NEA Scout Thermal Control

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The Near-Earth Asteroid Scout (NEA Scout) is a 6U CubeSat that will fly to a near earth asteroid using a solar sail. The mission is a joint project between NASA's Marshall Space Flight Center and the Jet Propulsion Laboratory. The CubeSat will be deployed as a secondary payload during the Space Launch System (SLS) Exploration Mission 1 (EM-1). The CubeSat will use an 85 m^2 (915 ft^2) aluminized polyimide solar sail for deep space propulsion. A multispectral camera will be used to characterize a small asteroid (<300 feet in diameter). The primary thermal architecture is a passive design with heaters to keep temperatures above the minimum allowable. Thermal vacuum testing was done on subsystems where possible. However for some long lead subsystems thermal vacuum testing will not be done until the final assembly.

Nomenclature

6U	=	6 Units
α	=	solar absorptivity
AFT	=	Allowable Flight Temperature
Ag	=	Silver
AMT	=	Active Mass Translator
AU	=	Astronomical Unit
BOL	=	Beginning of Life
BCT	=	Blue Canyon Tech
CDH	=	Command Data Handling
Comm	=	Communication
Con-Ops	=	Concepts of Operations
ε	=	Infrared Emissivity
EM-1	=	Exploration Mission 1
EPS	=	Electrical Power System
EOL	=	End of Life
ETF	=	Environmental Test Facility
FEP	=	Fluorinated Ethylene Propylene (Teflon)
FEM	=	Finite Element Model
FASTSAT	=	Fast, Affordable, Science and Technology Satellite
IDD	=	Interface Definition Document
ITO	=	Indium Tin Oxide
IMU	=	Inertial measurement Unit
JPL	=	Jet Propulsion Laboratory
LGA	=	Low Gain Antenna
LNA	=	Low Noise Amplifier
MPCV	=	Multi-Purpose Crew Vehicle
MSFC	=	Marshall Space Flight Center
MSA	=	MPCV Stage Adaptor
MGA	=	Medium Gain Antenna
MMA	=	Mountain Man Aerospace
NEA	=	Near Earth Asteroid

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PCB = Printed Circuit Board
PF = Protoflight
RCS = Reaction Control System
RF = Radio Frequency
RTV = Room-Temperature-Vulcanizing
RWA = Reaction Wheel Assembly
SLS = Space Launch System
SSPA = Solid State Power Amplifier
SSDM = Solar Sail Deployer Mechanism
UV = Ultra-violet
VBA = Vehicle Assembly Building

I. Introduction

THE Near Earth Asteroid (NEA) Scout[1] is a deep space CubeSat that will navigate to a near Earth asteroid. NEA Scout will be a secondary payload on the Space Launch System (SLS) Exploration Mission 1 (EM-1). Currently it is planned that it will be deployed from SLS's Multi-Purpose Crew Vehicle (MPCV) Stage Adaptor (MSA) soon after bus stop 1, which is the first location that secondary payloads are deployed, while MSA is travelling through the Van Allen Belts. NEA Scout will deploy an 85 m^2 (915 ft^2) aluminized polyimide solar sail, see Figure 1. This sail will be used as a propulsion element to allow the CubeSat to travel to the asteroid. Once at the asteroid a multispectral camera will be used to conduct scientific observations[2].

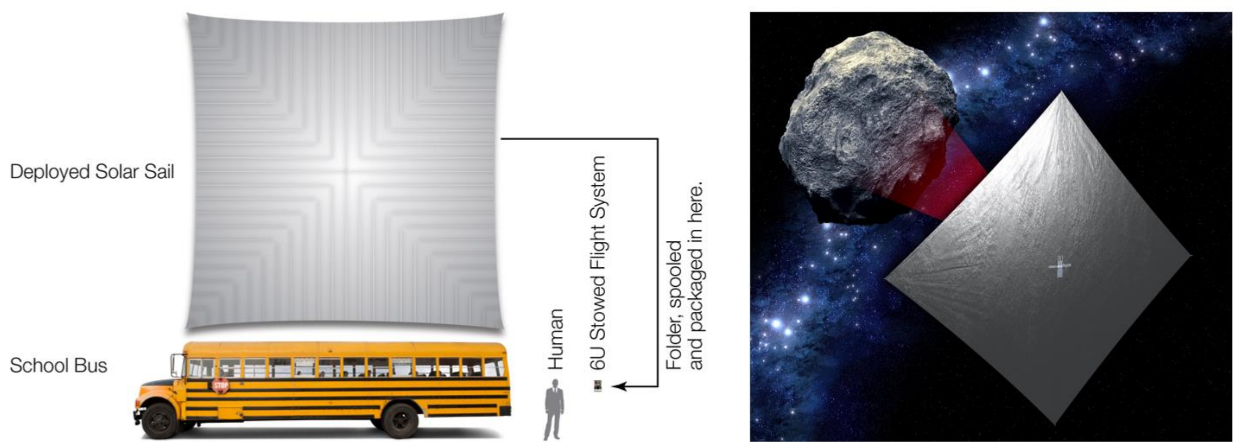

Figure 1. NEA Scout Approximate Scale

The NEA Scout flight system meets the 6U CubeSat form factor and was built as a collaborative effort of many parties as shown in Figure 2. The Jet Propulsion Laboratory (JPL) designed and will integrate the avionics box. Marshall Space Flight Center (MSFC) designed and built the active mass translator[3] and the sail deployer mechanism[4]. VACCO Industries is the vendor for the cold gas thruster for de-tumble and thrust correction maneuvers prior to solar sail deploy. Blue Canyon Technologies (BCT) is the vendor that provided attitude control systems and the 2x3U solar panel. Mountain Man Aerospace (MMA) provided the 1x3U tri-fold HaWK solar array. MSFC will perform the final integration and test of the spacecraft. On orbit, once the sail is deployed the deployer, active mass translator, and avionics box will be shaded from the Sun.

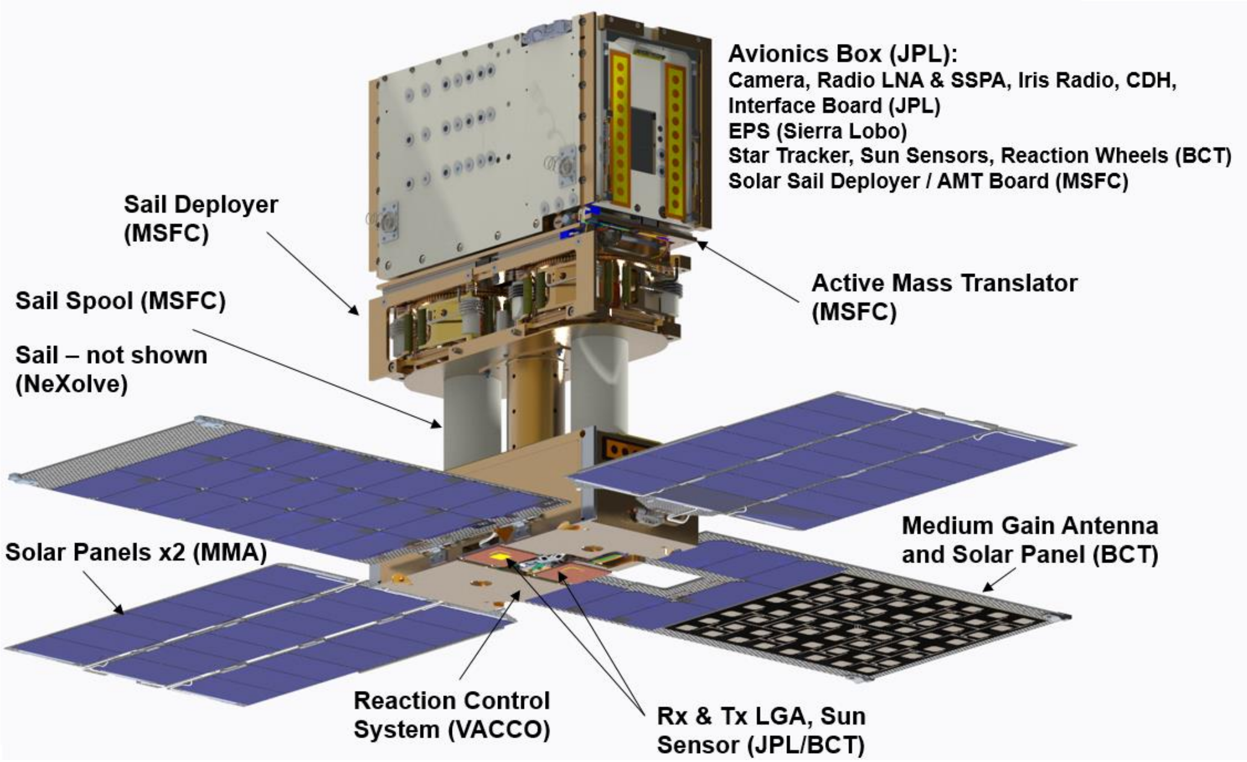

Figure 2. Flight Systems Overview

II. Mission Timeline and Environment

The mission timeline consists of three major phases. The first phase is pre-SLS launch. This includes integration in the payload processing facility, MSA integration at the Vehicle Assembly Building (VAB), roll out with no gas purge, and on-pad with gas purge (no tank and tanked). During this time the spacecraft will be unpowered and its temperature is assumed to follow the temperatures range as defined by the Interface Definition Document (IDD) with SLS, see Table 1 for some of the temperatures.

Table 1: SLS Secondary Payload Dispenser Temperature Range (not finalized, may be updated)

	Minimum Temperature	Maximum Temperature
VAB	4°C	32°C
Rollout, No Purge	-3°C	54°C
On-Pad, Purge	-2°C	38°C
On-Pad, Tanked, Purge	-8°C	35°C
Ascent	-4°C	31°C

The second phase covers the launch of the SLS until deployment from the MSA. The spacecraft will be stowed on the MSA that is located underneath the Orion Service Module. Once the Orion vehicle and service module separates from the MSA, the CubeSats and dispensers will ride along with the upper stage on a disposal trajectory towards the moon. Soon thereafter, the secondary payloads will begin to be dispensed. During this time the spacecraft will be unpowered and its temperature is assumed to follow the temperatures range as defined by the IDD with SLS.

The third phase is post deployment from the MSA when the CubeSat is exposed to space. During this phase a number of concepts of operations (con-ops) will occur. The thermally relevant ones are shown in Table 2; currently for in the thermal model these phases are analyzed to steady state, and will be updated to a transient run as needed. The highest heat dissipative con-ops usually occur when NEA Scout is in communication with the Earth using the IRIS radio. The assumed power in Table 2 are estimated expected values, the total power dissipated will not be measured until final thermal vacuum testing of the integrated vehicle. Currently the only fault case being analyzed is the battery recharge / safe case. The assumed space environment during this phase are shown in Table 3. The environmental fluxes will be updated as mission design refines their trajectory analysis based on launch date and potential target location.

Table 2: NEA Scout Thermally Relevant Con-Ops

	Notes	**Approximate Heat Loads**	**Sail Position**	**Solar Array's Angle to Sun**
Post MSA Deployment	Deploy solar arrays, use RCS to detumble and orient spacecraft to sun facing. Allow batteries to recharge.	30W in RCS, 20W in avionics	Stowed	Unknown at first, then sun facing (0°)
Thrust Control Maneuver	Use RCS to preform thrust control maneuver to get CubeSat on proper trajectory	30W in RCS, 40W in avionics	Stowed	Unknown
Sail Deploy	Deploy sail, while communicating with Earth	50W in avionics, 5W in sail mechanism	Deploying	0° to the Sun
Cruise	Spacecraft will spend most of life in this state	20W in avionics	Deployed	50° to the Sun (+/-5°)
Comm	Communicating with Earth	45W in avionics	Deployed	70° to the Sun
Battery Recharge / Safe	Battery recharge if depleted and safe mode	20W in avionics	Deployed	0° to the Sun
Science	Camera operations while communicating with Earth	45W in avionics	Deployed	Assuming 50° to the Sun

Table 3: NEA Scout Space Environments

Solar Distance	**Solar Flux**
0.978 to 1.017 AU	Cold Case = 1318 W/m2 Nominal Case = 1367.5 W/m2 Hot Case = 1433 W/m2

III. Overview of Thermal Model

The NEA Scout thermal model was created using Thermal Desktop, TD Direct, and FEMAP and solved using SINDA/FLUINT. The model initially was created solely using TD Direct, but has since been updated using FEMAP Finite Element Mesh (FEM) meshes and Thermal Desktop native entities. Each phase of the mission is broken down into a hot and cold cases. The spacecraft has four main subsystems that make up the body: avionics, AMT, SSDM, and the RCS seen in Figure 2. The original and updated thermal models can be found in Figure 3 and Figure 4.

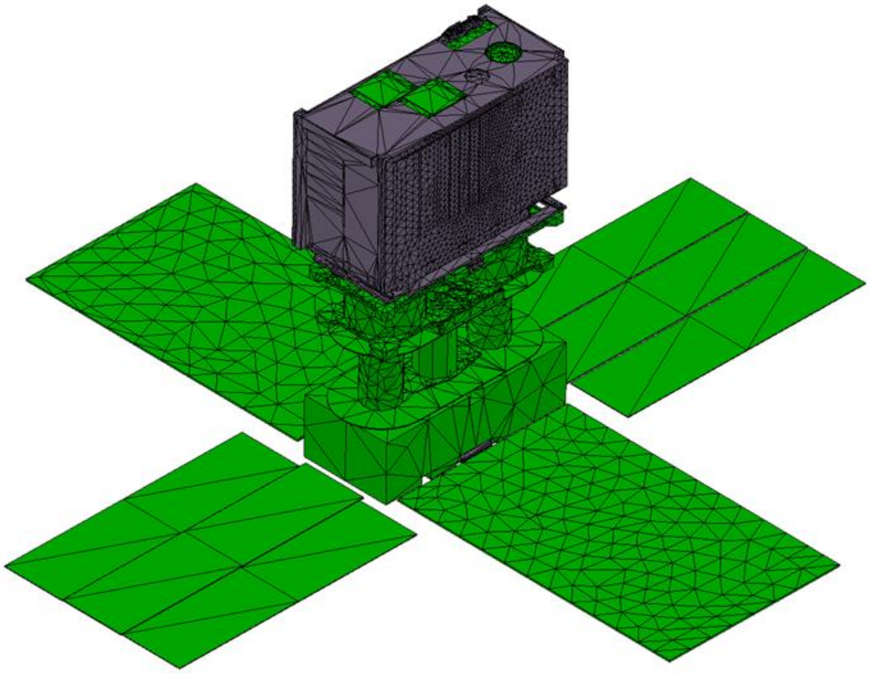

Figure 3: Original TD Direct Model

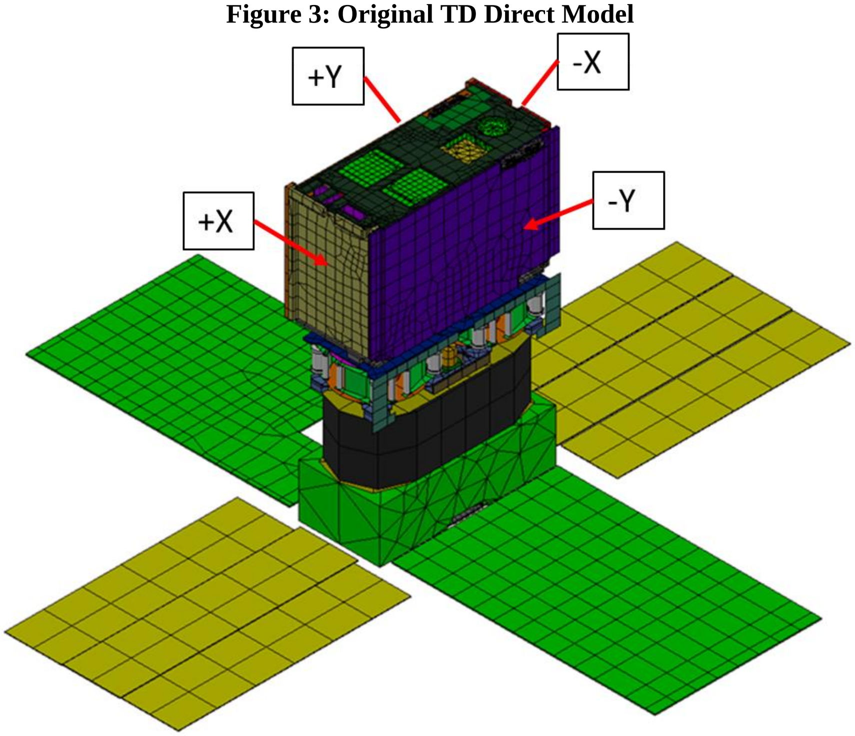

Figure 4: Updated Thermal Desktop Model

Figure 3 is the original model built using TD Direct, and Figure 4 was built using a combination of FEMAP and Thermal Desktop. It was determined during a model review of the first model that the majority of the model would need to be re-meshed due to mesh density issues and some components needed a geometry update. An example of the mesh errors can be seen in Figure 5 and Figure 6.

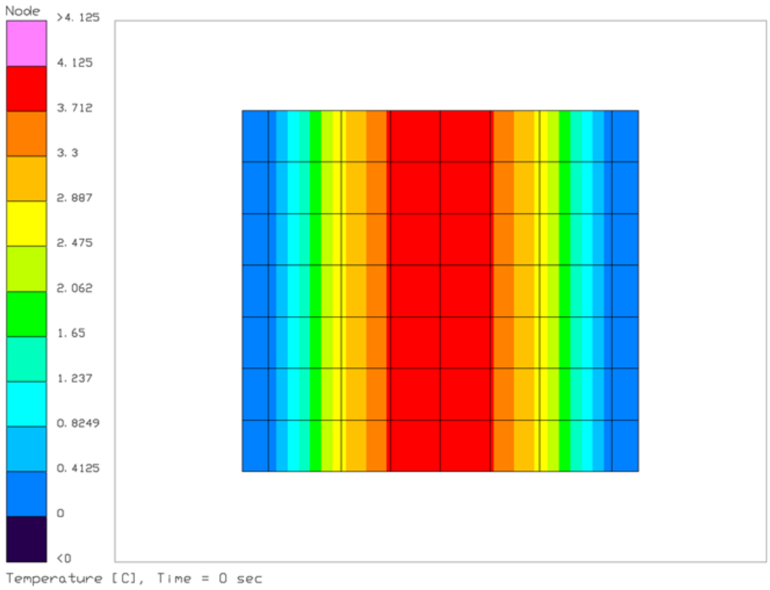

Figure 5: PCB with Updated Mesh

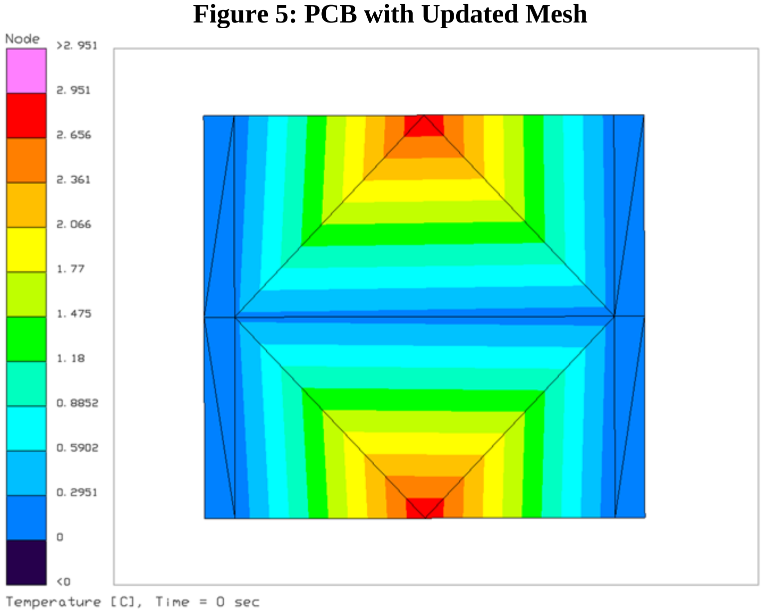

Figure 6: PCB with Original Mesh

Figure 5 and Figure 6 are showing the temperature contours of the PCB within the avionics of NEA Scout. The PCB was chosen due to being easily manipulated for a comparison of mesh densities. The PCB was pulled from the model and the same boundary conditions were applied to both. These conditions were similar to what is imposed from the overall model. The new mesh results follow expected contours from the boundary conditions applied and are about 1.2°C higher than the old mesh. This was similar with other components that were re-meshed throughout the model. While increases in temperature are not desired, it gives a greater confidence in the numeric accuracy of the model.

A unique thermal challenge for NEAS is due to the spacecraft being separated into two different thermal environments due to the solar sail. The avionics, AMT, and most of the SSDM are on the shaded, or dark side, of the sail. While the SSDM spool, the RCS, and solar arrays are on the sun facing side. These two sides can be seen in Figure 7 and Figure 8. This poses passive thermal control challenges through managing optical properties on the different surfaces. The optical properties used in the Thermal Desktop thermal model can be found in Table 4 below.

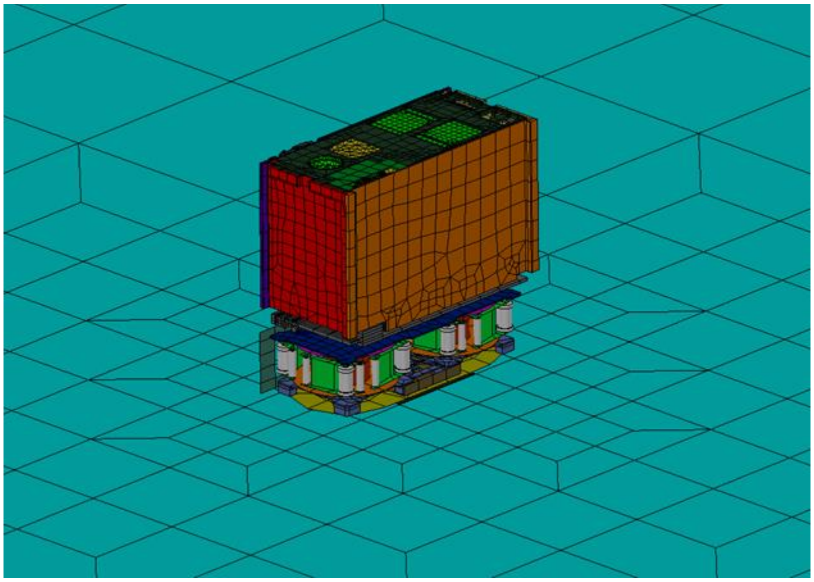

Figure 7: Shaded Side of Sail

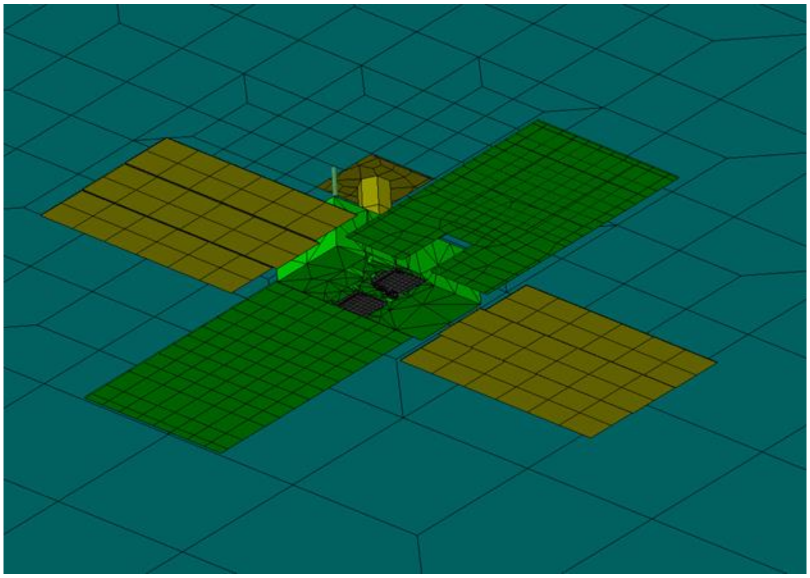

Figure 8: Sun Facing Side of Sail

Table 4: NEAS Optical Properties

Name	Solar Absorptivity	IR Emissivity	a/e
Chemfilm (Alodine < 2 min immersion)	0.356	0.048	7.417
Elgiloy (sail booms)	0.543	0.107	5.075
Carbon Fiber (solar array substrate) assumed to be similar to graphite optical properties	0.930	0.850	1.094
Kapton	0.120	0.880	0.136
MGA Effective Properties	0.452	0.710	0.637
Lens	0.090	0.030	3.000
Avionics PX-Paint (combination of Z93, Kapton over aluminum, and alodine)	0.304	0.681	0.446
RCS Alodine (Alodine 1201)	0.127	0.040	3.175
S13G/LO BOL (Avionics white paint)	0.190	0.890	0.213
S13G/LO EOL (Avionics white paint)	0.340	0.900	0.378
Sail Dark Side	0.170	0.270	0.630
Sail Sun Side	0.090	0.025	3.600
Solar Cells	0.684 - 0.8 (function of power draw)	0.850	0.805 - 0.941
Silver Teflon Tape 5 mil (used on antenna array panels	0.090	0.780	0.115
Silverized Teflon 2 mil (used on solar panels)	0.090	0.600	0.150
ITO Silver Teflon Tape 5 mil BOL (RCS Tape)	0.090	0.780	0.115
ITO Silver Teflon Tape 5 mil EOL (RCS Tape)	0.346	0.730	0.474
SSDM-AMT Alodine (Type 1, class 3, per MIL-DTL-5541)	0.428	0.081	5.284
Stainless Steel, Passivate	0.380	0.120	3.167
Z93 (Avionics White Paint)	0.150	0.910	0.165

The avionics box has five of the six external plates painted with two different variants of white paint. The first paint selected for all surfaces was Z93 which is a ceramic based paint. This paint is one of the best with regards to radiating heat, but during application on the plates it was flaking off around narrow surfaces and edges.

This is a risk with potentially contaminating the sensitive optics on the spacecraft as well as the deployer plate of the dispenser. The top plate interfaces with the deployer plate, which has a potential risk of chattering during launch. To mitigate the risk of flaking S13G/LO white paint was chosen for the top plate and +Y plate. Since it is silicon based it is less prone to flaking. S13G/LO will experience an increase in solar absorptivity due to ultraviolet (UV) radiation; however, since the avionics is on the shaded side of the sail this is not a risk to the spacecraft. The avionics' walls are the only radiator surfaces available to the spacecraft due to the avionics being thermally decoupled from the rest of the spacecraft.

To maximize the amount of heat that can be rejected from these plates and to better thermally couple the avionics components, thermal fillers are used between the mounting surfaces of different component interfaces. Two different RTVs are used in the avionics and their properties can be found in Table 5.

Table 5: RTV Properties

	Thermal Conductivity (W/m*K)	Volume Resistivity (ohm*cm)
Nusil CV-2946	1.49	5.3x10^14
Nusil CV-2646	1	0.007

The 2646 RTV is used in bonding avionics components to plates, as the electrically conductive quality is a necessity. The 2946 RTV is used between the structural plates where it is more important to have a better thermal conductivity. Only five interface surfaces will be using the 2946 RTV due to the avionics integration process and how the structural plates fit together. These surfaces are the top plate to the +X plate and all four interfaces with the +Y plate.

One of the challenging problems is the RCS optical properties' deviation from the thermal design. The RCS has an alodine aluminum finish, and the as built RCS surface properties did not match what was assumed in the thermal model. Originally the assumed properties had α: 0.2, ε: 0.11, α/ε: 1.81 whereas the alodine has properties of α: 0.45, ε: 0.12, α/ε: 3.75. These properties were measured at MSFC on a coupon of similar metal and surface finish using the same application process and vendor as the flight hardware. This is a two times increase to the α/ε which means when exposed to the sun the surfaces will absorb two times more heat. This lead to the RCS exceeding temperature requirements during large portions of the mission. A possible solution that is being developed is using an ITO coated Ag FEP (silver Teflon) tape to increase the radiative properties of the RCS surfaces. A low a/e material was needed because the component must operate in the Sun, and not all of the alodine surface will be able to be covered with a coating.

The ITO coating is needed to help minimize surface charging early in the mission while the spacecraft is still in the Van Allen Belts, which is expected to be <1hr (e.g. 10mins), but is no longer needed once the spacecraft gets to deep space. A long term issue with the ITO coating is it experiences UV degradation that results in an increase of solar absorptivity. The End of Life (EOL) properties for the tape increase the absorptivity by a factor of three in a five year span, which results in a four times higher α/ε. The solar absorptivity lifespan is shown below with an exponential curve fit applied for estimation of properties at two and a half years and can be seen in Figure 9.

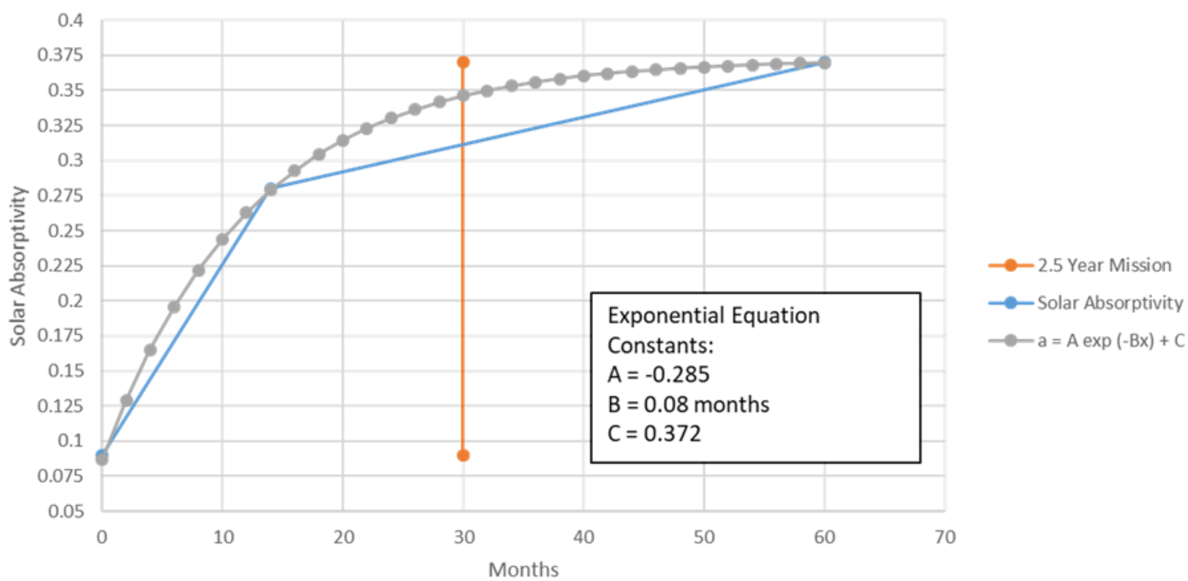

Figure 9: ITO Ag FEP 5 mil Tape Solar Absorptivity Degradation from UV Exposure

Another discrepancy exists between the design and the as built configuration optical properties for the Medium Gain Antenna (MGA). With some of the colder phases of the mission the MGA and solar panel are below their minimum Allowable Flight Temperature (AFT) operational limit. This is largely due to the optical properties of the MGA, as it is 55% of the sun facing surface area on the solar panel. Originally the thermal model assumed the MGA to have optical properties of Kapton found in Table 4, which is not a good representation with the multiple materials of the MGA in its as built configuration. For performance reasons the radio frequency radiating surfaces were not coated with the Kapton. The MGA can be seen in Figure 10. The effective MGA optical properties were found by breaking down the MGA into area percentages for the different materials. This combination led to an a/e increase from 0.136 to 0.636, which drastically increases the temperature of the solar array.

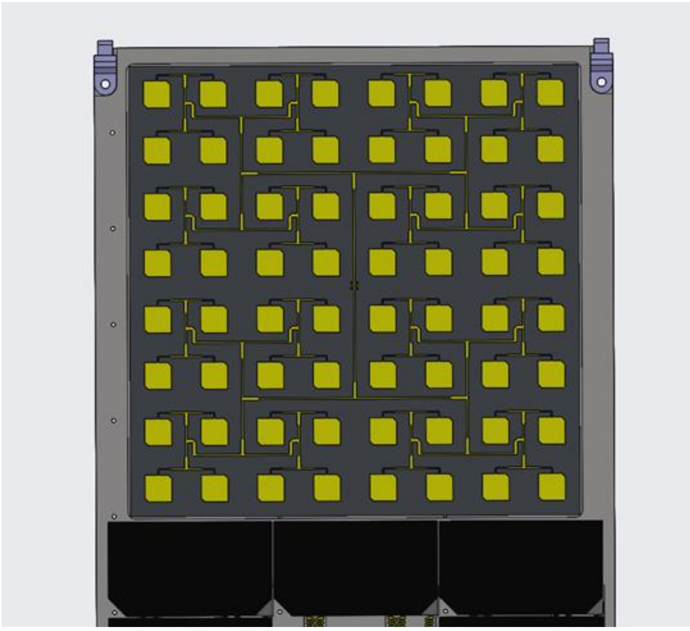

Figure 10: MGA modeled in Creo

IV. Summary of Thermal Results

The model has been analyzed by assigning each mission phase its own case set in Thermal Desktop. The majority of the mission life will be spent on an interplanetary cruise with the sail deployed, while communication and different pre-sail cases were analyzed as well. Components, such as the battery, who are sensitive to cold extremes have heaters available if needed.

Due to having a limited radiator area, most of the avionics components exceed the maximum temperature requirements during steady state runs of the communication cycle. Instead of being able to meet the operational limits of these components, the goal is to allow the spacecraft to meet its necessary communication time before these components exceed the maximum AFT operational limit. Running the model through a transient analysis shows the spacecraft can operate in a communication cycle for 80 minutes before the SSPA exceeds its maximum AFT, this is more than the desired 30 minutes.

The degradation in the ITO Ag FEP tape optical properties is also causing temperature exceedances. There is a wide range of temperatures that have to be balanced between the different mission phases. The pre-sail cases are the colder temperatures that the RCS will see, which is when the tape will have the best optical properties (lowest a/e ratio). While the hot cases are after the sail is deployed and the tape has the worst optical properties (highest a/e ratio). The thermal model results show that using the ITO Ag FEP tape the RCS will be over the maximum AFT operational limit by 20°C at the EOL. Currently alternative tapes are being looked at that have steadier BOL vs. EOL properties over time. Such as Ag FEP tape without an ITO coating. The steady state results from different phases of the mission with the as-built properties can be found in the following tables. The project levied thermal control is targeting to meet the

AFT limits that have ±10°C margin from the protoflight (PF) limits. The protoflight limits are set to the vendor specified limits.

Table 6: Steady State Cruise Temperatures

	AFT (allowable flight temp)						Cruise Margin	
	op		no op		Cruise Run		op - margin	
	cold	hot	cold	hot	cold	hot	cold	hot
Telecom								
Iris Radio	-20	50	-20	50	4.3	9.5	24	41
Low Gain Antenna	-50	80	-50	80	-8.1	35.7	42	44
Medium Gain Antenna	-50	80	-50	80	-3.7	15.2	46	65
Propulsion								
Colorless Polymer 1 (Sail)	-200	250	-200	250	-134.1	126.3	66	124
Eligiloy TRAC Booms	-215	250	-215	250	-115.4	185.9	100	64
AMT Motors	-35	40	-35	40	-22.4	-11.1	13	51
Motor Controller Board	-40	55	-55	55	0.8	5.7	41	49
ADCS								
RCS	-10	45	-24	45	15.5	35.9	26	9
RWA	-20	60	-20	60	4.5	7.6	24	52
Star Tracker	-20	60	-20	60	2.7	4.7	23	55
IMU	-40	85	-65	150	5.8	7.6	46	77
Power								
Solar Arrays	-25	90	-45	90	-3.7	58.3	21	32
EPS	-20	50	-20	50	5.7	14.0	26	36
Batteries	0	30	-10	30	-0.2	1.5	0	28
NEA Scout Instrument								
Camera	-25	50	-35	70	-0.5	1.8	25	48
Bus Electronics								
Flight Computer Board	-40	50	-40	50	5.7	10.1	46	40
Common Interface Board	-55	100	-65	110	5.7	10.6	61	89
Course Sun Sensor	-25	75	-40	85	-0.5	36.4	25	39
Reciver	-20	50	-20	50	4.4	9.1	24	41
Exciter	-20	50	-20	50	4.8	9.1	25	41
Radix	-20	50	-20	50	5.6	14.1	26	36
PSB	-20	50	-20	50	5.2	8.3	25	42
LNA	-20	50	-20	50	3.7	5.4	24	45
SSPA Board	-20	50	-20	50	3.6	5.1	24	45

Table 6 shows the results for cruise. Cruise is run with a 45° sun angle for the hot case and a 55° sun angle for the cold case and the avionics approximately producing 20W. The batteries shown here have negative margins due to having 0W of battery heater power. There is a 5W heater available if the batteries are too cold. The batteries have set points of 0°C to 30°C and is set to a 10s duty cycle. This is the majority of the spacecraft's lifespan and all other components have positive margins.

Table 7: Steady State Comm Temperatures

	AFT (allowable flight temp)						Comm Margin	
	op		no op		Comm Run		op - margin	
	cold	hot	cold	hot	cold	hot	cold	hot
Telecom								
Iris Radio	-20	50	-20	50	50.3	56.6	70	-7
Low Gain Antenna	-50	80	-50	80	-15.9	49.6	34	30
Medium Gain Antenna	-50	80	-50	80	-35.7	-31.0	14	111
Propulsion								
Colorless Polymer 1 (Sail)	-200	250	-200	250	-168.6	101.9	31	148
Eligiloy TRAC Booms	-215	250	-215	250	-134.2	162.0	81	88
AMT Motors	-35	40	-35	40	-31.6	-28.2	3	68
Motor Controller Board	-40	55	-55	55	41.5	46.2	81	9
ADCS								
RCS	-10	45	-24	45	-8.9	-3.5	1	48
RWA	-20	60	-20	60	48.3	52.9	68	7
Star Tracker	-20	60	-20	60	47.1	48.2	67	12
IMU	-40	85	-65	150	54.9	55.4	95	30
Power								
Solar Arrays	-25	90	-45	90	-35.7	-3.4	-11	93
EPS	-20	50	-20	50	54.6	67.8	75	-18
Batteries	0	30	-10	30	40.9	41.6	41	-12
NEA Scout Instrument								
Camera	-25	50	-35	70	39.3	40.4	64	10
Bus Electronics								
Flight Computer Board	-40	50	-40	50	54.0	57.3	94	-7
Common Interface Board	-55	100	-65	110	54.9	58.5	110	41
Course Sun Sensor	-25	75	-40	85	-8.6	42.8	16	32
Reciver	-20	50	-20	50	50.7	54.2	71	-4
Exciter	-20	50	-20	50	51.4	57.6	71	-8
Radix	-20	50	-20	50	53.3	62.1	73	-12
PSB	-20	50	-20	50	52.3	57.1	72	-7
LNA	-20	50	-20	50	57.4	59.8	77	-10
SSPA Board	-20	50	-20	50	62.7	81.8	83	-32

Table 7 shows the results for Earth communication (comm). Comm is run at the nominal 70° sun angle (there is a 3° pointing accuracy) for both the hot and cold cases with the avionics contributing approximately 45W. The solar arrays are showing negative margins due to the assumed Kapton properties, but the expected as built properties will raise the solar array temperatures. The majority of the avionics components have negative margins on the hot side. The worst component is the SSPA, which is the highest heat producer of the IRIS computer stack, which has a total power output of 35W[(5)]. This is the limiting component to the amount of time comm can be run, and the transient plot of the avionics can be seen in Figure 11. The transient was run powered on for four hours and powered off for four hours. The SSPA over temps at 80 minutes, which limits the rest of the avionics.

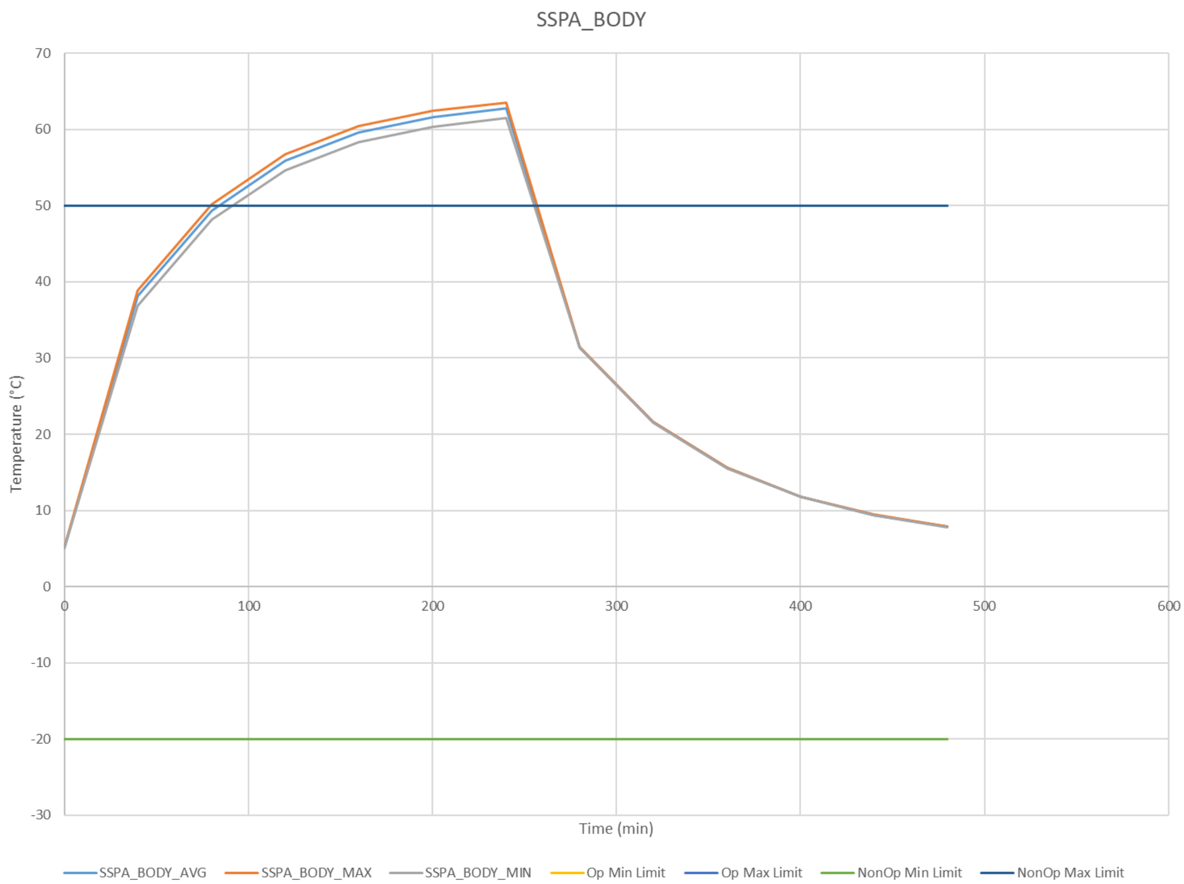

Figure 11: SSPA Body Transient Temperature Profile

Table 8: Steady State Cruise 0° Sun Angle Charging Temperatures

	AFT (allowable flight temp)						Cruise Margin	
	op		no op		Cruise Run		op - margin	
	cold	hot	cold	hot	cold	hot	cold	hot
Telecom								
Iris Radio	-20	50	-20	50	7.5	11.9	28	38
Low Gain Antenna	-50	80	-50	80	-5.4	50.2	45	30
Medium Gain Antenna	-50	80	-50	80	29.8	37.1	80	43
Propulsion								
Colorless Polymer 1 (Sail)	-200	250	-200	250	-46.8	141.0	153	109
Eligiloy TRAC Booms	-215	250	-215	250	-86.2	201.2	129	49
AMT Motors	-35	40	-35	40	-2.7	0.5	32	40
Motor Controller Board	-40	55	-55	55	4.1	8.2	44	47
ADCS								
RCS	-10	45	-24	45	45.2	50.5	55	-5
RWA	-20	60	-20	60	7.8	10.1	28	50
Star Tracker	-20	60	-20	60	6.0	7.2	26	53
IMU	-40	85	-65	150	9.1	10.0	49	75
Power								
Solar Arrays	-25	90	-45	90	29.8	86.1	55	4
EPS	-20	50	-20	50	9.0	16.5	29	33
Batteries	0	30	-10	30	3.0	4.0	3	26
NEA Scout Instrument								
Camera	-25	50	-35	70	2.8	4.3	28	46
Bus Electronics								
Flight Computer Board	-40	50	-40	50	9.0	12.6	49	37
Common Interface Board	-55	100	-65	110	9.0	13.1	64	87
Course Sun Sensor	-25	75	-40	85	2.8	51.4	28	24
Reciver	-20	50	-20	50	7.7	11.5	28	38
Exciter	-20	50	-20	50	8.0	11.6	28	38
Radix	-20	50	-20	50	8.9	16.5	29	33
PSB	-20	50	-20	50	8.4	10.8	28	39
LNA	-20	50	-20	50	7.0	7.8	27	42
SSPA Board	-20	50	-20	50	6.9	7.6	27	42

Table 8 shows the temperature results for charging during cruise. This is also the safe mode for the spacecraft and is oriented 0° to the sun with the same power draw as nominal cruise with approximately 20W. This is the hottest case for the RCS and with the EOL ITO Ag FEP tape properties, it is exceeding temperature limits. Tape alternatives (e.g. silver Teflon without an ITO coating) are being considered at this time.

Table 9: Steady State Comm Pre-Sail Deployment

	AFT (allowable flight temp)						Comm Margin	
	op		no op		Comm Run		op - margin	
	cold	hot	cold	hot	cold	hot	cold	hot
Telecom								
Iris Radio	-20	50	-20	50	61.5	69.1	82	-19
Low Gain Antenna	-50	80	-50	80	4.1	61.7	54	18
Medium Gain Antenna	-50	80	-50	80	-53.0	-48.6	-3	129
Propulsion								
Colorless Polymer 1 (Sail)	-200	250	-200	250	0.0	0.0	200	250
Eligiloy TRAC Booms	-215	250	-215	250	0.0	0.0	215	250
AMT Motors	-35	40	-35	40	90.4	98.3	125	-58
Motor Controller Board	-40	55	-55	55	52.5	58.3	93	-3
ADCS								
RCS	-10	45	-24	45	14.9	20.5	25	25
RWA	-20	60	-20	60	61.2	67.1	81	-7
Star Tracker	-20	60	-20	60	58.7	60.4	79	0
IMU	-40	85	-65	150	67.8	69.3	108	16
Power								
Solar Arrays	-25	90	-45	90	-53.0	-19.0	-28	109
EPS	-20	50	-20	50	65.9	80.3	86	-30
Batteries	0	30	-10	30	53.7	55.5	54	-26
NEA Scout Instrument								
Camera	-25	50	-35	70	51.1	53.1	76	-3
Bus Electronics								
Flight Computer Board	-40	50	-40	50	65.4	69.8	105	-20
Common Interface Board	-55	100	-65	110	66.1	71.0	121	29
Course Sun Sensor	-25	75	-40	85	15.1	54.6	40	20
Reciver	-20	50	-20	50	62.0	66.7	82	-17
Exciter	-20	50	-20	50	62.9	70.1	83	-20
Radix	-20	50	-20	50	64.6	74.5	85	-24
PSB	-20	50	-20	50	63.7	69.6	84	-20
LNA	-20	50	-20	50	69.2	72.3	89	-22
SSPA Board	-20	50	-20	50	74.2	94.2	94	-44

Table 9 shows the results for comm prior to the sail being deployed. The spacecraft is at a 70° sun angle with 45W being dissipated by the avionics. This is similar to the nominal comm cycle however temperatures are higher due to the avionics not being shielded from the sun by the solar sail. Similar to the comm results with the sail deployed (Table 7), the actual comm length will be limited to a specific duration as shown in Figure 11.

Table 10: Steady State Comm 0° Sun Angle Pre-Sail Deployment

	AFT (allowable flight temp)						Comm Margin	
	op		no op		Comm Run		op - margin	
	cold	hot	cold	hot	cold	hot	cold	hot
Telecom								
Iris Radio	-20	50	-20	50	51.1	57.7	71	-8
Low Gain Antenna	-50	80	-50	80	-7.4	50.7	43	29
Medium Gain Antenna	-50	80	-50	80	5.1	11.2	55	69
Propulsion								
Colorless Polymer 1 (Sail)	-200	250	-200	250	0.0	0.0	200	250
Eligiloy TRAC Booms	-215	250	-215	250	0.0	0.0	215	250
AMT Motors	-35	40	-35	40	0.8	5.9	36	34
Motor Controller Board	-40	55	-55	55	42.6	47.4	83	8
ADCS								
RCS	-10	45	-24	45	-2.0	3.6	8	41
RWA	-20	60	-20	60	49.2	53.9	69	6
Star Tracker	-20	60	-20	60	48.0	49.3	68	11
IMU	-40	85	-65	150	55.9	56.5	96	28
Power								
Solar Arrays	-25	90	-45	90	5.0	56.6	30	33
EPS	-20	50	-20	50	55.4	68.8	75	-19
Batteries	0	30	-10	30	41.7	42.5	42	-13
NEA Scout Instrument								
Camera	-25	50	-35	70	40.3	41.5	65	8
Bus Electronics								
Flight Computer Board	-40	50	-40	50	54.8	58.3	95	-8
Common Interface Board	-55	100	-65	110	55.8	59.6	111	40
Course Sun Sensor	-25	75	-40	85	-1.1	44.2	24	31
Reciver	-20	50	-20	50	51.5	55.2	71	-5
Exciter	-20	50	-20	50	52.3	58.6	72	-9
Radix	-20	50	-20	50	54.2	63.1	74	-13
PSB	-20	50	-20	50	53.2	58.1	73	-8
LNA	-20	50	-20	50	58.4	61.0	78	-11
SSPA Board	-20	50	-20	50	63.7	83.0	84	-33

Table 10 shows the comm cycle prior to the solar sail being deployed in a charging mode. The spacecraft is at 0° to the sun while still dissipating 45W from the avionics. Again, the avionics are over temperature, but not as high as the previous case due to the avionics not having a direct view to the sun. For this run, the RCS was ran with BOL properties. The RCS is the only component that is ran with BOL (Table 10) and EOL (Table 8) properties. As shown in the property table (Table 4) the only other property with specified degradation is the solar absorptivity of the S13G/LO white paint. However this paint is only on the avionics box, which is on the shaded side of the vehicle so it will not receive any solar flux once the sail is deployed.

V. Thermal Testing Overview

During hardware development, thermal testing was used to mature the design of each subsystem[3,4]. For the NEA Scout flight hardware, thermal testing will occur at the subsystem and integrated flight system level. Figure 12 shows a high-level overview of the test campaign for the flight system. Subsystem test will be to their proto-flight temperature limits, which are the AFTs plus 10°C margin. For many off the shelf items, the proto-flight temperature was set to be equal to the vendor specification. The integrated flight system test will be to the flight acceptance temperature, which are AFT plus 5°C margin

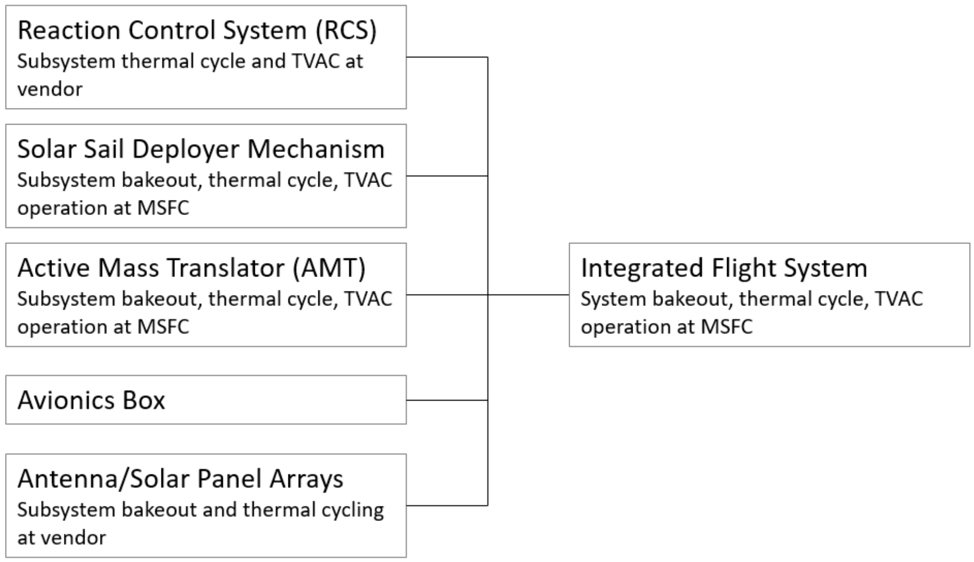

Figure 12. High-Level Overview of Thermal Testing for Flight Hardware

The integrated flight system thermal test will occur at MSFC's Environmental Test Facility (ETF) in either the Sunspot or V7[6] chamber. The purpose of the test is to thermal cycle the hardware, verify operation on a flight like environment, and provide data for thermal model correlation. Because the chamber uses a liquid nitrogen shroud a hot box will be used to simulate the sink temperatures of the spacecraft during flight. Infrared lamps were not chosen because of the additional work that is needed to characterize them to provide good data for model correlation.

The hot box that will be used during the flight systems integrated Thermal Vacuum (TVAC) test is based on the design that was used for the Fast, Affordable, Science and Technology Satellite (FASTSAT) project, Figure 13. It is a six sided aluminum box that is painted black and with Clayborn heater tape[7] installed. The design of the NEA Scout hot box is shown in Figure 14.

The test profile for the flight systems test will include at least three thermal cycles, including at least one cycle to non-operational temperature. Functional tests will be done to verify operation of the spacecraft at the maximum and minimum of each cycle to operational temperatures. Additionally, during at least one cycle, the spacecraft will be operated to allow the spacecraft to reach steady state so data for thermal model correlation can be obtained. A specific case to correlate too has not been chosen yet. It will likely be the cruise case and/or comm case. A sink temperature to run the hot box at has not been determined yet, but the thermal model will be used to find one. This will be done by running the model with no power input to see float temperature of the spacecraft, or using the TSINK command in Thermal Desktop. During the integrated TVAC test the booms and sails will be stowed. The booms will be verified to operate as part of their subsystem flight acceptance TVAC testing; though a full deployment is not possible, so only 6 inches will be deployed. A partial deployment of the sail under TVAC occurred as part of engineering development work.

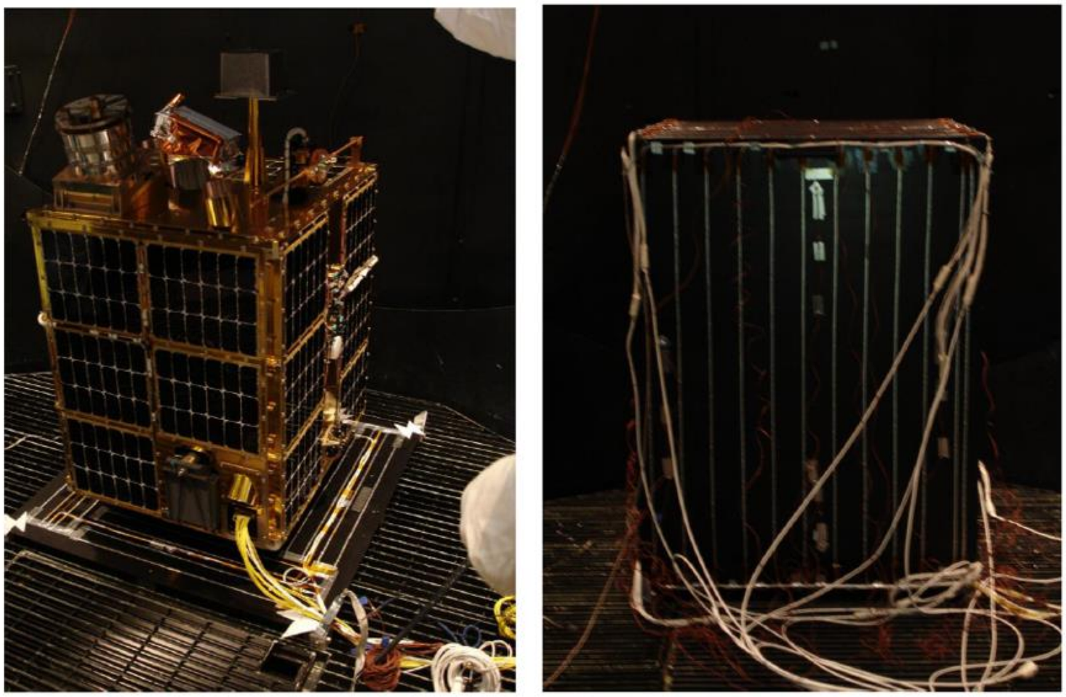

Figure 13. Showing FASTSAT Hot Box. A Similar Hot Box will be Built for NEA Scout

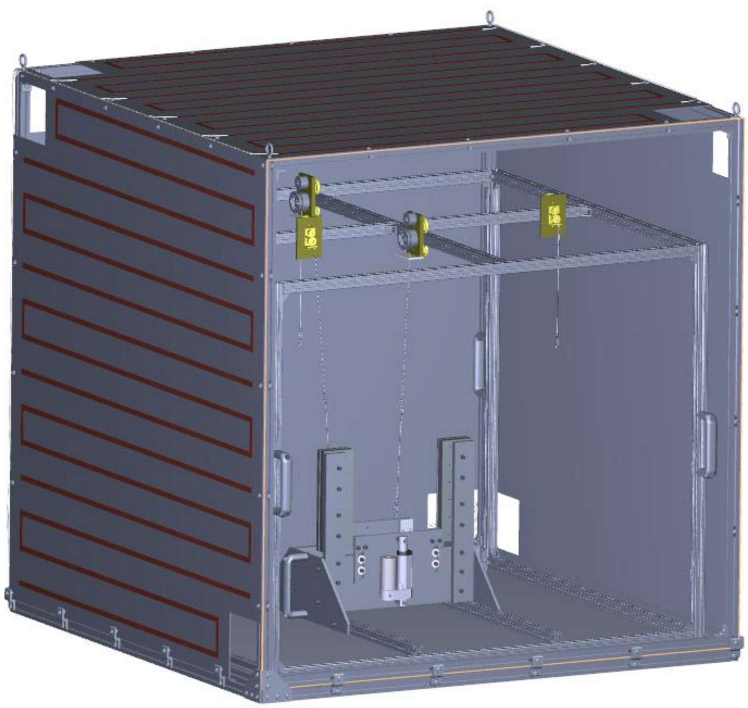

Figure 14. NEA Scout Hot Box. With One Side Removed, and Showing the Deployment Fixture

VI. Lessons Learned

- Always double check assumed optical properties. The difference between what was assumed during design and the as built configurations can lead to large temperature changes.
- Take into consideration EOL and BOL optical properties. Do not assume properties stay constant. Common resources used to look up properties often do not have EOL/BOL so be sure to talk to a coatings expert about potential degradation.
- Changes to the thermal architecture after a Critical Design Review (CDR) are risky because they will not be reviewed to the same level that occurred at CDR.
- CubeSats are very sensitive to overall dimensions, be sure to claim part of the allowable envelop to place tapes and coatings early on in the design cycle. Coming in during manufacturing saying that you need to add 7mils of tape may cause envelop exceedances.
- Include applications of thermal fillers (e.g. RTV's) early in discussions on assembly work flow.
- Ceramic based paints (e.g. Z93) are best applied to large acreage with minimal edges, and not to surfaces that have lots of penetrations or edges.

VII. Conclusion

NEA Scout poses a unique thermal control challenge with the use of the solar sail for its primary propulsion source. This sail adds complexity to the different mission cycles with having to take into consideration the effects of components getting direct views to the sun early in the mission timeline, and being completely shaded later in the mission. Most components are currently predicted to meet their AFT limits while the maximum temperature limit exceeding components during the high power draw cycles are not predicted to impact the operation durations.

VIII. Acknowledgements

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