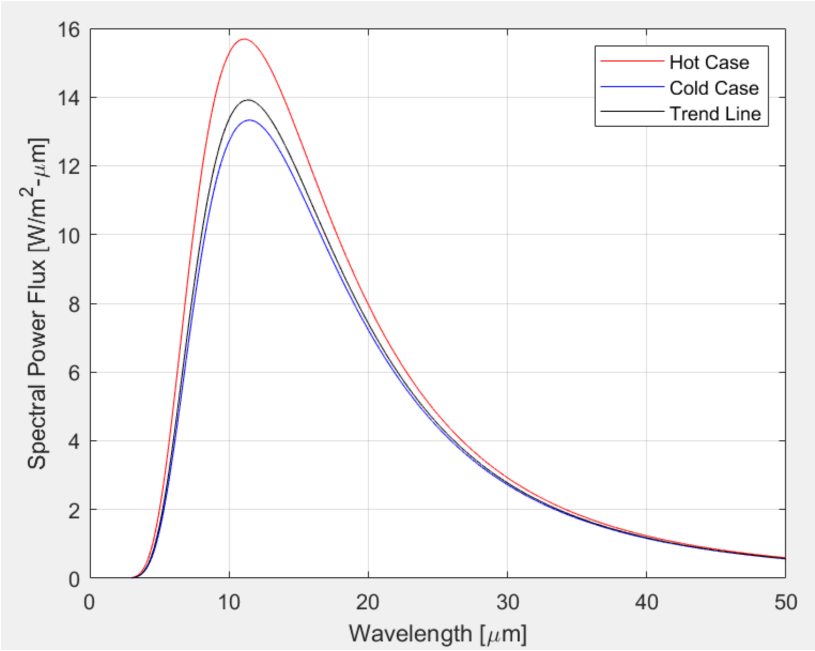


Figure 14: Spectral Emissive Power of the Earth over Wavelength

To calculate the surface heat flux of the Earth (which is needed for the heat flux calculation), the Spectral Emissive power was integrated over the wavelength range of 3.0 – 50 microns, since 99% of the Earth's radiation falls within that range. This integration was done in MATLAB using the Compact Simpsons 1/3 Method. The trend Earth surface heat flux is shown below.

$$Q''_{earth,surf} = 229\, W/m^2$$

4.2.3.2.2 Direct Earth Emission Heat Flux

The calculation for Earth emission heat flux is shown in (Eq. 14, where $Q''_{Emission}$ represents the heat flux from the Earth at a given distance in W/m^2. $Q''_{earth,surf}$ is the heat flux at Earth's surface, calculated in section 3.2.2, D_{earth} is the distance from the satellite to the Earth, and R_{earth} is the radius of the Earth.

$$Q''_{Emission} = \frac{Q''_{earth,surf}}{\left(\frac{D_{earth}}{R_{earth}}\right)^2} \tag{Eq. 14}$$

4.2.3.2.3 Direct Earth Emission Angle of Incidence

The components of the Direct Earth Emission Angle of Incidence can then be calculated by

$$AOI_{Ems,Earth} = \left[\cos^{-1}\left(\frac{V_{CsE,x} \cdot V_{ScF,x}}{\left|\overrightarrow{V_{CsE}}\right| * \left|\overrightarrow{V_{ScF}}\right|}\right)\cos^{-1}\left(\frac{V_{CsE,y} \cdot V_{ScF,y}}{\left|\overrightarrow{V_{CsE}}\right| * \left|\overrightarrow{V_{ScF}}\right|}\right)\cos^{-1}\left(\frac{V_{CsE,z} \cdot V_{ScF,z}}{\left|\overrightarrow{V_{CsE}}\right| * \left|\overrightarrow{V_{ScF}}\right|}\right)\right] \tag{Eq. 15}$$

where $\overrightarrow{V_{CsE}}$ is the vector of the spacecraft to the Earth.

4.2.3.2.4 Direct Earth Emission Projected Area

The Direct Earth Emission Projected Area is the area of a spacecraft face which is receiving radiation. It is dependent on the area of the whole face then the angle of incidence of each component normal to the face that is receiving radiation. It is assumed that the faces of the spacecraft are flat and rectangular. The total projected area for a face given by Direct Earth Emission is given by

$$A_{EMS,projected} = (L_{CS} * w_{CS}) * \cos(\theta_x) * \cos(\theta_y) * \cos(\theta_z) \tag{Eq. 16}$$

where L_{CS} is the length of the spacecraft face, w_{CS} is the width of the spacecraft face, and θ is the component angle of the Direct Earth Emission Angle of Incidence.

4.2.3.2.5 Direct Earth Emission Effective Absorptivity

4.2.3.2.6 Direct Earth Emission Absorbed Heat

Direct Earth Emission Absorbed Heat per face can be calculated by using

$$Q_{Ems,Earth} = Q''_{EMS} * A_{EMS,projected} * |¿_{EMS,Earth}|¿ \tag{Eq. 17}$$

4.2.3.3 Earth Albedo Absorbed Heat

Earth albedo heating is the portion of radiation that is generated by the sun and reflected off the surface of the Earth. Earth albedo heating is dependent on:

- Eclipse shadowing from the Sun
- Satellite distance to Earth
- Solar zenith angle
- Albedo factor of the reflected surface
- Earth Albedo Projected Area
- Earth Albedo Angle of Incidence
- Earth Albedo Effective Absorptivity

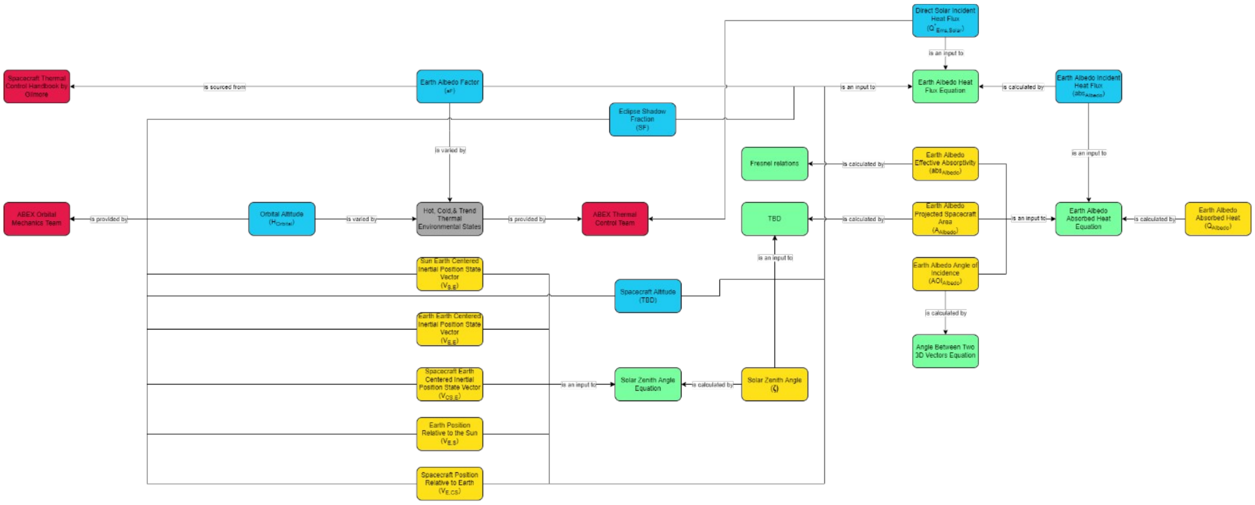


Figure 15: Earth Albedo Absorbed Heat

4.2.3.3.1 Solar Zenith Angle Calculation

Shown below in Figure 16, is the schematic of the Solar Zenith Angle, ξ, which is the angle between vector from the center of the Earth to the spacecraft and the vector from the center of the Earth to the center of the Sun.

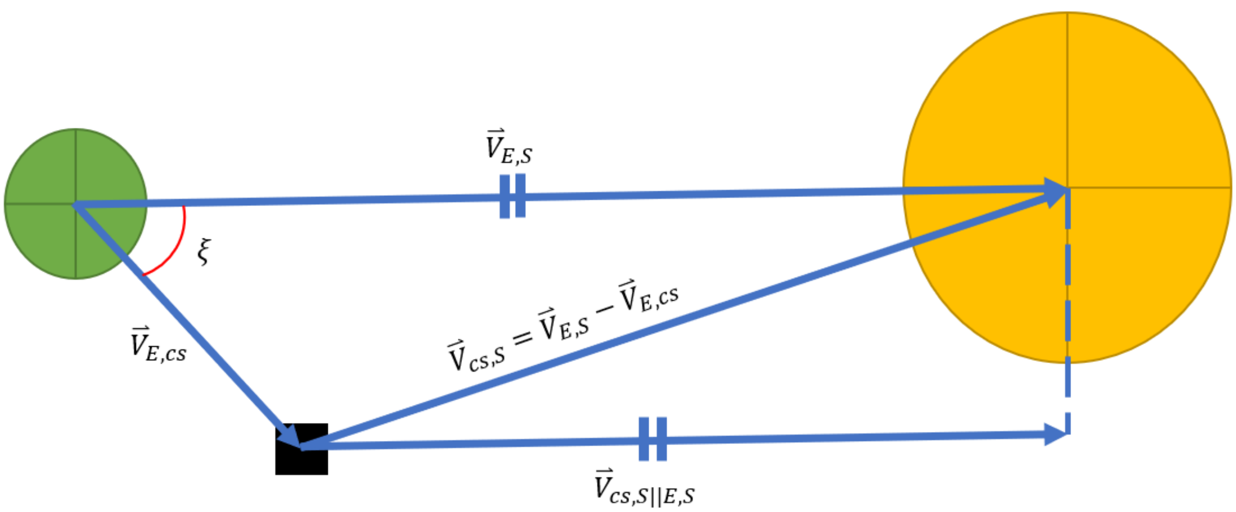


Figure 16: Solar Zenith Angle Schematic

When $-90° \leq \xi \leq 90°$, i.e., if the magnitude is less than or equal to $90°$, the spacecraft does not experience eclipse, and therefore, undergoes full exposure to direct solar and partial exposure to Earth albedo radiation. The vector from the Earth to the Sun, $\vec{V}_{E,S}$, and the vector from the Earth to the spacecraft, $\vec{V}_{E,cs}$, were given from the STK data and the vector from spacecraft to the center of the Sun, $\vec{V}_{cs,S}$, is found with simple vector subtraction shown above in the figure. To determine if the spacecraft is behind the Earth relative to the Sun, the vector component of $\vec{V}_{cs,S}$ in a direction parallel to $\vec{V}_{E,S}$ can be computed and if that component is longer than $\vec{V}_{E,S}$, then it the spacecraft is, in fact, behind the Earth relative to the Sun. The equation for the parallel component is shown below in (Eq. 18.

$$\vec{V}_{cs,S \vee ¿ E,S} = \frac{\vec{V}_{cs,S} \cdot \vec{V}_{E,S}}{\left|\vec{V}_{E,S}\right|^2} \hat{V}_{E,S} \qquad \textit{(Eq. 18)}$$

which expands to

$$\vec{V}_{cs,S \vee ¿ E,S} = \frac{\vec{V}_{cs,S} \cdot \vec{V}_{E,S}}{\left|\vec{V}_{E,S}\right|^2} \left\langle \frac{\vec{V}_{E,S}(1)}{\left|\vec{V}_{E,S}\right|}, \frac{\vec{V}_{E,S}(2)}{\left|\vec{V}_{E,S}\right|}, \frac{\vec{V}_{E,S}(3)}{\left|\vec{V}_{E,S}\right|} \right\rangle$$

If $¿ \vec{V}_{cs,S \vee ¿ E,S} \vee ¿ \vee \vec{V}_{E,S} \vee ¿$ then the spacecraft is behind the Earth relative to the Sun.

4.2.3.3.2 Earth Albedo Heat Flux

The calculation for Earth albedo heat flux is shown in (Eq. 19 where Q''_{Albedo} represents the heat flux from the Sun reflected by the Earth at a given distance in W/m^2. Q''_{solar} is the heat flux from the Sun at a given distance in in W/m^2 (calculated in section 3.1), aF is the Earth albedo factor, and ξ is the solar zenith angle.

$$Q''_{Albedo} = \frac{Q''_{solar}}{\left(\frac{D_{orbit} + R_{Earth}}{R_{Earth}}\right)^2} \cdot \alpha \cdot aF \cdot \cos(\xi) \qquad \text{(Eq. 19)}$$

4.2.3.3.3 Earth Albedo Angle of Incidence

4.2.3.3.4 Earth Albedo Projected Area

The Earth Albedo Projected Area is the area of a spacecraft face which is receiving radiation. It is dependent on the area of the whole face then the angle of incidence of each component normal to the face that is receiving radiation. It is assumed that the faces of the spacecraft are flat and rectangular. The total projected area for a face given by Earth Albedo is given by

$$A_{Albedo, projected} = (L_{CS} * w_{CS}) * \cos(\theta_x) * \cos(\theta_y) * \cos(\theta_z) \qquad \textit{(Eq. 20)}$$

Where L_{CS} is the length of the spacecraft face, w_{CS} is the width of the spacecraft face, and θ is the component angle of the Earth Albedo Angle of Incidence.

4.2.3.3.5 Earth Albedo Effective Absorptivity

4.2.3.3.6 Earth Albedo Absorbed Heat

Earth Albedo Absorbed Heat per face can be calculated by using

$$Q_{Albedo} = Q''_{Albedo} * A_{Albedo, projected} * \left| ¿_{Albedo} \right| ¿ \qquad \textit{(Eq. 21)}$$

4.2.3.4 Absorbed Free Molecular Heating (FMH)

Free molecular heating is a result of atmospheric drag. When our satellite is travelling at high velocities, it experiences particle collisions in the atmosphere. These collisions transfer kinetic energy to the satellite, which results in a resulting heat flux. Free molecular heating is dependent on:

- Atmospheric density
- Satellite distance to Earth
- Thermal accommodation coefficient
- Satellite velocity
- Free Molecular Projected Spacecraft Area

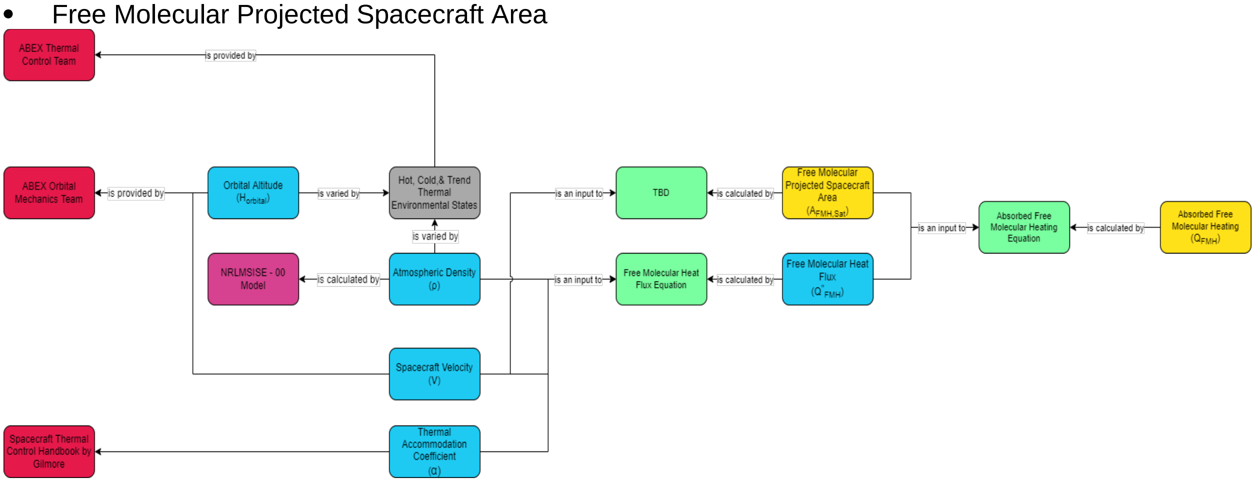


Figure 17: Absorbed Free Molecular Heating

4.2.3.4.1 Atmospheric Density

The atmospheric density was calculated using the NRLMSISE-00 model. This model takes inputs of satellite longitude, latitude, and altitude, as well as day of the year. The model output is atmospheric density and is fairly accurate, although not exact.

4.2.3.4.2 Free Molecular Heat Flux

The calculation for free molecular heat flux is shown in Equation 24, where Q''_{FMH} represents the heat flux due to atmospheric drag in W/m². V is the satellite velocity and is obtained by our orbital team using the Vis-Viva equation, ρ is the atmospheric density calculated by the NRLMSISE-00 model, and α is the thermal accommodation coefficient (unity in this analysis for conservatism).

$$Q''_{FMH} = \alpha \cdot \left(\frac{1}{2}\right) \cdot \rho \cdot V^3 \qquad \textit{(Eq. 22)}$$

4.2.3.4.3 Free Molecular Projected Area

Free Molecular Projected Area is calculated by the face which is in line with the velocity vector of the spacecraft.

4.2.3.4.4 Absorbed Free Molecular Heating

Absorbed Free Molecular Absorbed Heat can be calculated using the equation below.

$$Q_{FMH} = Q^{\text{}} \text{ rsub \{FMH\} * \{A\} rsub \{FMH, projected¿}} ¿ \qquad \textit{(Eq. 23)}$$

4.2.3.5 Charged Particle Heating (CPH)

Radiation absorbed by the external walls and internal components of the spacecraft. charged particle heating in space comes from four sources: Van Allen Belt trapped protons, Van Allen Belt trapped electrons, solar energetic particles, and galactic cosmic rays. While the combined fluxes from these four sources is low, radiation shielding must be used to counter the high energy waves of the four sources (especially galactic cosmic rays). The breakdown of how to calculate the flux of each source is detailed in supporting Radiation Analysis Plan document.

4.2.3.5.1 CPH Heat Flux

Charged particle heat flux is the sum of the heat fluxes of each of the four sources detailed in the Radiation Analysis Plan. The equation below can be used to calculate CPH Heat Flux once the fluxes of the four sources are known.

$$Q''_{CPH} = Q''_{VAB,TP} + Q''_{VAB,TE} + Q''_{SEP} + Q''_{GCR} \quad \textit{(Eq. 24)}$$

4.2.3.5.2 Absorbed CPH

To calculate CPH heat absorbed, the CPH heat flux is multiplied by the projected area for CPH shown in the equation below. The projected area for CPH is the area of each of the six spacecraft faces, since CPH heat flux comes in from every direction orthogonally to the spacecraft faces.

$$Q_{CPH} = Q^{\text{\}\} rsub \{CPH\} * \{A\} rsub \{CPH, projected¿}} ¿ \quad \textit{(Eq. 25)}$$

4.3 Implemented Patch Heater Wattage

The thermal environment in space is extremely complex, as radiation is incident on the spacecraft at many different locations from constantly changing angles at any given time. Due to this complexity, on-board solutions for both heating and cooling of the spacecraft must be plausible. While the thermal environment is not likely to reside at the intensive ends of its complicated spectrum, it is necessary to outfit the satellite with technologies that can handle these extremes and preserve component life.

Section 4.3 will focus on the calculation and deliberation leading to the selection of a required patch heater wattage, beginning with an isothermal approximation performed in MATLAB that will inform a transient, quasi-3D model constructed in Simulink that considers the relationship between patch heater wattage and radiator area. This Simulink model will then provide plausible combinations of the two to a Thermal Desktop model, which will then perform a transient, spatially accurate simulation to, once again, narrow down effective combinations and eventually select an implemented solution. However, this section will utilize a top-down approach, and will start with the final Implemented Patch Heater Wattage value. It is also worth noting that DKMs in this section will be highly similar to those in the Implemented Radiator Area section because the two are intricately tied. Components of each will even be included in DKMs of the other.

<table>
<tr><td>Min Thermal Condition
4.3.1.1.1.1.1</td><td rowspan="3">Isothermal Patch Heater Wattage
4.3.1.1.1.1</td><td rowspan="10">Simulink Required Patch Heater Wattage Range
4.3.1.1.1</td><td rowspan="10">Thermal Desktop Patch Heater Wattage Range
4.3.1.1</td><td rowspan="10">Implemented Patch Heater Wattage
4.3.1</td></tr>
<tr><td>Operational Heat Generator
4.3.1.1.1.1.2</td></tr>
<tr><td>Radiation Out
4.3.1.1.1.1.3</td></tr>
<tr><td>Max Thermal Condition
4.3.1.1.1.2.1</td><td rowspan="3">Isothermal Radiator Area
4.3.1.1.1.2</td></tr>
<tr><td>Operational Heat Generation
4.3.1.1.1.2.2</td></tr>
<tr><td>Radiation Out
4.3.1.1.1.2.3</td></tr>
<tr><td colspan="2">Thermal Resistances
4.3.1.1.1.3</td></tr>
<tr><td colspan="2">Thermal Contact Resistances
4.3.1.1.1.4</td></tr>
<tr><td colspan="2">Environmental Parameters
4.3.1.1.1.5</td></tr>
<tr><td colspan="2">Component Parameters
4.3.1.1.1.6</td></tr>
</table>

Figure 18: Implemented Patch Heater Wattage Knowledge Point Breakdown

4.3.1 Implemented Patch Heater Wattage

4.3.1.1 Thermal Desktop Patch Heater Wattage Range

4.3.1.1.1 Simulink Required Patch Heater Wattage Range

The Simulink model is not as computationally intensive as the Thermal Desktop Model but is still able to provide a high level of granularity compared to the isothermal MATLAB model. The DKM describing this model and its breakdown are below.

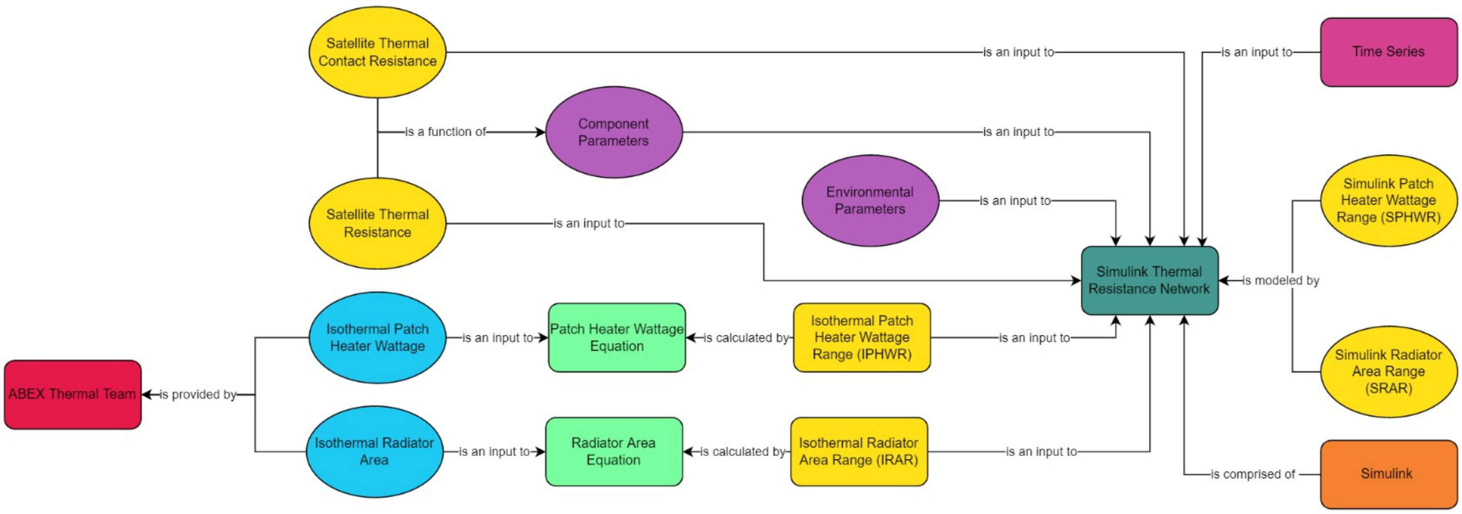


Figure 19: Overall Simulink DKM

As shown in the DKM above, the non-isothermal required patch heater wattage is calculated in tandem with the radiator area, since the heat out from the radiator must be considered in the model. The Simulink required patch heater wattage range is the array of values that will be tested using the brute force method in the Simulink model. It is a function of the isothermal required patch heater wattage and consists of incremental values between$Q_{PH,min}$ and $Q_{PH,max}$, which is calculated from Equation 37. Each value in this range is simulated in the model with every combination of values in the Simulink required radiator area range, and the combinations that result in component temperatures that remain within their respective requirements will be returned as viable combinations.

$$Q_{PH,min}=0$$

$$Q_{PH,max}=2*Q_{PH,isothermal} \qquad 26$$

The viable combinations returned by the Simulink models are used as inputs to the Thermal Desktop model. The Thermal Desktop model uses 3D CAD to create more spatially precise temperature distributions that can used to validate the results of the Simulink simulations. Analysis is done utilizing FEA within Thermal Desktop. The patch heater wattages in the combinations that result in the satellite components to remain within temperature boundaries in the Thermal Desktop model as well, will make up the Thermal Desktop Required patch heater wattage range.

4.3.1.1.1.1 Isothermal Required Patch Heater Wattage

This DKM is based entirely on an isothermal energy balance in MATLAB. This energy balance accounts for incoming heat, operationally generated heat, the minimum satellite temperature, and radiation out. In this case, the energy balance solves for its singular unknown: patch heater wattage necessary to maintain a certain temperature.

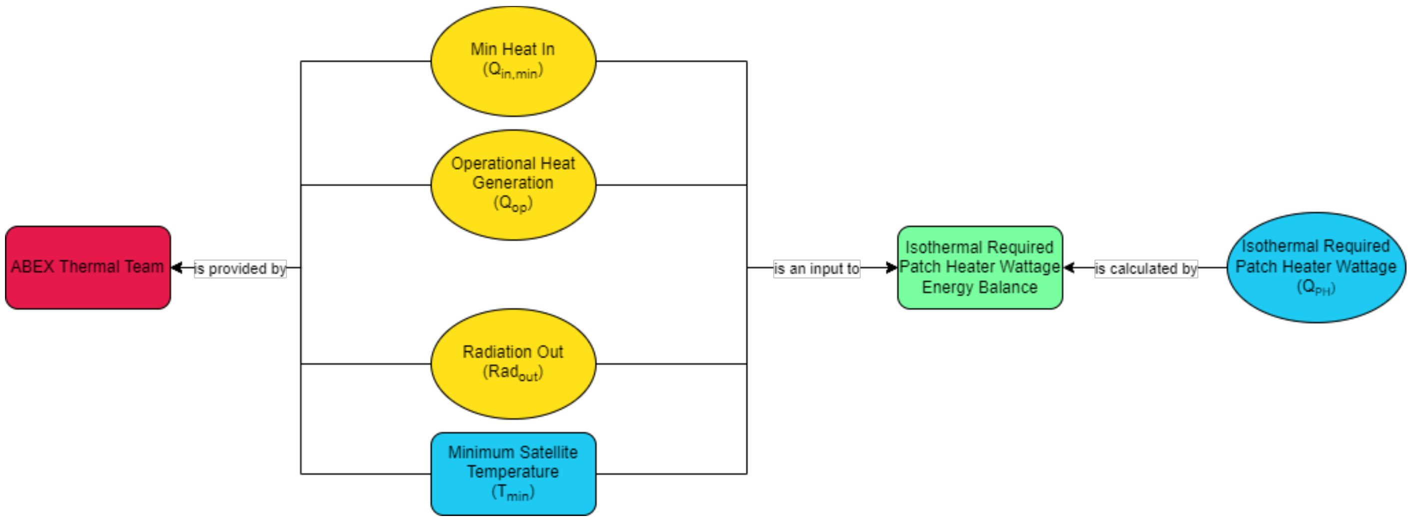


Figure 20: Isothermal Required Patch Heater Wattage

After acquiring each of these variables, they can be input into an isothermal energy balance.

The estimated patch heater wattage is calculated in the MATLAB code. The patch heater wattage was estimated for each 5-minute orbit time step, at each permutation, for each TES (hot, cold, trend). The patch heater is set to only turn on if the isothermal temperature is less than 0°C. A new satellite temperature, T_{sat} was prescribed for this calculation.

$$T_{ph}=0\,℃ \tag{27}$$

The patch heater wattage was solved using the same energy equation used to estimate the temperature, but with Q_{PH} acting as the unknown instead of T.

The general energy equation is:

$$E_{¿}-E_{out}=0 \tag{28}$$

where $E_{¿}$ is the energy into the system, which is the sum of all heating due to the external and internal heat sources. E_{out} is the energy out of the system, calculated using the Stefon-Boltzmann equation of radiation.

$E_{¿}$ can be solved for using below,

$$E_{¿}=Q_{¿}+Q_{PH}+Q_{op} \quad (29)$$

where $Q_{¿}$ is the total external heat that enters the system, Q_{op} is the operational internal heat generation, and Q_{ph} is the heat generation from flexible patch heaters attached to the satellite (what we are solving for).

The equation for E_{out} was split up by face. Two deployable radiators will be installed on the +Y and -Y face and need to be accounted for in the E_{out} calculation.

Since the radiator area is unknown and dependent on satellite temperature and patch heater wattage, a reasonable area was used in this calculation. For the temperature estimation calculation and patch heater wattage calculation only,

$$A_{rad}=0.02\,m^2$$

4.3.1.1.1.1.1 Min Thermal Condition

This Knowledge Point DKM is shown in Figure 4.

4.3.1.1.1.1.2 Operational Heat Generation

This Knowledge Point DKM is shown in Figure 5.

4.3.1.1.1.1.3 Radiation Out

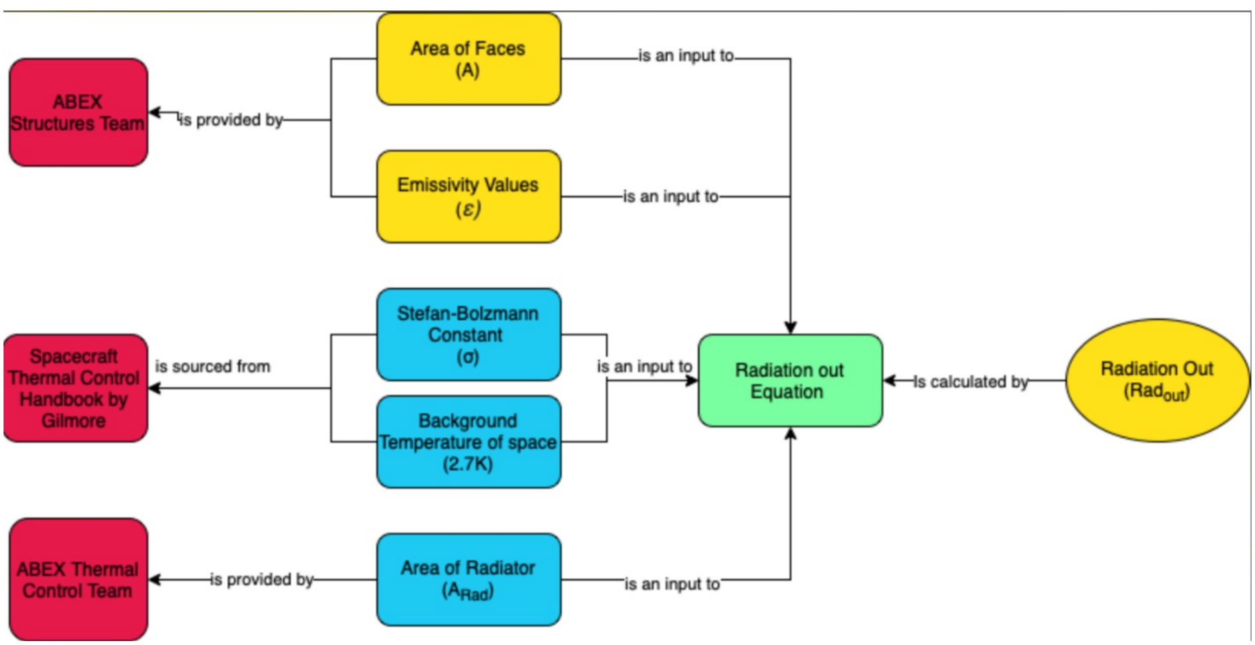


Figure 21: Radiation Out DKM

The E_{out} equations are shown below,

$$Rad_{out,X+¿} = \varepsilon_{X+¿} \cdot \sigma \cdot \left(T_{ph}^4 - T_{sur}^4\right) \cdot A_{X+¿¿} ¿ \quad 31$$

$$Rad_{out,X-¿} = \varepsilon_{X-¿} \cdot \sigma \cdot \left(T_{ph}^4 - T_{sur}^4\right) \cdot A_{X-¿¿} ¿ \quad 32$$

$$Rad_{out,Z+¿} = \varepsilon_{Z+¿} \cdot \sigma \cdot \left(T_{ph}^4 - T_{sur}^4\right) \cdot A_{Z+¿¿} ¿ \quad 33$$

$$Rad_{out,Z-¿} = \varepsilon_{Z-¿} \cdot \sigma \cdot \left(T_{ph}^4 - T_{sur}^4\right) \cdot A_{Z-¿¿} ¿ \quad 30$$

$$Rad_{out,Yboth} = ¿¿ \quad 31$$

where σ is the Steffon-Boltzmann constant, $T_{ph}$is the heating temperature of the patch heater (set to a constant 0°C), T_{sur} is the ambient temperature of space, and $A_{x+¿¿}$ is the total area of the X+ side, $A_{x-¿¿}$ is the total area of the X- side, $A_{y+¿¿}$ is the total area of the Y+ side not including the radiator, $A_{y-¿¿}$ is the total area of the Y- side not including the radiator, $A_{z+¿¿}$ is the total area of the Z+ side, $A_{z-¿¿}$ is the total area of the Z- side, A_{rad} is the area of the radiator, $\varepsilon_{X+¿¿}$ is the emissivity of the X+ side, $\varepsilon_{X-¿¿}$ is the emissivity of the X- side, $\varepsilon_{Y+¿¿}$ is the emissivity of the Y+ side, $\varepsilon_{y-¿¿}$ is the emissivity of the Y+ side, $\varepsilon_{z+¿¿}$ is the emissivity of the Z+ side, $\varepsilon_{z-¿¿}$ is the emissivity of the Z- side, and ε_{rad} is the emissivity of the radiator

Solving for "Q_{PH}" in MATLAB does not require a root finding method or equation solver, since once simplified, there is only one Q_{PH} term.

Q_{PH} was solved for 2 cases: with solar arrays deployed and assuming the solar array area in the emissivity term is 0.

Update

Figure 22: Patch Heater Wattage per Permutation with Arrays Deployed

Update

Figure 23: Patch Heater Wattage per Permutation Assuming no Array Area

As the analysis becomes more complex, the isothermal assumption will be discarded due to its inaccuracy and a non-isothermal model will be used. To make the bridge from an isothermal model to a more detailed model Simulink and Thermal Desktop will be used.

4.3.1.1.1.2 Isothermal Required Radiator Area

This DKM is based entirely on an isothermal energy balance in MATLAB. This energy balance accounts for incoming heat, operationally generated heat, the maximum satellite temperature, and radiation out. In this case, the energy balance solves for its singular unknown: radiator area necessary to maintain a certain temperature. A patch heater wattage is assumed in this calculation.

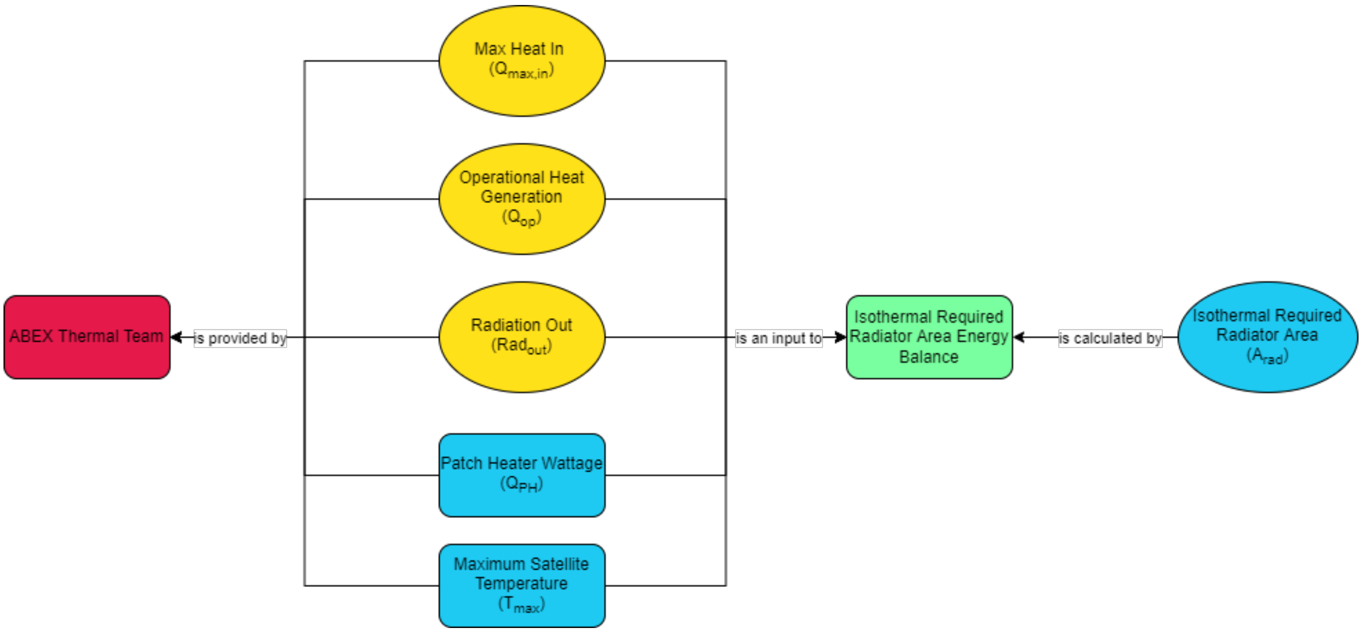


Figure 24: Isothermal Required Radiator Area DKM

The purpose of this DKM is to compute the radiator area needed for the CubeSat to have a temperature <= 40°C. This area was calculated for the hot, cold, and trend TES at each permutation.

The equation to calculate radiator area is identical to the temperature equation but rearranged to solve for the area of the radiator. This general equation is shown below,

$$E_{¿} - E_{out} = 0 \qquad 38$$

where $E_{¿}$ is the radiation entering the system and E_{out} is the radiation leaving the system

Note that there will be two radiators present (on the +Y and -Y faces), which has a different emissivity than the Aluminum-6061 panel. To account for this emissivity difference, the “radiation out” term was split up by face. The radiation in term was calculated the same way as in the Isothermal Required Patch Heater Wattage Section. The radiation in equation is shown below.

$$Rad_{¿} = Q_{¿} + Q_{PH} + Q_{op} \qquad 39$$

4.3.1.1.1.2.1 Max Thermal Condition

This Knowledge Point DKM is shown in Figure 4.

4.3.1.1.1.2.2 Operational Heat Generation

This Knowledge Point DKM is shown in Figure 5.

4.3.1.1.1.2.3 Radiation Out

This DKM is the same as shown in the previous Radiation Out Section, however, the energy balance now solves for the radiator area. The following equations encompass the radiation exiting the spacecraft.

$$Rad_{out,X+¿} = \varepsilon_{X+¿} \cdot \sigma \cdot \left(T_{rad}^4 - T_{sur}^4\right) \cdot A_{X+¿¿}¿ \quad (40)$$

$$Rad_{out,X-¿} = \varepsilon_{X-¿} \cdot \sigma \cdot \left(T_{rad}^4 - T_{sur}^4\right) \cdot A_{X-¿¿}¿ \quad (41)$$

$$Rad_{out,Z+¿} = \varepsilon_{Z+¿} \cdot \sigma \cdot \left(T_{rad}^4 - T_{sur}^4\right) \cdot A_{Z+¿¿}¿ \quad (42)$$

$$Rad_{out,Z-¿} = \varepsilon_{Z-¿} \cdot \sigma \cdot \left(T_{rad}^4 - T_{sur}^4\right) \cdot A_{Z-¿¿}¿ \quad (43)$$

$$Rad_{out,Yboth} = ¿¿ \quad (32)$$

These equations were combined and the equation solver in MATLAB was used to solve for A_{rad}. A_{rad} was solved for 2 cases: with solar arrays deployed and with solar arrays stashed.

Will be updated by 8/6 when final thermal management plan is due

Figure: Radiator Area per Permutation with Arrays Deployed

Will be updated by 8/6 when final thermal management plan is due

4.3.1.1.1.3 Thermal Resistances

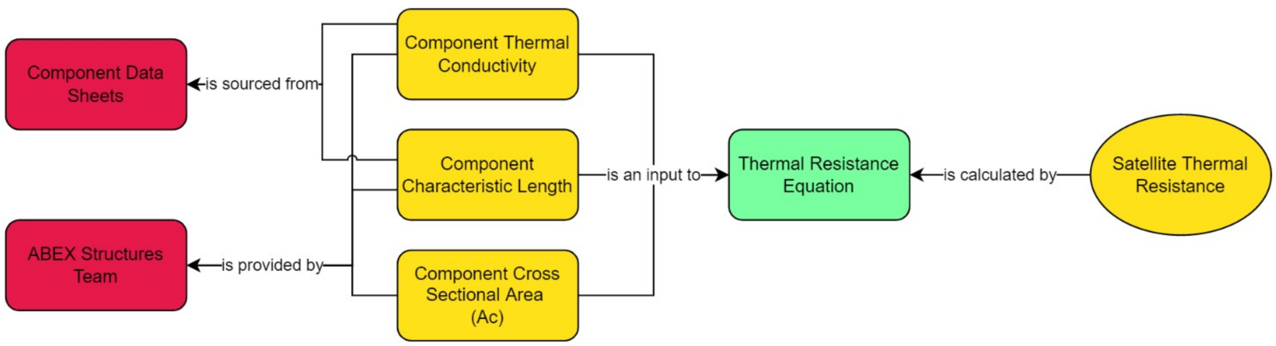


Figure 25: Thermal Resistance DKM

4.3.1.1.1.4 Thermal Contact Resistances

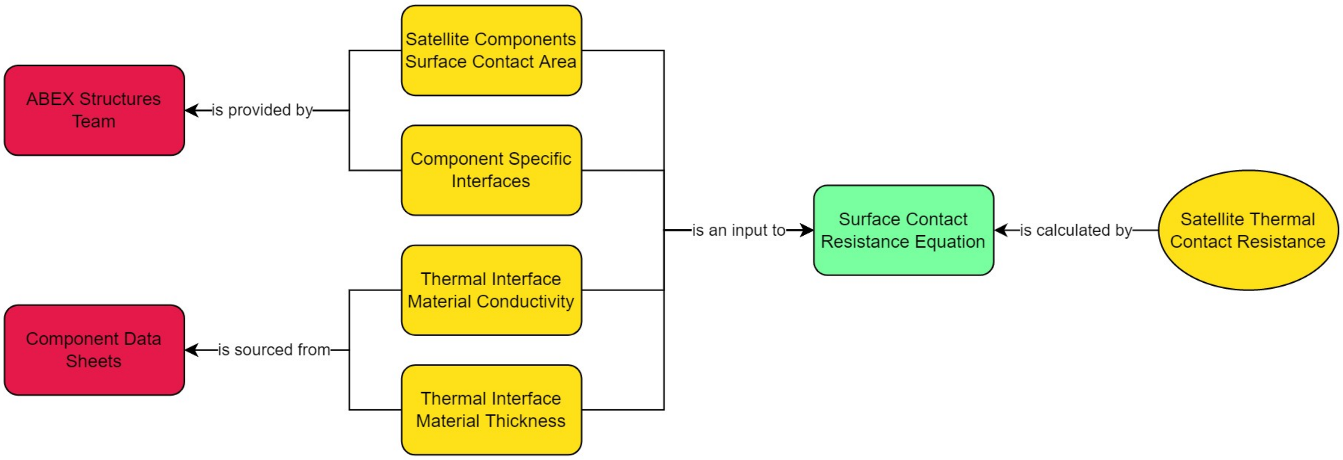


Figure 26: Thermal Contact Resistance DKM

4.3.1.1.1.5 Environmental Parameters

4.3.1.1.1.6 Component Parameters

4.4 Implemented Radiator Area

The Implemented Radiator Area calculation follows an almost identical path compared to the Implemented Patch Heater Wattage calculation, but involves the maximum thermal environmental state as opposed to the minimum. While the two are similar, the different strategies will be detailed, and the similarities will be restated.

<table>
<tr><td>Min Thermal Condition 4.4.1.1.1.1.1</td><td rowspan="3">Isothermal Patch Heater Wattage 4.4.1.1.1.1</td><td rowspan="10">Simulink Required Patch Heater Wattage Range 4.4.1.1.1</td><td rowspan="10">Thermal Desktop Patch Heater Wattage Range 4.4.1.1</td><td rowspan="10">Implemented Patch Heater Wattage 4.4.1</td></tr>
<tr><td>Operational Heat Generator 4.4.1.1.1.1.2</td></tr>
<tr><td>Radiation Out 4.4.1.1.1.1.3</td></tr>
<tr><td>Max Thermal Condition 4.4.1.1.1.2.1</td><td rowspan="3">Isothermal Radiator Area 4.4.1.1.1.2</td></tr>
<tr><td>Operational Heat Generation 4.4.1.1.1.2.2</td></tr>
<tr><td>Radiation Out 4.4.1.1.1.2.3</td></tr>
<tr><td colspan="2">Thermal Resistances 4.4.1.1.1.3</td></tr>
<tr><td colspan="2">Thermal Contact Resistances 4.4.1.1.1.4</td></tr>
<tr><td colspan="2">Environmental Parameters 4.4.1.1.1.5</td></tr>
<tr><td colspan="2">Component Parameters 4.4.1.1.1.6</td></tr>
</table>

4.4.1 Implemented Radiator Area

4.4.1.1 Thermal Desktop Radiator Area Range

4.4.1.1.1 Simulink Required Radiator Area Range

The Simulink model is not as computationally intensive as the Thermal Desktop Model but is still able to provide a high level of granularity compared to the isothermal MATLAB model. The DKM describing this model and its breakdown are below.

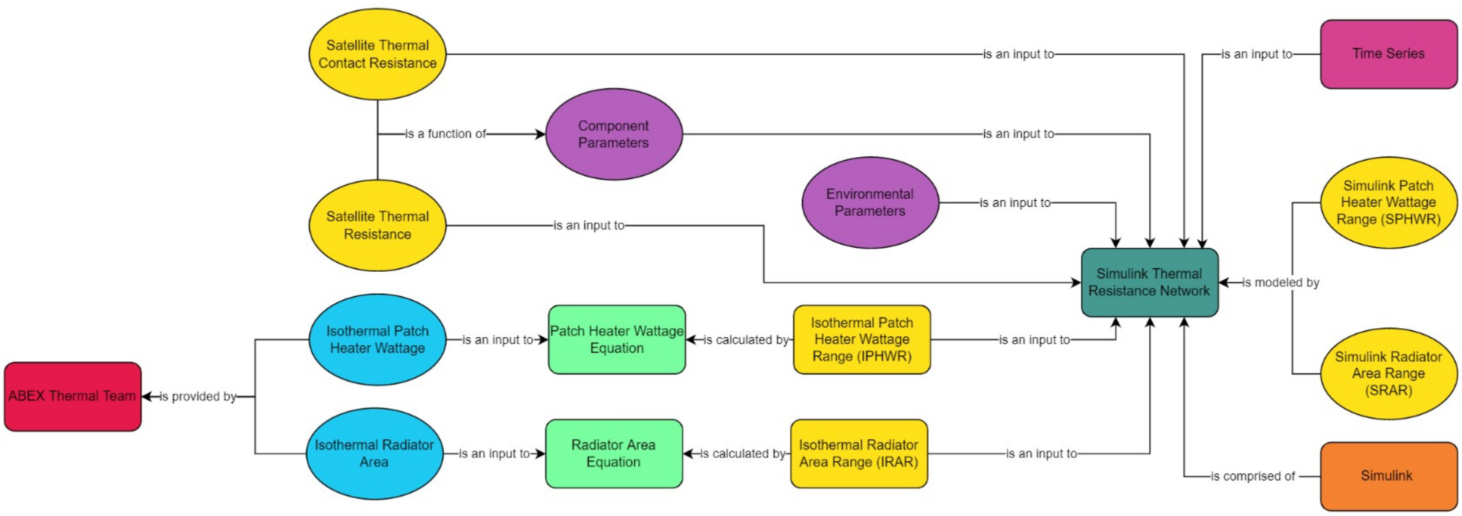


Figure 27: Overall Simulink DKM

As shown in the DKM above, the non-isothermal required radiator area is calculated in tandem with the patch heater wattage, since the heat generated by the patch heaters must be considered in the model. The Simulink required radiator area range is the array of values that will be tested using the brute force method in the Simulink model. It is a function of the isothermal required radiator area and consists of incremental values between $A_{rad,min}$ and $A_{rad,max}$, which is calculated from Equation 37. Each value in this range is simulated in the model with every combination of values in the Simulink required patch heater wattage range, and the combinations that result in component temperatures that remain within their respective requirements will be returned as viable combinations.

$$A_{rad,max} = 2 * A_{rad,isothermal} \quad (33)$$

The viable combinations returned by the Simulink models are used as inputs to the Thermal Desktop model. The Thermal Desktop model uses 3D CAD to create more spatially precise temperature distributions that can used to validate the results of the Simulink simulations. Analysis is done utilizing FEA within Thermal Desktop. The radiator areas in the combinations that result in the satellite components to remain within temperature boundaries in the Thermal Desktop model as well, will make up the Thermal Desktop Required radiator area range.

4.4.1.1.1.1 Isothermal Required Patch Heater Wattage

This DKM is based entirely on an isothermal energy balance in MATLAB. This energy balance accounts for incoming heat, operationally generated heat, the minimum satellite temperature, and radiation out. In this case, the energy balance solves for its singular unknown: patch heater wattage necessary to maintain a certain temperature.

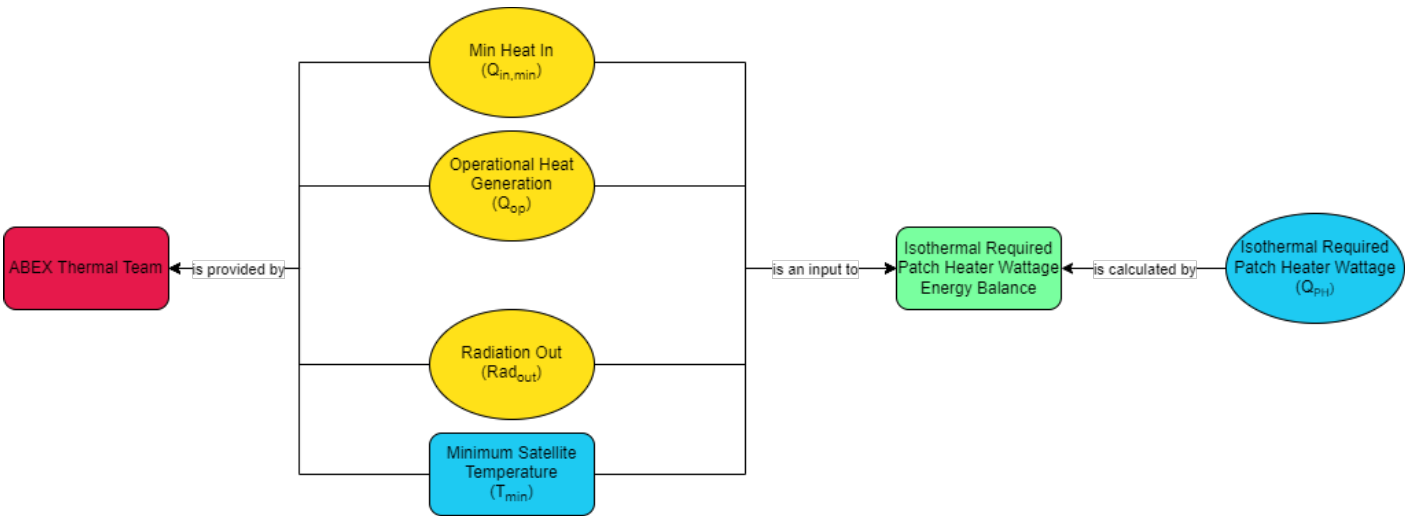


Figure 28: Isothermal Required Patch Heater Wattage

After acquiring each of these variables, they can be input into an isothermal energy balance.

The estimated patch heater wattage is calculated in the MATLAB code. The patch heater wattage was estimated for each 5-minute orbit time step, at each permutation, for each TES (hot, cold, trend). The patch heater is set to only turn on if the isothermal temperature is less than 0°C. A new satellite temperature, T_{sat} was prescribed for this calculation.

$$T_{ph}=0\,℃ \tag{34}$$

The patch heater wattage was solved using the same energy equation used to estimate the temperature, but with Q_{PH} acting as the unknown instead of T.

The general energy equation is:

$$E_{¿}-E_{out}=0 \tag{35}$$

where $E_{¿}$ is the energy into the system, which is the sum of all heating due to the external and internal heat sources. E_{out} is the energy out of the system, calculated using the Stefon-Boltzmann equation of radiation.

$E_{¿}$ can be solved for using below,

$$E_{¿} = Q_{¿} + Q_{PH} + Q_{op} \qquad \textbf{36}$$

where $Q_{¿}$ is the total external heat that enters the system, Q_{op} is the operational internal heat generation, and Q_{ph} is the heat generation from flexible patch heaters attached to the satellite (what we are solving for).

The equation for E_{out} was split up by face. Two deployable radiators will be installed on the +Y and -Y face and need to be accounted for in the E_{out} calculation.

Since the radiator area is unknown and dependent on satellite temperature and patch heater wattage, a reasonable area was used in this calculation. For the temperature estimation calculation and patch heater wattage calculation only,

$$A_{rad} = 0.02 m^2$$

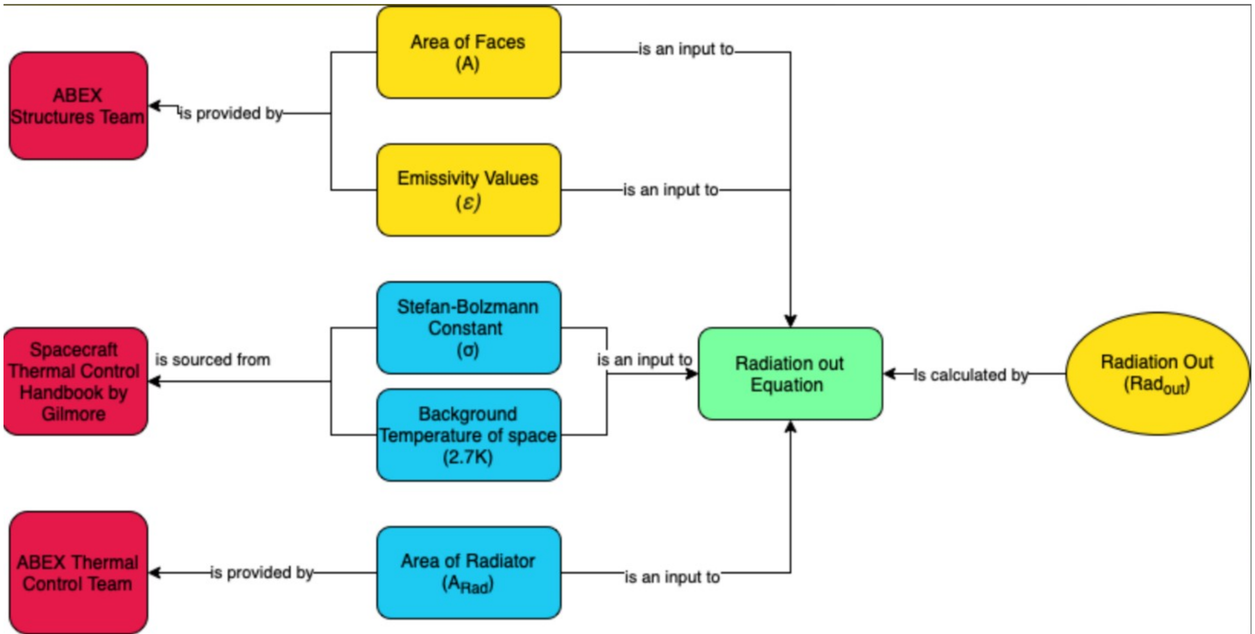
4.4.1.1.1.1.1 Min Thermal Condition

This Knowledge Point DKM is shown in Figure 4.

4.4.1.1.1.1.2 Operational Heat Generation

This Knowledge Point DKM is shown in Figure 5.

4.4.1.1.1.1.3 Radiation Out



The E_{out} equations are shown below,

$$Rad_{out,X+¿} = \varepsilon_{X+¿} \cdot \sigma \cdot \left(T_{ph}^4 - T_{sur}^4\right) \cdot A_{X+¿¿}¿ \quad 31$$

$$Rad_{out,X-¿} = \varepsilon_{X-¿} \cdot \sigma \cdot \left(T_{ph}^4 - T_{sur}^4\right) \cdot A_{X-¿¿}¿ \quad 32$$

$$Rad_{out,Z+¿} = \varepsilon_{Z+¿} \cdot \sigma \cdot \left(T_{ph}^4 - T_{sur}^4\right) \cdot A_{Z+¿¿}¿ \quad 33$$

$$Rad_{out,Z-¿} = \varepsilon_{Z-¿} \cdot \sigma \cdot \left(T_{ph}^4 - T_{sur}^4\right) \cdot A_{Z-¿¿}¿ \quad 37$$

$$Rad_{out,Yboth} = ¿¿ \quad 38$$

where σ is the Steffon-Boltzmann constant, $T_{ph}$is the heating temperature of the patch heater (set to a constant 0°C), T_{sur} is the ambient temperature of space, and $A_{x+¿¿}$ is the total area of the X+ side, $A_{x-¿¿}$ is the total area of the X- side, $A_{y+¿¿}$ is the total area of the Y+ side not including the radiator, $A_{y-¿¿}$ is the total area of the Y- side not including the radiator, $A_{z+¿¿}$ is the total area of the Z+ side, $A_{z-¿¿}$ is the total area of the Z- side, A_{rad} is the area of the radiator, $\varepsilon_{X+¿¿}$ is the emissivity of the X+ side, $\varepsilon_{X-¿¿}$ is the emissivity of the X- side, $\varepsilon_{Y+¿¿}$ is the emissivity of the Y+ side, $\varepsilon_{y-¿¿}$ is the emissivity of the Y+ side, $\varepsilon_{z+¿¿}$ is the emissivity of the Z+ side, $\varepsilon_{z-¿¿}$ is the emissivity of the Z- side, and ε_{rad} is the emissivity of the radiator

Solving for "Q_{PH}" in MATLAB does not require a root finding method or equation solver, since once simplified, there is only one Q_{PH} term.

Q_{PH} was solved for 2 cases: with solar arrays deployed and assuming the solar array area in the emissivity term is 0.

Update

Figure 30: Patch Heater Wattage per Permutation with Arrays Deployed

Update

Figure 31: Patch Heater Wattage per Permutation Assuming no Array Area

As the analysis becomes more complex, the isothermal assumption will be discarded due to its inaccuracy and a non-isothermal model will be used. To make the bridge from an isothermal model to a more detailed model Simulink and Thermal Desktop will be used.

4.4.1.1.1.2 Isothermal Required Radiator Area

This DKM is based entirely on an isothermal energy balance in MATLAB. This energy balance accounts for incoming heat, operationally generated heat, the maximum satellite temperature, and radiation out. In this case, the energy balance solves for its singular unknown: radiator area necessary to maintain a certain temperature. A patch heater wattage is assumed in this calculation.

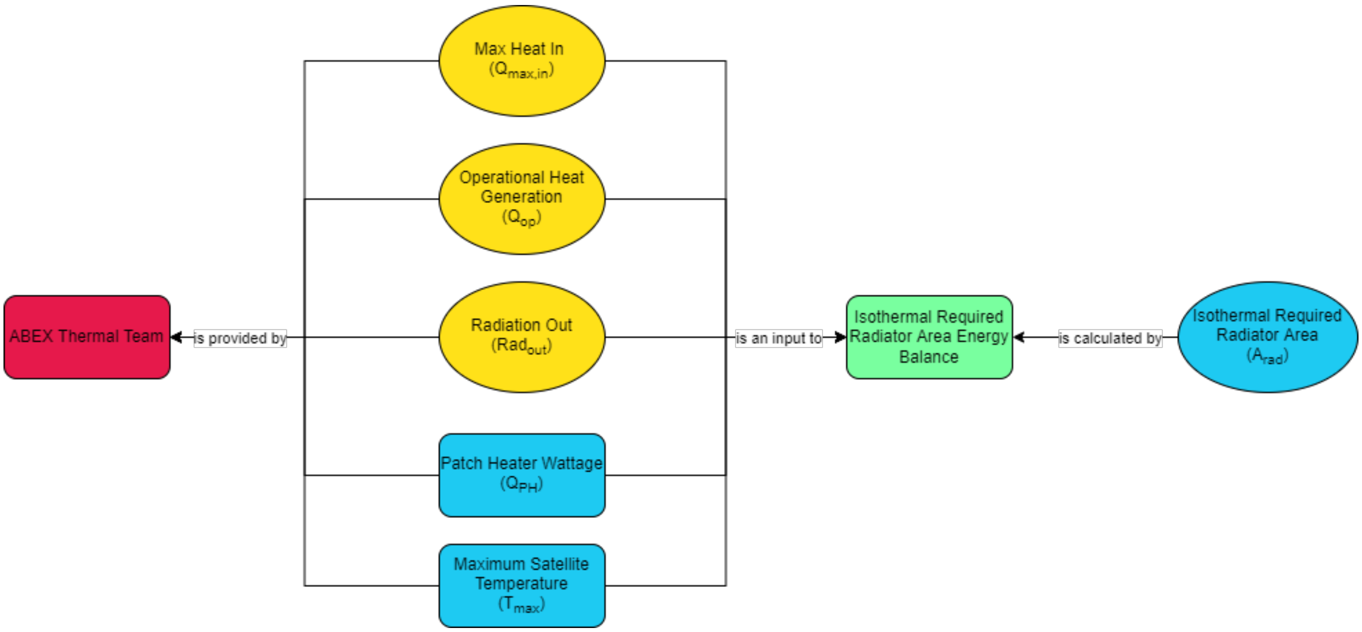


Figure 32: Isothermal Required Radiator Area DKM

The purpose of this DKM is to compute the radiator area needed for the CubeSat to have a temperature <= 40°C. This area was calculated for the hot, cold, and trend TES at each permutation.

The equation to calculate radiator area is identical to the temperature equation but rearranged to solve for the area of the radiator. This general equation is shown below,

$$E_{¿} - E_{out} = 0 \tag{38}$$

where $E_{¿}$ is the radiation entering the system and E_{out} is the radiation leaving the system

Note that there will be two radiators present (on the +Y and -Y faces), which has a different emissivity than the Aluminum-6061 panel. To account for this emissivity difference, the “radiation out” term was split up by face. The radiation in term was calculated the same way as in the Isothermal Required Patch Heater Wattage Section. The radiation in equation is shown below.

$$Rad_{¿} = Q_{¿} + Q_{PH} + Q_{op} \tag{39}$$

4.4.1.1.1.2.1 Max Thermal Condition

This Knowledge Point DKM is shown in Figure 4.

4.4.1.1.1.2.2 Operational Heat Generation

This Knowledge Point DKM is shown in Figure 5.

4.4.1.1.1.2.3 Radiation Out

This DKM is the same as shown in the previous Radiation Out Section, however, the energy balance now solves for the radiator area. The following equations encompass the radiation exiting the spacecraft.

$$Ra\,d_{out,X+¿}=\varepsilon_{X+¿}\cdot\sigma\cdot\left(T_{rad}^4-T_{sur}^4\right)\cdot A_{X+¿¿}¿¿ \quad (40)$$

$$Ra\,d_{out,X-¿}=\varepsilon_{X-¿}\cdot\sigma\cdot\left(T_{rad}^4-T_{sur}^4\right)\cdot A_{X-¿¿}¿¿ \quad (41)$$

$$Ra\,d_{out,Z+¿}=\varepsilon_{Z+¿}\cdot\sigma\cdot\left(T_{rad}^4-T_{sur}^4\right)\cdot A_{Z+¿¿}¿¿ \quad (42)$$

$$Ra\,d_{out,Z-¿}=\varepsilon_{Z-¿}\cdot\sigma\cdot\left(T_{rad}^4-T_{sur}^4\right)\cdot A_{Z-¿¿}¿¿ \quad (43)$$

$$Ra\,d_{out,Yboth}=¿¿ \quad (39)$$

These equations were combined and the equation solver in MATLAB was used to solve for A_{rad}. A_{rad} was solved for 2 cases: with solar arrays deployed and with solar arrays stashed.

Will be updated by 8/6 when final thermal management plan is due

Figure: Radiator Area per Permutation with Arrays Deployed

Will be updated by 8/6 when final thermal management plan is due

4.4.1.1.1.3 Thermal Resistances

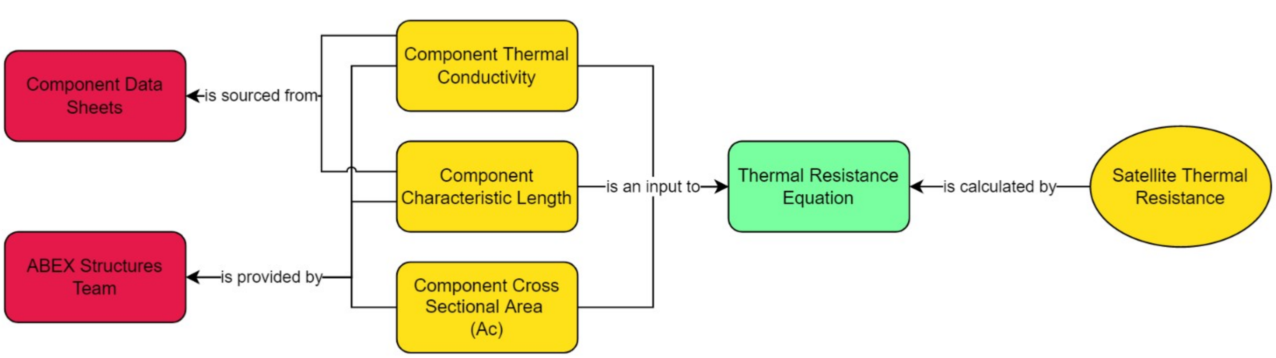


Figure 33: Thermal Resistance DKM

4.4.1.1.1.4 Thermal Contact Resistances

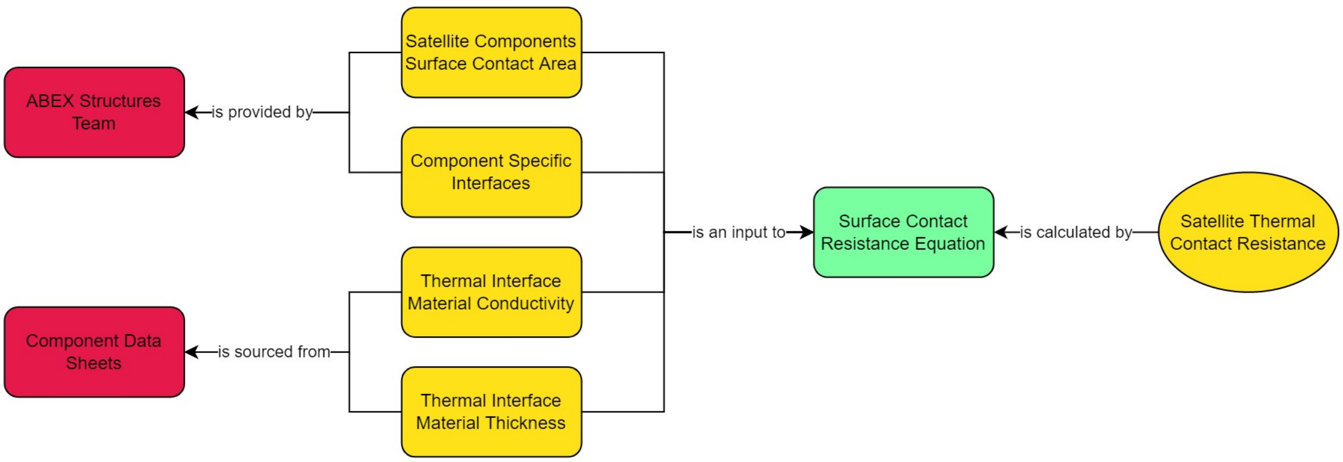


Figure 34: Thermal Contact Resistance DKM

4.4.1.1.1.5 Environmental Parameters

4.4.1.1.1.6 Component Parameters

5 Subsystem Technology Development & Integration

5.1 Overview

The subsystem integration and testing plan is intended to mature the subsystem from its current TRL of [insert value here] to a TRL of 5. This process will involve the development of hardware that demonstrates the system in both its function and its function in the relevant environment, including the flight system to a level ready for qualification testing or acceptance testing, which is in the subsystem Qualification Plan and Acceptance Plan respectively.

5.2 Technology Advancement Plan

Describe here the general advancement plan as context for the table below.

For guidance on testing types and TRL levels see Appendix E.

Table 9: Description of Remaining Technology Steps and Testing Goals to be Accomplished

TRL Step	Development (Type I) Testing Goals		
	Functional Tests	Integration Tests	Environmental Test

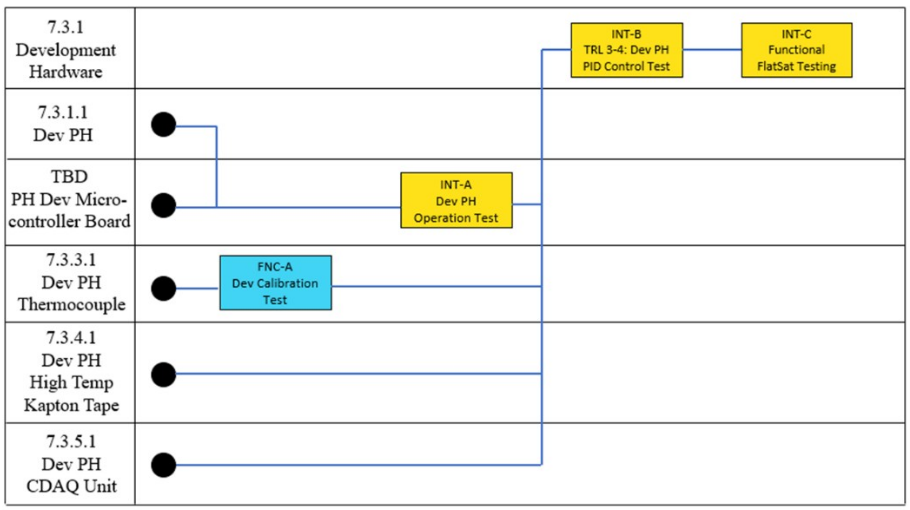
3 – 4	• Bulleted list of goals • Bulleted list of goals	• Bulleted list of goals • Bulleted list of goals	• Bulleted list of goals • Bulleted list of goals
4 – 5	• Bulleted list of goals • Bulleted list of goals	• Bulleted list of goals • Bulleted list of goals	• Bulleted list of goals • Bulleted list of goals

5.3 Development Integration Flow

To bring the satellite to a TRL level of 5, there must be several theoretical and then functional tests. To begin the testing procedure, a component must be raised to level 3. This level corresponds to analytical and experimental critical function testing. This would look like running a computer simulation that is verified and receiving positive results. For a Thermal Team, this would be running the three programs detailed in this document and getting reasonable and similar answers with statistical significance.

To advance to TRL 4, there must be testing within a laboratory environment showing that the analytical testing done before is correct.

To advance to TRL 5, there must be a few tests done for thermal qualification. Most namely are the Thermal Vacuum Chamber (TVAC) and the Vibrations Test (Vibe Test). The former is creating a like space environment to test the components capabilities within a simulated environment. The later is done by strapping the component to test to a vibrations table and seeing if the component can handle the rocket launch vibrations. Without these two tests a component cannot be raised to TRL 5.



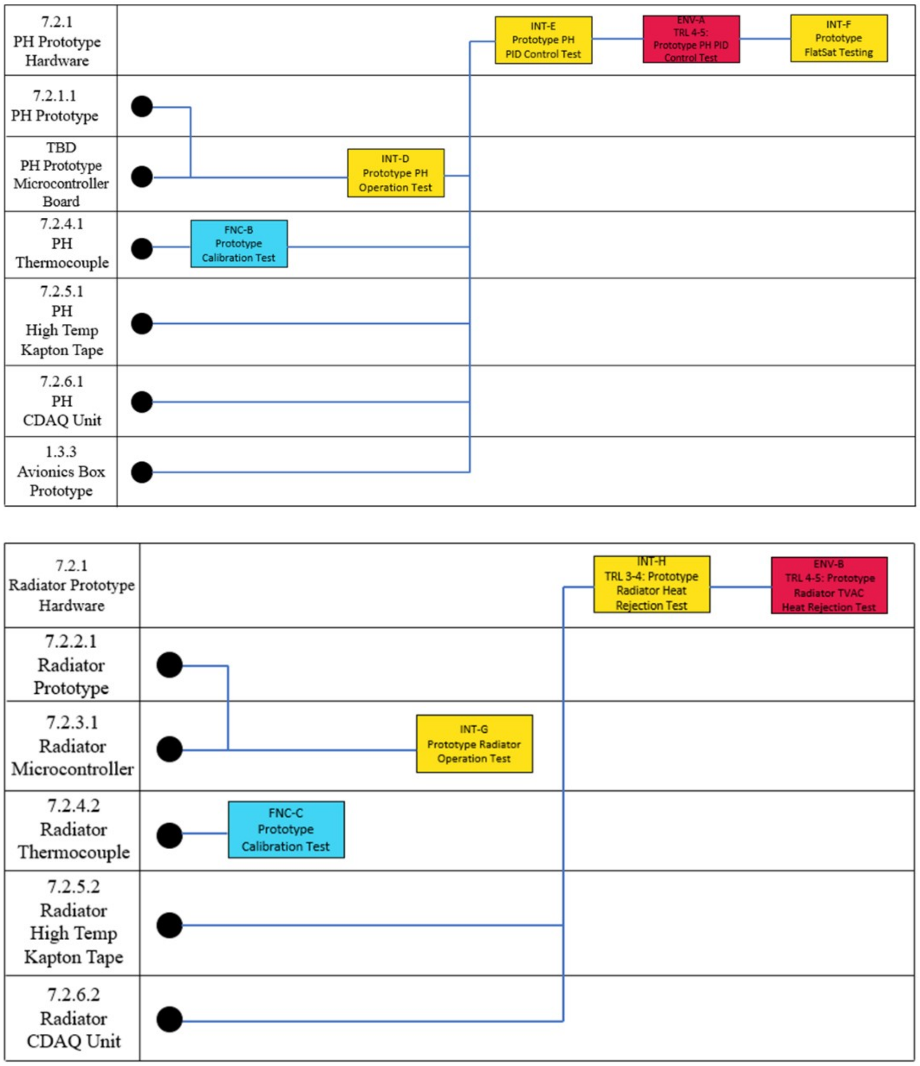


Figure 35: Integration Flow Diagram

5.3.1 Integration Points Descriptions

Describe in detail the integration points as they exist in the integration flow. Provide context here as needed on this outside of the information in the table.

Table 10: Descriptions of Integration Points from the Integration Flow Diagram

Integration Point Name	Description

5.3.2 Test Event Descriptions

Describe in detail the test events as they exist in the integration flow. Provide context here as needed on this outside of the information in the table.

Table 11: Descriptions of Test Events from the Integration Flow Diagram

Test Event Name	Type	Description

References

[1] E. Author, "IEEE Paper Example," *IEEE,* vol. 1, no. 1, p. 1, 2020.

[2] "NASA Systems Engineering Handbook," p. 297, 2016.

[3] W. J. Larson, D. Kirkpatrick, J. J. Sellers, L. D. Thomas and D. Verma, Applied Space Systems Engineering, McGraw Hill, 2018.

Appendix A Acronyms, Terminology, & Nomenclature

A.1 Acronyms

Acronym	Definition
ABEX	Alabama Burst Energetics Explorer
APRA	Astrophysics Research & Analysis
ASGC	Alabama Space Grant Consortium
CAD	Computer-Aided Design
C&DH	Command & Data Handling
CDR	Critical Design Review
CE	Chief Engineer
CLB	Configurable Logic Block
COTS	Commercial Off The Shelf
CS	Chief Scientist
CSLI	CubeSat Launch Initiative
DAC	Design Analysis Cycle
EAR	Export Administration Regulations
EPMs	Educational Performance Measures
EPS	Electrical Power System
FPGA	Field Programmable Gate Array
FSW	Flight Software
GN&C	Guidance, Navigation, & Control
GPS	Global Positioning System
GRB	Gamma-ray Burst
GRD	Gamma-ray Detector
HV	High Voltage
ICP	Instrument Calibration Plan
IMS	Integrated Master Schedule
IMU	Inertial Measurement Unit
ISM	Integrated Systems Model
ITAR	International Traffic in Arms Regulations
IV&T	Integration, Verification, & Test
KDP	Key Decision Point
KPPs	Key Performance Parameters
LSE	Lead System Engineer
MBSE	Models Based System Engineering
MCR	Mission Concept Review
NDA	Non-Disclosure Agreement
NIST	National Institute of Standards and Technology
PC	Program Coordinator
PDR	Preliminary Design Review

Acronym	Definition
PIU	Payload Interface Unit
PM	Project Manager
POP	Period of Performance
QP	Qualification Plan
QR	Qualification Report
QPSK	Quadrature Phase Shift Keying
SE	Systems Engineering
SEMP	System Engineering Management Plan
SIS	Software Interface Specification
SME	Subject Matter Expert
SMP	Software Management Plan
SQP	Structural Qualification Plan
SQR	Structural Qualification Report
SRD	System Requirements Document
SRR	System Requirements Review
STDP	Subsystem Technology Development Plan
STP	Subsystem Testing Plan
TCP	Technology Control Plan
TID	Total Ionizing Dose
TPM	Technical Performance Measure
TQP	Thermal Qualification Plan
TQR	Thermal Qualification Report
TRL	Technology Readiness Level
TT&C	Telemetry, Tracking, & Command
V&V	Verification and Validation
WBS	Work Breakdown Structure
XRD	X-ray Detector

A.2 Terminology

Term	Description
Acceptance	A type of verification procedure specifically for testing and analysis. Acceptance test/analysis criteria show that the manufacturing/workmanship of the unit conforms to the design that was previously verified/qualified. Acceptance activities are performed on each of the flight units as they are manufactured and readied for flight/use (NASA Systems Engineering Handbook, 2016)
Analysis	Verification by analysis is a predicted compliance to requirements. The use of mathematical modeling and analytical techniques to predict the suitability of a design to stakeholder expectations based on calculated data or data derived from lower system structure end product verifications. Analysis is generally used when a prototype; engineering model; or fabricated, assembled, and integrated product is not available. Analysis includes the use of modeling and simulation as analytical tools (NASA Systems Engineering Handbook, 2016).

Term	Description
Assembly	The mechanical mating of components to form a system.
Certification	The audit process by which the body of evidence that results from the verification activities and other activities are provided to the appropriate certifying authority to indicate the design is certified for flight/use. The Certification activity is performed once regardless of how many flight units may be generated (NASA Systems Engineering Handbook, 2016).
Configuration Item	The combination of two components, subsystems, or systems of lesser complexity resulting in a combined assembly, subsystem, or system with greater complexity. Configuration Items exist at Integration Points; a sequence of Configuration Items along several Integration Points comprises an Integration Chain.
Demonstration	Verification by demonstration is an observed compliance to requirements accomplished by showing that the use of an end product achieves the individual specified requirement. It is generally a basic confirmation of performance capability, differentiated from testing by the lack of detailed data gathering. Demonstrations can involve the use of physical models or mock-ups (NASA Systems Engineering Handbook, 2016).
Inspection	Verification by inspection is a documented compliance to requirements. The visual examination of a realized end product. Inspection is generally used to verify physical design features or specific manufacturer identification (NASA Systems Engineering Handbook, 2016).
Integration	The process of combining less complex functions, understanding those functions, and controlling those functions to achieve a system satisfying its requirements.
Integration Chain	A series of Integration Points. Integration Chains can be represented as tree or fishbone diagrams where many components, subsystems, or systems of lesser complexity are combined as Configuration Items at Integration Points to create a system of higher complexity. Integration Chains are generally defined to realize a Technical Performance Measure.
Integration Point	The location on a schedule where two or more components, subsystems, or systems of lesser complexity are combined as a Configuration Item with greater complexity. A series of Integration Points comprises an Integration Chain.
Interface	An interface represents a constraint based on the logical and physical boundary conditions between two or more entities within a level of abstraction, between System of Interest elements, between other mission systems, between enabling systems, or between the System of Interest and its Operational Environment. Interfaces can be for physical connection, energy transfer (power or heat), matter, or data (Wasson, 2016).
Key Performance Parameter	Those capabilities or characteristics (typically engineering-based or related to health and safety or operational performance) considered most essential for successful mission accomplishment. They characterize the major drivers of operational performance, supportability, and interoperability (NASA Systems Engineering Handbook, 2016).
Mode	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 non-tangible, non-physical concept.
Model	A mathematical representation of reality (NASA Systems Engineering Handbook, 2016).
Operational Environment	The surrounding systems, materials, or occurrences defining a system's ability to externally interact. The Operational Environment is comprised of a Human

Term	Description
	Systems Environment, a Natural Environment, and an Induced Environment (Wasson, 2016).
Primary TPM	A regular Technical Performance Measure, either a Key Performance Parameter, Technical Performance Parameter, or Technical Environmental Parameter; this distinction exists only as an organizational hierarchy.
Qualification	A subset of the verification program that is performed at the extremes of the environmental envelope and will ensure the design will operate properly with the expected margins. Qualification is performed once regardless of how many flight units may be generated as long as the design doesn't change.
Secondary TPM	A subdivision of a Primary TPM for the purpose of representing portions of Primary TPM concepts within a Domain Knowledge Map. Secondary TPMs are not tracked or reported and do not required a target threshold. Secondary TPMs exist only as an organizational hierarchy for conceptual organization. Secondary TPMs can break down further into more Secondary TPMs.
Simulation	The manipulation of a model (NASA Systems Engineering Handbook, 2016).
State	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.
Technical Performance Measure	A set of performance measures that are monitored by comparing the current actual achievement of the parameters with that anticipated at the current time and on future dates (NASA Systems Engineering Handbook, 2016)
Technical Performance Parameter	Those capabilities or characteristics (typically engineering-based or related to health and safety or operational performance) considered relevant to operational performance, supportability, and interoperability at any level.
Technical Environmental Parameter	Those capabilities or characteristics relevant to the definition of system interactions with the Operational Environment.
Test	Verification by test is a measured compliance to requirements. : The use of an end product to obtain detailed data needed to verify performance or provide sufficient information to verify performance through further analysis. Testing can be conducted on final end products, breadboards, brassboards, or prototypes. Testing produces data at discrete points for each specified requirement under controlled conditions and is the most resource-intensive verification technique. As the saying goes, "Test as you fly, and fly as you test" (NASA Systems Engineering Handbook, 2016).
Validation	Validation of a product shows that the product accomplishes the intended purpose in the intended environment—that it meets the expectations of the customer and other stakeholders as shown through performance of a test, analysis, inspection, or demonstration (NASA Systems Engineering Handbook, 2016).
Verification	Verification is a formal process, using the method of test, analysis, inspection or demonstration, to confirm that a system and its associated hardware and software components satisfy all specified requirements. The Verification program is performed once regardless of how many flight units may be generated as long as the design doesn't change (NASA Systems Engineering Handbook, 2016).

A.3 Nomenclature

Symbol	Description	Unit

Appendix B Defining TPMs

B.1 Purpose

TPMs are methods of tracking the technical performance of a system over time. They help decision making in the design through capturing the impact to the project into helpful parameters. TPMs can be defined for components, subsystems, and systems. System-level TPMs are considered Key Performance Parameters (KPP). TPMs can also be defined for capabilities, which are functions with a specified level of performance. Project will specify a certain level of performance through capability requirements, and that level of performance can be described using a TPM.

B.2 Structure

ABEX defines three types of TPMs: Technical Performance Parameters (TPP), Technical Environmental Parameters (TEP), and KPPs.

Table 12: TPM Type Definitions with an Example

TPM Type	Description	Example
Technical Performance Parameter	Those capabilities or characteristics (typically engineering-based or related to health and safety or operational performance) considered relevant to operational performance, supportability, and interoperability at any level. TPPs are elevated to KPPs if they are relevant to the entire system	Subsystem Mass
Technical Environmental Parameter	Those capabilities or characteristics relevant to the definition of system interactions with the Operational Environment. TEPs are never elevated to KPPs	Surface Heat Flux
Key Performance Parameter	Those capabilities or characteristics (typically engineering-based or related to health and safety or operational performance) considered most essential for successful mission accomplishment. They characterize the major drivers of operational performance, supportability, and interoperability [2]	Spacecraft Mass Margin

TPPs, TEPs, and KPPs are all TPMs. They're used to calculate relevant aspects of a design and communicate the value of that design to project stakeholders. TPMs must be tracked and reported each DAC. A basic visualization of this is shown in Figure 36. Well-defined TPMs have defining features:

- Should be important and relevant to the subsystem design
- Should be relatively easy to measure for reporting
- The performance or knowledge of performance should be expected to improve with time
- A target, threshold, or expectation of uncertainty should be known and if the measure crosses its threshold, corrective action should be known
- The measured parameter should be controllable by the design decisions
- Should be tracked and documented
- Should be tailored for the project

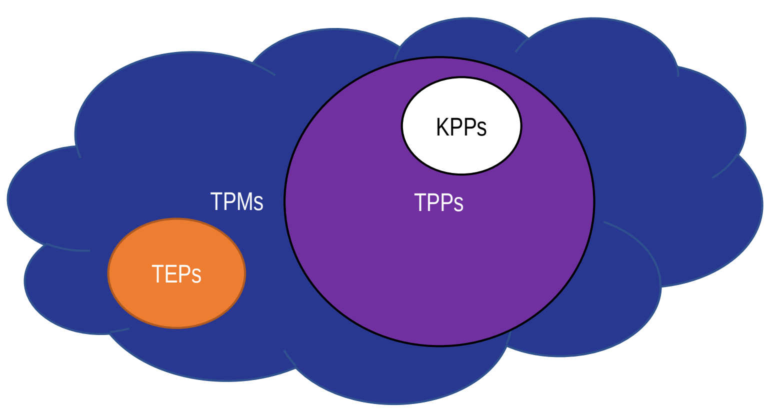


Figure 36: ABEX TPM strategy visualization

Appendix C Domain Knowledge Maps

C.1 Purpose

Domain Knowledge Maps (DKMs) are designed to show and teach designers on calculating a TPM by describing the inputs to it, the equations or models used in determining it, and where information is sourced from for the inputs. DKMs are represented as a visual diagram, with the point of DKMs are to organize and represent information in a clear manner using a defined technical domain category to organize the entities and specific relationships between them.

C.2 Structure

The DKMs are expressed as blocks organized around the technical domain categories with relationships between them as semantic triplets, an example of which is shown below:

Subject	**Predict**	**Object**
Scalar Parameter 1	is an input to	Equation 1

These semantic triplets, read from left to right, are used to construct a natural language expression of the relationship between concepts behind a TPM. However, if every possible relationship is put on a DKM for every parameter, the DKM may become overly complicated and less useful for concept representation. Only the main relationships should be highlighted, and others can be implied, such as the “is a function of” relationship. An example of a simple diagram is shown below:

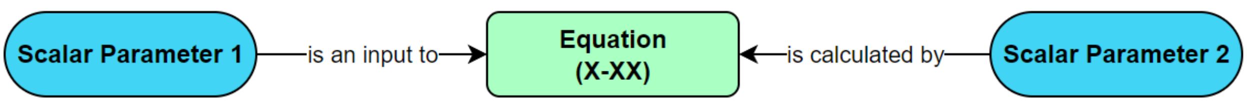


Figure 37: Simple DKM with Implied Relations

In this Simple DKM with implied relations, the “is a function of” relationship would exist between scalar parameter 2 and equation 1 but is left implied by the relationship of “is calculated by”. This is ignored to be more direct and easier to read. The color coding indicates what the entity is with respect to the technical domain categories.

C.3 Guidance

DKMs can be made in whatever format is most understandable to the team making the DKM while adhering to the established categories and relationships in the DIP. DKMs are supposed to be straightforward and easy to read. If the DKM becomes complicated, then it should be broken up into smaller DKMs (around convenient secondary parameters) for comprehension. Bulleted are some tips for easy DKM creation.

- All TPMs are ovals, everything else is a rectangle.
- Keep all shapes a similar size.
- If a line is drawn, give it a distinct relationship using the semantic triplet. If keeping relationships near each other looks bad, move the line before trying to convey that multiple lines have a single relationship prescribed to it.
- Make the DKMs read left-to-right, when possible, to follow the natural direction of reading.
- Use the triplet structure to capture the methodology and flow of information when calculating a TPM.
- The color of the shape should be an expression of how that thing is represented. using the established categories of the domain (e.g., scalar parameter, matrix parameter, etc.).

- Equations should reference the numbering in the DIP.

With each DKM descriptions should be provided explaining any connection to other DKMs and on the methodology for calculating it with equations discussed. If the DKM is large then it should be split up along convenient points, such as intermediate parameters.

C.4 DKM Creation & Example

The easiest way to create a DKM is in a free, web-based flowchart platform called draw.io. Typing draw.io into an internet browser will initiate the creation of a new diagram. Flowchart is closest to the types of diagram structures utilized in DKM creation. Remember to not show the background grid and text and image have enough resolution to be readable when imported into the DIP[1]. The only things needed to create robust DKMs are rectangle blocks, oval blocks, text in the blocks, lines with text, and the colors. An example of a DKM is shown below for reference.

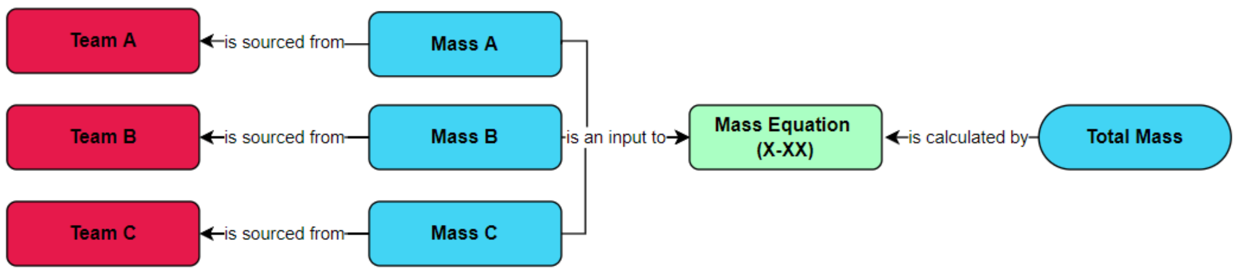


Figure 38: Mass Calculation Example DKM

This Example DKM is for a simple mass calculation that takes in inputs from different teams. The number (X-XX) corresponds to the TPM section and equation in that section as would be shown in the DIP. Total mass is a scalar parameter, but also a TPM as shown by its oval. The flow calculation is done from left to right.

[1] Tip on exporting: Under the diagram tab uncheck grid and set background to white. Then you can use the Windows Snipping Tool (or to select sections of the diagram).

Appendix D Integration & Testing Process

Integration and Testing (I&T) is a common grouping of important SE concepts. This grouping is intended to describe the gradual combination of less complex systems into a system of increasing complexity and the confirmation that the combination of less complex systems functions as anticipated. Testing is an implicit part of integration and an explicit method of verification. For a given subsystem design, engineers must verify component functionality, integrate components into subsystems, verify subsystem functionality, integrate subsystems into systems, and verify system functionality. The system is then put through qualification tests to raise its Technology Readiness Level. Verification by test at each individual step is the least risky option for component, subsystem, or system functionality verification, but each test costs both time and money. Programs have been cancelled because they tested too much and too often, but failure to plan sufficient integration testing can also led to mission failure. System integration planning must balance both cost and risk when developing an integration strategy. The following material provides insight into what I&T is and how to organize it effectively.

D.1 Integration

Integration is one of the most challenges aspects of engineering complex systems. A complicated system is one with many inputs and outputs; a complex system is one with many interfaces for mass, energy, data, or physical interactions. Issues or shortcomings with design specifications are exposed, and the entire process is increasingly difficult as the number of parts or interfaces grows. Integration is not the same as assembly. Assembly emphasizes the mechanical mating of components to form a system while integration encompasses the entire process of combining less complex function to achieve a system satisfying its requirements. Integration is accomplished by defining Integration Points (IP), Configuration Items (CI), and Integration Chains (IC).

D.1.1 Configuration Items

A CI is a combination of two or more components or subsystems. It's what happens when two things with individual functionality are combined. Two PCBs put into a stack might be a CI, or a heater applied to a battery box might be a CI. CIs are steppingstones on the way to the final system. If a system is put together all at once and doesn't work, it might be difficult to determine what went wrong or how to fix the problem. Functionality of a CI must be determined at each IP to move from one CI to the next; functionality must therefore be verified by inspection, demonstration, analysis, or test, usually test.

D.1.2 Integration Points

An IP happens when a CI is created; the IP is the event while the CI is the combination of components or subsystems.

D.1.3 Integration Chains

An IC is a planned set of IPs that result in the successful operation of some system functionality. A set of functions and subsystems that result in a critical parameter or TPM is called a chain. If the TPM includes computational operations or code execution, this chain may include real-time, information intensive interfaces or the inclusion of sensors, actuators, or mechanisms. While it may seem easier to start with the least complex parts and work to combine components to create a more complex system, it is often more efficient to start with the finalized system or subsystem and work backwards to define an IC.

D.1.4 Integration Considerations

While ICs are determined by working backwards from a complete system, integration is enacted first by the provisional testing of individual components at the lowest levels. Faults can be identified, isolated, and recovered from within a small and controllable scope, and problems caught at lower levels are easier and cheaper to fix. While not every miniscule combination of components merits testing, for this reason testing at lower levels should not be avoided. When deciding if a combination of components represents a CI that must be tested for functionality, several questions can be asked.

- How critical is the component or combination of the components to the operation of the subsystem?
- What aspects of the CI are sufficiently close to final operation that feedback from this CI makes sense?
- How much time, money, or effort must be invested in this IP?

D.1.5 Software Integration

Hardware integration is moderately linear; there are direct paths joining other paths that result in a final product. Software integration, conversely, is modular and incremental. When integrating software into hardware, there are four levels of decreasing abstraction: Model-In-The-Loop (MIL), Software-In-The-Loop (SIL), Processor-In-The-Loop (PIL), and Hardware-In-The-Loop (HIL). As the software fidelity increases, the more realistic the test cases become. These XIL tests can be planned into an IC. In Table 13, simulated refers to an entirely computational representation without hardware, development signifies an intermediate or prototype design or deployment, and production means the flight model.

Table 13: XIL Nomenclature Descriptions and Connection to Major Integration Elements

Elements	Model-In-The-Loop	Software-In-The-Loop	Processor-In-The-Loop	Hardware-In-The-Loop
Code	Development	Development	Production	Production
Controller	Simulated	Development	Production	Production
Sensors	Simulated	Simulated	Simulated	Production
Environment	Simulated	Simulated	Simulated	Simulated

D.2 Testing

Testing means operating a system (or one or more of its parts) in a predefined way to verify its behavior.

D.2.1 Testing Types

When planning to test individual components, subsystems, systems, or CIs, each test includes both a test type and test category. Development tests are generally either functional tests or integration tests. Qualification tests are environmental tests, and very often environmental testing includes and builds upon functional testing. Acceptance tests are a type of environmental test that consider workmanship or final system checkout rather than testing the system to its limits of operation within a given operational environment.

D.2.1.1 Type 1: Development Tests

Development tests validate new design concepts, techniques, configurations, or the combination of any of those. Development testing occurs early during the development life cycle phase and typically confirm aspects of a design such as performance margins, manufacturability, reliability, failure modes, or a systems ability to be tested. Documentation for development tests may not be as rigid as that of qualification or acceptance tests because development tests occur earlier and more frequently in the design life cycle. Operational conditions should be varied during development tests. The purpose is to inform system design (Larson, Kirkpatrick, Sellers, Thomas, & Verma, 2018).

D.2.1.2 Type II: Qualification Tests

Qualification tests validate a design for the intended operational environment. While the test should occur for the extremes of the environment, the objective is to show evidence of applicable design margins, not to cause failure modes unrealistic to nominal operation. Qualification tests are rigorous and usually expensive, so they should be performed when both a hardware and software design are finalized. Qualification testing occurs directly before acceptance testing of build products.

D.2.1.3 Type III: Acceptance Tests

Acceptance testing assures conformance to specification requirements and provides quality control assurance against workmanship or material deficiencies (Larson, Kirkpatrick, Sellers, Thomas, & Verma, 2018). Acceptance tests verify requirements associated with build-to or product specifications, readiness for delivery to customers, and acceptance by the customers. They check for workmanship, not the design of the system. The purpose for acceptance tests is to show that the production version of a system was manufactured, assembled, and integrated correctly. These testing activities occur for each item produced whereas qualification tests occur only once for a given design. Acceptance tests occur in the late development life cycle phase.

D.2.2 Test Categories

Test categories exist within test types. Both the type and category should be specified for an IP.

D.2.2.1 Functional Tests

Functional tests answer the question, “Does it work?” They are used to verify the system’s functional requirements at all levels, be those component, subsystem, or the spacecraft level. Deployment of a solar array, antenna frequency tests, or attitude control system operation would be functional tests. It is important to note the difference between function and capability. A capability is a function at a level of performance; a function itself is unitless. Testing for a capability counts as testing for a function, but the distinction should be made in test planning whether the test is for a function or a function at a specified level of performance. (Larson, Kirkpatrick, Sellers, Thomas, & Verma, 2018).

D.2.2.2 Integration Tests

Integration tests answer the question, “Do the pieces work together as intended?” At a given IP, a CI may be tested using both integration and functional tests. Integration tests may include mechanical, electrical, or software checks, with software requiring the most planning. Electrical integration checks may include data or power interfaces, power consumption, data transfer, grounding, or soldering. Software integration checks may include any version of XIL testing. Mechanical interfaces, alignments, and mass properties should be checked during integration (Larson, Kirkpatrick, Sellers, Thomas, & Verma, 2018).

D.2.2.3 Environmental Tests

Environmental tests answer the question, “Does it work in the operational environment?” Note that the environmental test question includes and builds upon the functional test question. Environmental tests verify that components will withstand conditions the system will be exposed to and typically include thermal vacuum, mechanical vibration, mechanical shock, electromagnetic compatibility, or radiation testing. Some systems may require humidity, salt spray, or leakage testing (Larson, Kirkpatrick, Sellers, Thomas, & Verma, 2018).

D.3 Documenting Integration Flows

To describe the integration flow, a UML diagram should be used (or a similarly structured item) which captures the sequence of events. The diagram will be organized using horizontal swimlanes for each PBS item and vertical swimlanes for the testing types (development, qualification, acceptance). Blocks are used with connecting lines to show the tests and integration points organized around the PBS swimlanes. For integration tests not explicitly organized along the PBS a block can be placed between swimlanes to show its integration of them. An example of this is shown in Figure 39, with the color coding and naming scheme described in Table 14.

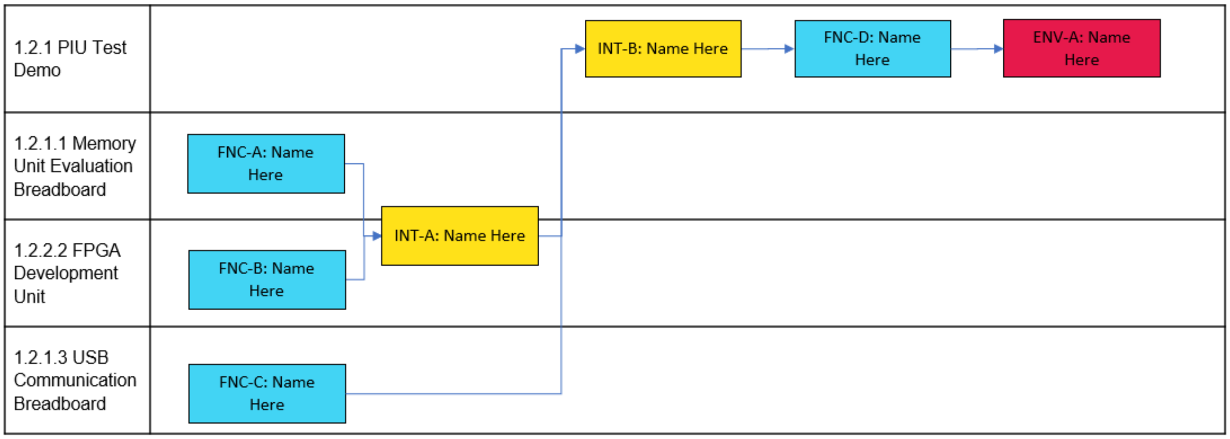


Figure 39: An Example of a Test Flow

In this example test flow, vertical swimlanes are not shown. Integration points generally map to configuration items and a configuration chain is shown as a roll up into the higher PBS items. Note that the INT-A test is not part of the PBS but a possible integration point to get towards PBS 1.2.1 and lives between the swimlanes it needs.

Table 14: Color Codes for Test Types

Test Type	Acronym / Shorthand	Color Code
Functional Test	FNC-ID#	*Hex#: 42D4F4*
Integration Test	INT-ID#	*Hex#: FFE119*
Environmental Test	ENV-ID#	*Hex#: E6194B*

The IDs in the integration flow are used to link to the tables in the DIP which describe these points and tests in detail.

Appendix E Technology Readiness Levels

E.1 Purpose

Technology readiness is used to determine how ready a technology is to operate as a part of a system, based on its demonstrated functional ability within the target environment and how ready it is to be integrated within the flight system. Levels are defined within this to act as milestones during technology development, integration, and testing.

E.2 Level Classifications

NASA defines 9 levels of technology readiness. These are described in Table 15 including the descriptions for what this means for the hardware and software elements of a technology with exit criteria expectation.

Table 15: TRL Definitions Adapted from NASA NPR 7123.1C

Lvl	Definition	Hardware Description	Software Description	Exit Criteria
1	Basic principles observed and reported.	Scientific knowledge generated underpinning hardware technology concepts/applications.	Scientific knowledge generated underpinning basic properties of software architecture and mathematical formulation.	Peer reviewed publication of research underlying the proposed concept/application.
2	Technology concept and/or application formulated.	Invention begins, practical application is identified but is speculative, no experimental proof or detailed analysis is available to support the conjecture.	Practical application is identified but is speculative, no experimental proof or detailed analysis is available to support the conjecture. Basic properties of algorithms, representations and concepts defined. Basic principles coded. Experiments performed with synthetic data.	Documented description of the application/concept that addresses feasibility and benefit.
3	Analytical and experimental critical function and/or characteristic proof of concept.	Analytical studies place the technology in an appropriate context and laboratory demonstrations, modeling and simulation validate analytical prediction.	Development of limited functionality to validate critical properties and predictions using non-integrated software components.	Documented analytical/experi-mental results validating predictions of key parameters.
4	Component and/or breadboard validation in laboratory environment.	A low fidelity system/component breadboard is built and operated to demonstrate basic functionality and critical test environments, and associated performance predictions are defined relative to the final operating environment.	Key, functionally critical, software components are integrated, and functionally validated, to establish interoperability and begin architecture development. Relevant Environments defined and performance in this environment predicted.	Documented test performance demonstrating agreement with analytical predictions. Documented definition of relevant environment.
5	Component and/or breadboard validation in	A medium fidelity system/component brassboard is built and operated to demonstrate	End-to-end software elements implemented and interfaced with existing systems/simulations conforming to target environment.	Documented test performance demonstrating agreement with

Lvl	Definition	Hardware Description	Software Description	Exit Criteria
	relevant environment.	overall performance in a simulated operational environment with realistic support elements that demonstrates overall performance in critical areas. Performance predictions are made for subsequent development phases.	End-to-end software system, tested in relevant environment, meeting predicted performance. Operational environment performance predicted. Prototype implementations developed.	analytical predictions. Documented definition of scaling requirements.
6	System/sub-system model or prototype demonstration in an operational environment.	A high-fidelity system/component prototype that adequately addresses all critical scaling issues is built and operated in a relevant environment to demonstrate operations under critical environmental conditions.	Prototype implementations of the software demonstrated on full-scale realistic problems. Partially integrate with existing hardware/software systems. Limited documentation available. Engineering feasibility fully demonstrated.	Documented test performance demonstrating agreement with analytical predictions.
7	System prototype demonstration in an operational environment.	A high-fidelity engineering unit that adequately addresses all critical scaling issues is built and operated in a relevant environment to demonstrate performance in the actual operational environment and platform (ground, airborne, or space).	Prototype software exists having all key functionality available for demonstration and test. Well integrated with operational hardware/software systems demonstrating operational feasibility. Most software bugs removed. Limited documentation available.	Documented test performance demonstrating agreement with analytical predictions.
8	Actual system completed and "flight qualified" through test and demonstration.	The final product in its final configuration is successfully demonstrated through test and analysis for its intended operational environment and platform (ground, airborne, or space).	All software has been thoroughly debugged and fully integrated with all operational hardware and software systems. All user documentation, training documentation, and maintenance documentation completed. All functionality successfully demonstrated in simulated operational scenarios. Verification and Validation (V&V) completed.	Documented test performance verifying analytical predictions.
9	Actual system flight proven through successful mission operations.	The final product is successfully operated in an actual mission.	All software has been thoroughly debugged and fully integrated with all operational hardware/software systems. All documentation has been completed. Sustaining software engineering support is in place. System has been successfully operated in the operational environment	Documented mission operational results.

E.3 Raising TRL

In general, you can conceptualize the process of raising a TRL (from a subsystem perspective) in the following manner:

1. Define the requirements for a subsystem. For ABEX, these are provided by Management.
2. Define an architecture (what) that characterizes the functions described in the requirements.
3. Determine the TPMs that a design can provide to meet the specification requirements.
4. Define a design (how) that specifically details what components or subsystems are providing the functionality described in the requirements. Characterize the entire design space to meet the problem space before selecting the design.
5. Break the subsystem design into constituent components, organized into a PBS. For ABEX we include both the development hardware elements and flight hardware.
6. Establish the current TRL level of the subsystem based on the concept design.
7. Determine the entry and exit criteria for each TRL level from its current point to TRL 6. Determine the test categories required to verify compliance of the subsystem design to its requirements as they relate to each TRL level. Define the development hardware we need and include that into the PBS.
8. Determine the configuration items that could exist and associated integration chains for each TRL advancement. This might include several TRL steps at once.
9. Once the subsystem has demonstrated TRL 6 from tests done independently to it, the flight unit is ready to be built and tested to the operational environment with the whole spacecraft together to achieve TRL 8.
10. After flight the subsystem is at TRL 9.

Appendix F Product Breakdown Structure

F.1 Purpose

The Product Breakdown Structure (PBS) is designed to organize the system elements into a hierarchy to organize work, planning, documentation, and requirements around the physical things that a project needs to make. The PBS for ABEX will capture both the flight elements and the development products needed to mature the systems technology. The PBS will also establish a common terminology for the system for the team and for the project's system engineering effort. Flight units are the things which we need for the final mission. Development products are the things that we need to raise the subsystem TRL, but that will not end up in the final mission.

F.2 Guidance

When creating the PBS, the following points should be kept in-mind:

- Products are organized around a tree structure that represents how products fit within (or integrate) into each other in the final implementation.
- The leveling approach should go as far as needed to capture the major elements of the project but does not need to recreate a full parts list (such as each capacitor and resistor).
- PBS is used to organize the project, so the hierarchy should also group elements together based on how we interact with the system, such as showing categories with sub elements for connectors and cables for interfacing.
- Development is to be included in the ABEX PBS and should be organized around major technology advancement hardware units/setups.
- Each PBS should use an ID system, this numbering can be relative to the system (with the ability to map/integrate into the projects PBS).

An example of this is shown below in Table 16 for a fictional system.

Table 16: Example Product Breakdown Structure

PBS ID	Product Name	Product Description
1	System Name	System example name
1.1	Flight Products	Flight system elements
1.1.1	Power Conversion Board	Power conversion system
1.1.2	Data Acquisition Unit	Data logging and conversion unit
1.2	Development Products	All the technology development hardware
1.2.1	Tech Demo	Demonstration hardware setup for TRL 4/5.
1.2.1.1	Power System Test Unit	This is a flight-like power system unit with test interfaces
1.2.1.2	Data Acquisition Breadboard	Scale data acquisition breadboard with main processing FPGA.
1.2.2	Concept Demo	FPGA development demo with data interfaces for TRL 3.

Appendix G [Subsystem Specific Information]