EcAMSat – NASA's first 6U Biological Spacecraft: System Integration and Environmental Test Technical Paper

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Abstract/Introduction

The E. coli AntiMicrobial Satellite (EcAMSat) is the first 6U biological spacecraft designed at NASA Ames Research Center. The EcAMSat mission leveraged heritage from past cubesat spacecraft such as GeneSat1, Organism/Organics Exposure to Orbital Stresses (O/OREOS), PharmaSat, and Microsatellite in-situ Technologies (MisST). The EcAMSat design has defined the 6U industry standard now widely utilized in public and private industry. This technical report will cover the challenges overcome and accomplishments in successfully flying NASA's first 6U biological mission. Mission operations and science information is covered in additional reports.

EcAMSat was built and designed following the tailored NASA procedural requirements found in NPR 7120.5E and based on standards for Class D missions found in NPR 8705.4. Additionally, tailoring was defined with Ames Procedural Requirement APR 8070.2.

Mechanical Design Approach

The physical design requirements of the 6U were derived from two factors:

First, to support a platform compatible with multiple payloads, the team needed to ensure ample power was available to complete the mission experiment. The original design had called for two 3Us mechanically connected together with deployable solar panels. The team created space between the two payloads for the solar panel deployment mechanism. When the deployable solar panel design was found to be insufficient in meeting power requirements, the decision was made to revert back to using six heritage body mounted 3U solar panels. This resulted in a new engineering design effort to develop a solar panel attachment device to tie the 3U payloads together.

Second, it was determined that extending the length of the spacecraft to be longer than a traditional 3U would be beneficial for a global 6U standard, to accommodate many future 6U design solutions in the industry. This originally came from the NASA designed NLAS (Nanosat Launch Adapter System) dispenser. When the NLAS dispenser was initially being designed, NASA Ames was performing the build for the O/OREOS spacecraft. Given that mission was to be launched to a high inclination, it was determined the spacecraft needed a deorbit mechanism to ensure it would reenter Earth's atmosphere within the required 25-year deorbit window. The team had to modify an NLAS dispenser to accommodate a deorbit mechanism with regard to the back panel, creating additional space for a new deorbit design.

The design for EcAMSat was reengineered and the second payload was removed with the new design consisting of one 2U payload and a 1U BUS on one side and a mass sim on the other. The additional solar panels for the 6U configuration were kept as is to ensure ample power for the single remaining payload.

ADCS changes - Hysteresis Rods and Permanent Magnet Engineering

EcAMSat utilizes heritage passive attitude control design of GeneSat, PharmaSat, and O/OREOS, using magnets and hysteresis rods to align the spacecraft to the Earth's magnetic field. This allows the satellite to better align with Santa Clara University ground station antennas with two passes a day including patch antenna alignment and omni-directional beacon transmission. During initial deployment of the spacecraft, the hysteresis rods are effective in dampening oscillations of the spacecraft pointing while permanent magnets are aligning with the Earth's magnetic field lines during initial tip-off tumbling phase. The systems dampening effect stabilizes the spacecraft to allow nominal spacecraft transmission through the S-band transceiver.

The initial mechanisms of de-tumbling the spacecraft during deployment was designed to convert to a spin about the z-axis due to the pull of the magnets. Without hysteresis rods, the spin axis oscillates about the Earth's magnetic field lines. The hysteresis rods dampen the oscillation while at the same time slowing the spin. Residual magnetism in the rods produced an angular offset in the final spin axis, which effectively defines a boundary condition upper limit with respect to the number of rods that should be implemented.

Overall, the heritage GeneSat/PharmaSat magnet and rod design were used in EcAMSat, but additional modifications to quantities and geometry were changed to make up for the increase in mass and moment of inertia for the 6U configuration. Although the spacecraft mass was approximately doubled, the moment of inertia increased nonlinearly, leading to changes in the attitude control dynamics in the 6U versus the 3U. Increasing the number of magnets and rods in proportion to the increase in moment of inertia was unrealistic due to the mass feeding back into further increases in moment of inertia. A fixture was developed for the auxiliary payload volume to be populated with a combination of carefully located and aligned magnets, rods, and ballast to locate the satellite center of mass.

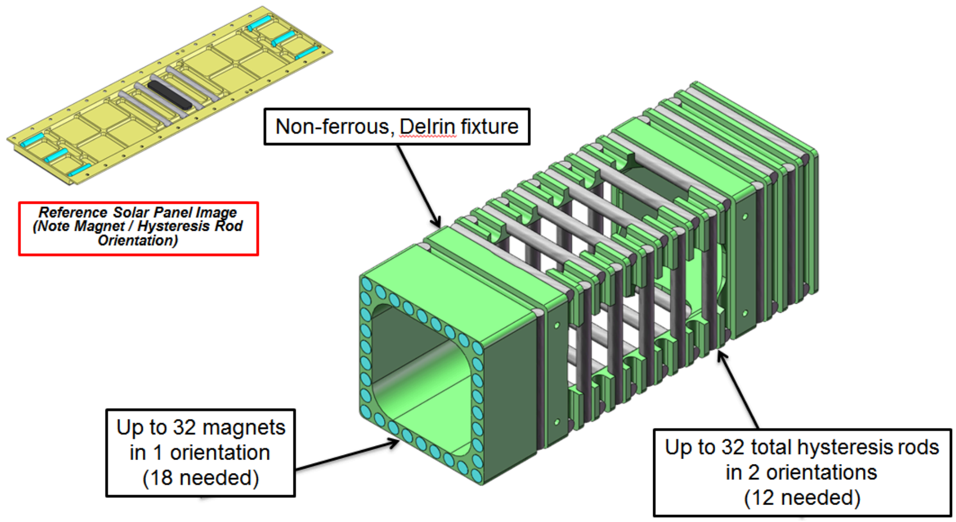


Figure 1 Mass Sim with magnets and hysteresis rods

Simulations utilizing MatLab and STK were performed to determine adequate quantities of magnets and rods to meet the de-tumble time, spin rate, and z-axis alignment requirements of the EcAMSat science mission (3 days to reach 0.001-g from a 5 degree/sec tipoff; +/- 20 degrees max misalignment with magnetic field), given possible deployment tipoff rate and expected orbit scenarios. Through an engineering control board configuration managed process, the number of permanent magnets were increased from PharmaSat's 24 to 54 for the EcAMSat configuration. In addition, the number of hysteresis rods were increased from PharmaSat's 16 to 36 for EcAMSat.

Software Rework

Initial software was modified from baselined flight code developed for prior NASA Ames missions Genesat, PharmaSat and O/OREOS. The software platforms were written for multiple PIC18 microcontrollers in the embedded C programming language. The codebase was modified with updates to satisfy new mission experiment payload protocol requirements. Given the different organism and new science experiment phasing protocol much of the software had to be modified to accommodate timing of events and trigger actions. The duration of each experiment phase for feeding, growth, and absorption were all adapted to meet the new science requirements. In addition, pumping speeds were changed to be pulse modulated to slow down the pumping phases and not over pressurize the system.

There was also one new ISS levied requirement that required a specific fix. The radio for the spacecraft was required to remain off after deployment for 90 minutes. Previously, the radio began broadcasting immediately after being deployed.

Electrical & Thermal Rework

A Tiger Team was formed in Mid-July of 2015 to address an issue that had flight implications due to new launch altitude: The payload heater's duty cycle was too high during our environmental testing of new flight dynamics. The duty cycle was predicted at 90% which exceeded the mission margin requirements. The power consumption resulting from this duty cycle would potentially run the battery to an unacceptable depth of discharge. The mission of the team was to search for minimally intrusive alterations to the spacecraft that would lower the payload heater's duty cycle to 70%, or less.

The root cause of the issue was diagnosed and it was determined that the temperature of the batteries fell to levels that would degrade the battery performance (below 0°) during a 450x720 orbit.

The issue was first discovered through the Thermal Vacuum Power Management (TVPM) test. The Tiger Team assembled and reviewed every aspect of the spacecraft model:

1. Mechanical interfaces
2. Optical properties
3. Electrical operations
4. Orbital operations

Table 1 is a summary of the updates made to the model post-TVPM.

Item	TVPM Model	Post-TVPM Model Updates
Energy transferred from the remote regulator (connected to the stepper pump) into the payload can	Not included	Included
Titanium screws and Ultem washers	Not included	Included
Conduction of Titanium screws	Factor of 9 lower than the correct value	Correct value
Surface between the bus assembly and patch antenna endplate	Included	Removed
Payload Board Mount Assembly (M342) and Detector Board Mount Assembly (M344)	Modeled as Al-6061	Modeled as Delrin
Auxiliary bus endplate	Not thermally connected to the satellite	Thermally connected through metal-to-metal contact between the endplate and auxiliary payload can
Thermal spreader temperature control sensor	Control was assigned to only one side of the fluidics card	Control was reassigned to both sides of the card

Table 1 Updates to the S/C thermal model (TVPM to Post-TVPM)

A re-correlation was performed after ALL updates were made. The re-correlation was made to first determine and quantify the performance of the spacecraft as tested during TVPM. From this point forward, changes could then be made with confidence to achieve the goal of lowering the fluidic heater's duty cycle from 90% to 70%. Table 2 summarizes the thermal model's predictive accuracy.

Sheet:	Cold	Hot	Cold	Hot	Cold	Hot
Parameter:	Avg	Avg	Actual	Actual	T(P-A)	T(P-A)
Node/Group	*[°C]*	*[°C]*	*[°C]*	*[°C]*	*[°C]*	*[°C]*
Solar Panel						
Solar 1A	-8.6	14.1	-8	17.1	-0.6	-3.0
Solar 1B	-6.2	16.2				
Solar 2	-9.1	13.6	-12	13.3	2.9	0.3
Solar 3A	-8.9	13.9				
Solar 3B	-6.1	16.3	-9	16.5	2.9	-0.2
Solar 4	-6.1	16.5	-5	21.1	-1.1	-4.6
Solar - P11	-6.2	16.2	-3	20.1	-3.2	-3.9
Solar - P9	-6.1	16.5	-3	21.2	-3.1	-4.7
Solar - P12	-9.1	13.6	-8	15.9	-1.1	-2.3
Solar - P14	-6.2	16.3	-7	16.7	0.8	-0.4
Payload						
Payload Can Top	-3.0	21.6	-3	22	0.0	-0.4
Battery Cells						
Pri Cell	3.2	23.5	2	26	1.2	-2.5
Aux Cell	1.1	23.0	3	25	-1.9	-2.0
Bus PCB						
CD&H	-3.7	18.2				
EPS	6.9	28.9				
Backplane	-1.8	20.0				
MicroHard	10.3	27.4	11	28	-0.7	-0.6
Chamber and Heater Enclosure						
Chamber - Top	-91.2	-79.3	-86	-74	-5.2	-5.3
Enclosure - Top	-21.0	3.9	-20	5	-1.0	-1.1
Enclosure - Bot	-4.9	21.7	-7	22	2.1	-0.3
Enclosure - Side 1	-16.2	10.5	-12	12	-4.2	-1.5
Enclosure - Side 2	-15.5	11.5	-15	15	-0.5	-3.5

Table 2 Re-correlation of the TVPM thermal model predictions

Initially, the team spent some effort trying to find heat sources within the bus to create thermal bridges from those sources to the battery packs such as increasing the duty cycle of the radios to generate more heat.

The team realized the baseline temperature of the satellite was too low, and most of the team's work was trying to compensate within each subsystem instead of addressing the root problem. The team started looking at increasing the temperature without using powered heaters. This required a new thermal approach.

We looked at changing the way the thermal energy flowed through the satellite. The payload container is thermally isolated from the rest of the structure using joints with Ultem washers and titanium screws. The team looked at using similar joints in other areas and changing larger aluminum parts to titanium. Lastly, we looked at the impact of altering the surface treatments on the exterior panels.

To increase the baseline temperature, the satellite needs to absorb more heat and/or reject less heat. Several different selective surfaces were evaluated: black nickel; black chrome; vapor-deposited gold; electro-deposited gold; Laser gold ™; Catalac Black paint; copper foil tape; anodized versus alodined aluminum; and even “Black Magic”. The absorption and emissivity properties were compared, and it turned out that the ratio of the two properties is key. It does not help much if the absorption is changed, but the emissivity changes with it.

The satellite had already been qualified for flight and the team wanted to avoid invalidating the completed testing by making significant structural changes. This desire, in addition to the absorption/emissivity ratio, led us to copper tape. The thermal model was run with copper tape values and came back with acceptable baseline temperatures for the assigned orbit. This amounted to almost a cosmetic change to the structure, none of the previous qualification testing needed to be repeated. This modification was evaluated as having minimal design parameters and was determined not to require a flight requalification.

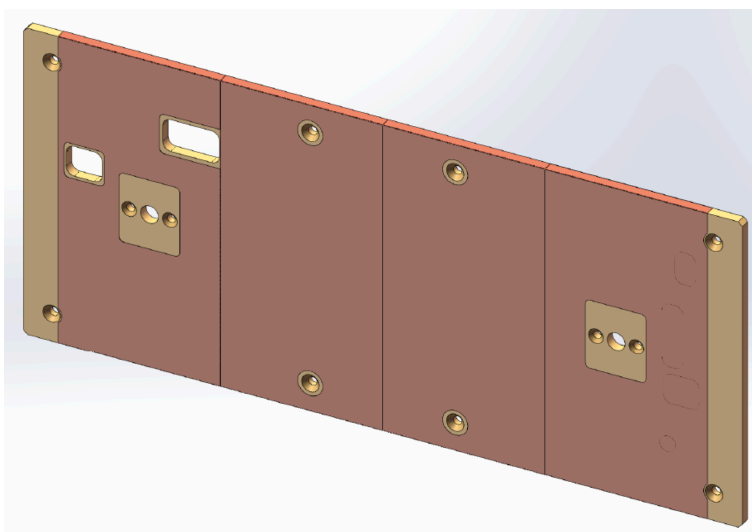


Figure 2 Example Panel with Copper Foil Tape Over Gold Plating

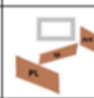
With the new higher baseline temperature, the thermal/power model closed without requiring an additional battery pack, and the battery pack temperature no longer dropped dangerously low. Several design challenges were resolved by the simple application of copper foil tape to retune the thermal balance.

A series of parametric cases were made to determine the effect on coating selection on heater duty cycle. Again, the objective was to manipulate the optical finishes of the exposed external surfaces of the spacecraft that resulted in heater duty cycles of 70% or less. The eligible surfaces considered were as follows:

1. Auxiliary Close Plate
2. Solar Panel Attachment Plate (2 locations)
3. Payload Closeout Plate
4. Antenna Closeout

Figure 4 summarizes the various cases as the specific spacecraft surfaces were modified. After inspection of all temperatures and heater duty cycles, it was determined that applying copper foil tape to the first three surfaces on the list was an adequate solution. The Antenna Closeout was left as white paint.

Number of cases run since SC3	Case No.	Changes to Model from SC3 Model	Surface Treatment(s)	Diagram
1.	3	None	50% striping of Gold and Black nickel on Payload	XXX
2.	4	Simulation: 500,000 – 600,000 seconds	None	None
3.	4x	Simulation: 500,000 – 600,000 seconds	None	None
4.	4.1	Simulation: 600,000 – 800,000 seconds	T Cu Payload	XXX
5.	4.2	See Case No. 4.1	T Cu Payload + Bus panel	XXX
6.	4.3	See Case No. 4.1	T Cu Payload + Four Al washers on PL endplate	XXX
7.	4.5	See Case No. 4.1	T Cu Payload + Bus + Auxiliary	
8.	5	XXX	None	None
9.	5.1	See Case No. 5	T Cu Payload	
10.	5.1.1	See Case No. 5	T Cu Payload + 4 Al washers on PL endplate	
11.	5.2	See Case No. 5	T Cu Payload + Bus closeout	
12.	6	Added to the model? Silver epoxy between patch antenna back layer and outer bus frame, Aaron, any corrections or other changes?	None	None
13.	6.1	See Case No. 6	T Cu Payload + Solar attachments	
14.	6.2	See Case No. 6	T Cu Payload + Auxiliary + Bus closeout	
15.	6.3	See Case No. 6	T Cu Payload + Auxiliary + Solar attachments	
16.	6.3.1	See Case No. 6	T Cu Payload + Auxiliary + Solar attachments + Bus closeout	
17.	6.3.2	See Case No. 6	R Cu Payload + Auxiliary + Solar attachments + Bus closeout	
18.	6.3.3	See Case No. 6	R Cu (Payload + Auxiliary + Solar attachments) + Alodine (Bus inner + outer)	
19.	6.3.4	See Case No. 6	R Cu (Payload + Auxiliary + Solar attachments) + Alodine (Bus outer) + White paint (Bus inner)	
20.	6.4	See Case No. 6	T Cu Payload + Auxiliary	
21.	7	Simulate Model 7c. Remote regulator values made more accurate: thermal model accounted for the remote regulator providing only 5V of energy into the payload can, as opposed to the 8V that was previously	None	None

		modeled. Additional heat sources were added next to the fluidics card. Aaron, any corrections or other changes?		
22.	7.1	See Case No. 7	R Cu (Payload + Auxiliary + Solar attachments) + Alodine (Bus inner + outer)	
23.	7.2	See Case No. 7	R Cu Payload + Auxiliary + Solar attachments + Bus closeout	
24.	7.3	See Case No. 7	R Cu Payload + Solar attachments + Bus closeout	
25.	7.4	See Case No. 7	Clear Alodine Payload + Solar attachments + Auxiliary	
26.	7.5	See Case No. 7, Variable voltage (Chris, Aaron? More info)	None	None

Key:

Alodine **White paint** Tarnished Copper (T Cu) **Regular Copper** (R Cu)

= unchanged part

Figure 3 Parametric Case Run Matrix

The surfaces shown on Figure 4 were changed to Copper Foil:

1. Aux Closeout
2. Solar Panel Attachment plate (2)
3. Payload Closeout

These simple changes showed analytically that the duty cycle of the fluidic card heater would be lower. Figure 4 shows that the predicted duty cycle was 59%, which was lower than the goal of 70%.

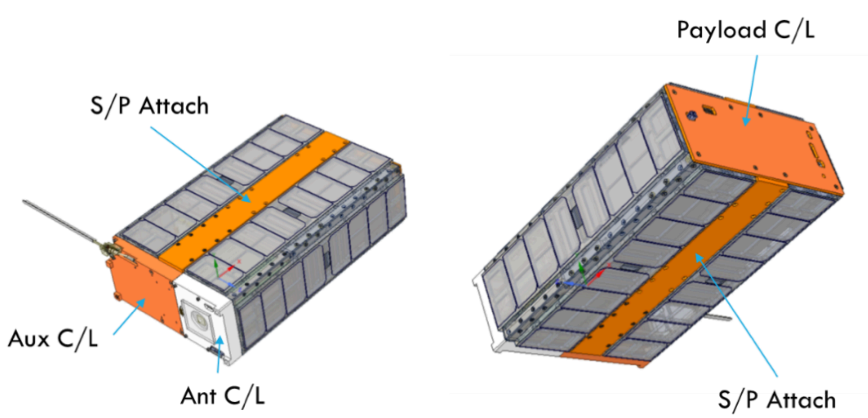


Figure 4 Surface taped with copper foil

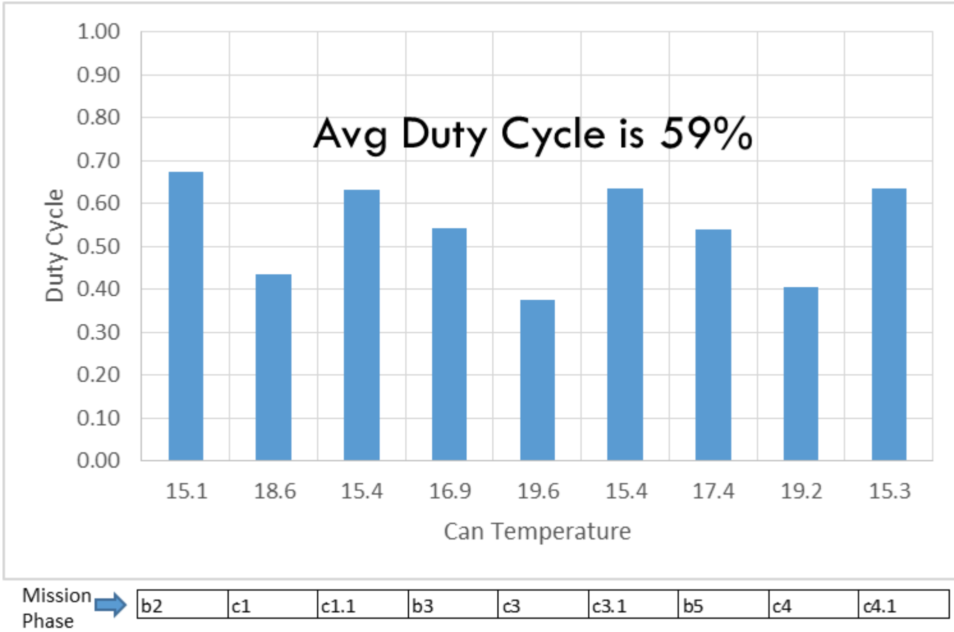


Figure 5 Fluidic card heater duty cycle a function of phase

The thermal model was built and solved with Thermal Desktop. The environmental conditions were derived from solar vectors calculated in STK. Vectors were then loaded into Thermal Desktop where solar environmental loads on surfaces were calculated.

Environmental Test Program

EcAMSat's Environmental Test Plan was conducted in accordance with NASA General Environmental Verification Standard (GEVS): GSFC-STD-7000.

ESS (Environmental Stress Screening)

The intent of the ESS test is to validate workmanship and locate potential hardware failures by increasing thermal margins beyond test-like-you-fly conditions.

ESS thermal cycling was completed on all components of the spacecraft. This NASA designed test verifies functionality of the spacecraft through the nominal temperature range of the mission. It is based on predicted external satellite temperatures developed through Thermal Desktop simulations.

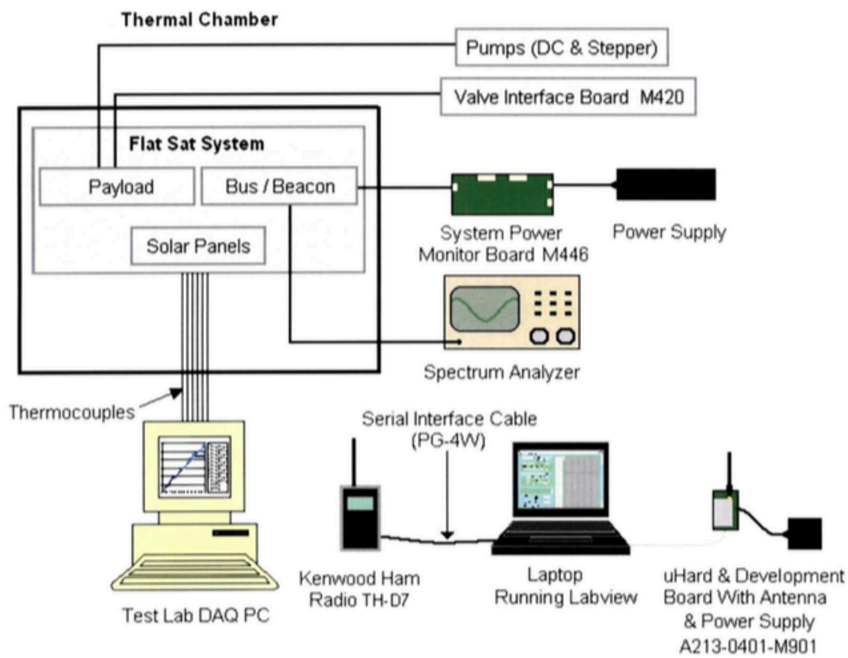


Figure 6 ESS Test setup block diagram

The test setup involves placing the powered spacecraft in a thermal chamber and cycling through 8 hot and cold 30-minute periods. Total time for this test including the ramp times is roughly 14 hours. Pre- and post- functional tests are performed on the spacecraft to verify nominal functionality and check for damage. Additionally, 12 thermocouples were placed in various locations on the spacecraft to monitor the temperature of the components throughout the test.

Sensor	Location
Thermocouple1	Payload Board
Thermocouple2	C&DH
Thermocouple3	EPS
Thermocouple4	Battery Pack (On battery if possible)
Thermocouple6	uHard
Thermocouple7	Beacon Board
Thermocouple8	Detector Payload Control Board
Thermocouple9	Solar 1
Thermocouple10	Solar 2
Thermocouple11	Solar 3
Thermocouple12	Solar 4

Table 3 Thermocouple locations

The spacecraft starts at ambient temperature and is ramped up to 45°C at a rate of 3°C per minute. Once the target is achieved, the temperature is held for 45 minutes. Battery temperature should not exceed a charging temperature range of 0°C to 45°C, and a discharging range of -10 to 60°C or the test will be halted. If the battery temperature drops below 5°C on the cooling cycle, the external power supply will be shut off. If the battery rises above 40°C on the heating cycle, the external power supply will be shut off.

After the initial ramp-up the temperature is reduced down to -10C at the same rate and held for 45 minutes. All thermocouples are closely monitored by two team members at all times.

Health and status of the system will be recorded throughout the test and monitored at the end of hot and cold dwell cycles using the uHard radio. The beacon signal will be monitored with a portable ham band radio during duration of the test. The audible acknowledgement of beacon signal at least every 15 minutes is noted.

TVPM (Thermal Vacuum Power Management).

The intent of the TVPM test is to validate the thermal and power models within a full mission scenario. This test is conducted to assess the power balance of the spacecraft when subjected to flight-like thermal conditions. Power scenarios are based on predicted on orbit operations. This is not a workmanship test but a test as you fly scenario.

Test Configuration

EcAMSat was assembled in a flight configuration with the following exceptions:

1) The payload was not loaded with biology
2) The kill switch was wired to an external kill switch
3) Solar Panel 1 input was brought out of the spacecraft to enable an external charge source
4) The footswitch was wired to the M446 diagnostic board, with a sense resistor at the connector.

The spacecraft was charged using a current-limited power supply, set to 10 V and 1.35 A, hooked up to a garden timer to simulate 60 minutes of charging followed by 30 minutes of eclipse, shown in Figure 3 on the right side of the photo.

The test was conducted at ARC in the Engineering Evaluation Lab (EEL). A vacuum chamber and cold plate were used to drive the spacecraft to expected flight temperatures; see Figure 1.

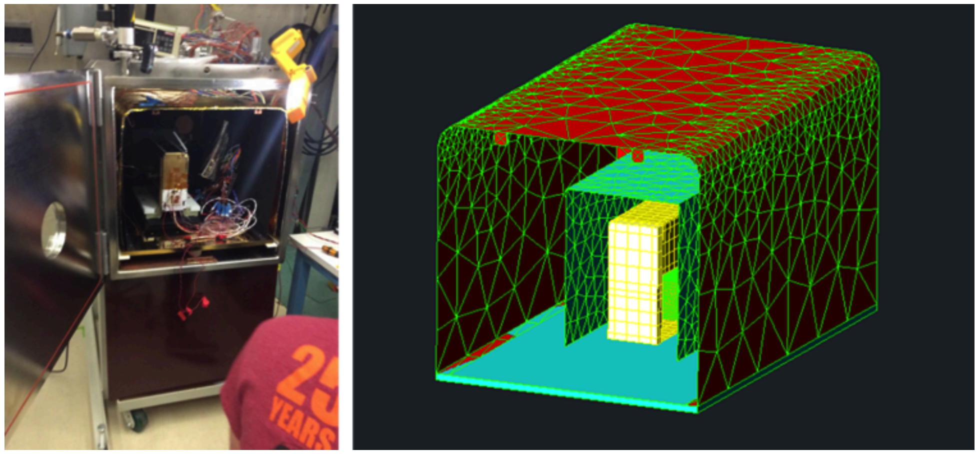


Figure 7 Vacuum Chamber and Thermal Representation

The spacecraft was placed inside a heater-controlled enclosure to achieve and maintain desired temperatures; see Figure 2. The cold plate was mainted at -115°C throughout the duration of the test with a vacuum of at least 10^{-4} Torr.

Data telemetry was provided by both the spacecraft bus and by EEL installed thermocouples. A comprehensive list is shown on Table 1.

S/C Telemetry	EEL Thermocouples
Bus Temperture [P11] [T1] [Solar 1A]	COWL-LEFT
PL Can Outside Bus Temperture [P10] [T2] [Bus Temp 2]	COWL-RIGHT
Bus Temperture [P12] [T5] [Solar 2]	COWL-FRONT
Bus Temperture [P13] [T6] [MicroHard]	COWL-BACK
Bus Temperture [P14] [T7] [Solar 3B]	COWL-TOP
Bus Temperture [P15] [T8] [AUX, PLY]	COWL-BOTTOM
Bus Temperture [P9] [T11] [Solar 4]	PLATE
Bus Temperture [P8] [T12] [Pri. Battery Holder]	SIDE WALL
Payload Pressure Vessel Temperature	SHROUD TOP
Solar Panel 1 Current	SOLAR 3B
Solar Panel 2 Current	SOLAR 1A
Solar Panel 3 Current	BUS END
Solar Panel 4 Current	PAYLOAD END
Solar Panel 1 Voltage	SOLAR 2
Solar Panel 2 Voltage	SOLAR 4
Solar Panel 3 Voltage	PL CAN TOP
Solar Panel 4 Voltage	AUX BATTERY
MHX Current Draw	BATTERY PACK
Sensors and Muxes Current Draw	UHARD
C&DH Current Draw	
Payload Heater Current Draw	
Payload Current Draw	
Beacon Crrent Draw	
MHX Voltage	
Sensors and Muxes Voltage	
Bus Voltage (BusV)	
Battery Voltage (BattV)	

Table 4 – Spacecraft Telemetry List

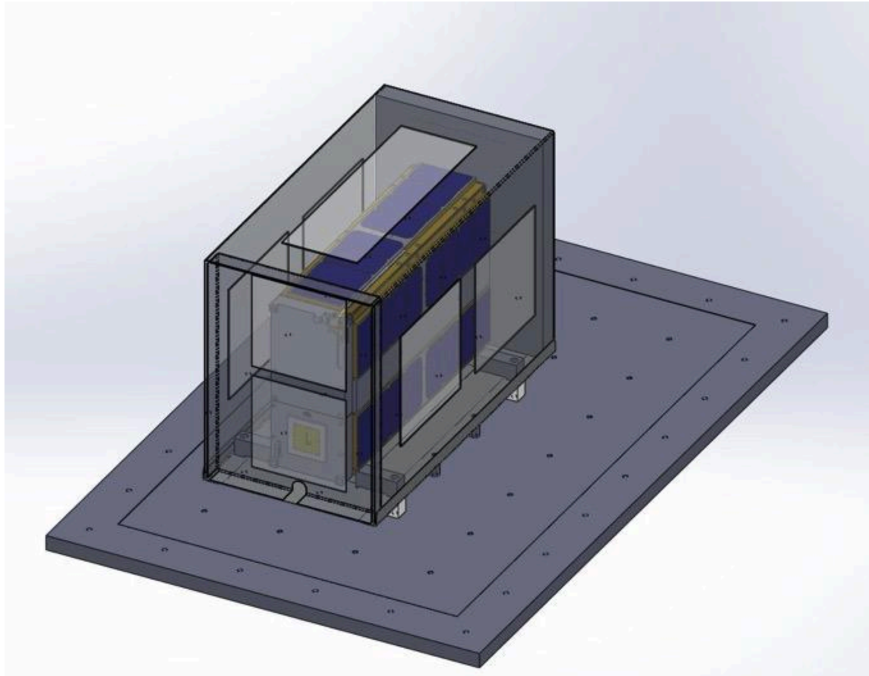


Figure 8 Test unit shown in thermal shroud

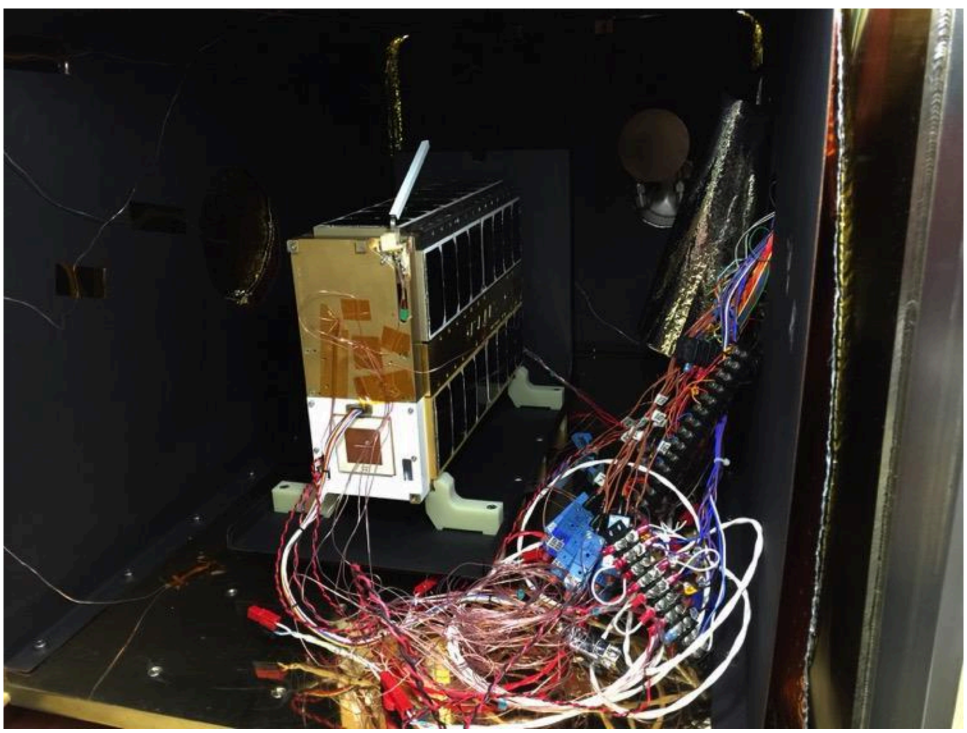


Figure 9 Flight unit loaded into test chamber

Test Operation
Two operators continuously staffed TVPM. This was done to ensure that there was never an instance when the spacecraft was left unattended, ensuring any anomalies could be safely mitigated in real time. A combined 600 work-hours were dedicated to supporting this test effort.

All test operators participated in a brief facility safety discussion, which included instructions on what to do in the event of a test anomaly, procedures for safing the spacecraft, and details about mitigating hazardous situations (such as a low oxygen alarm).

Test Events
Once setup in the chamber, the experiment was armed at 11:13 am on January 30th, 2015. The spacecraft was then powered down, so that the chamber could be evacuated to a pressure below 10^{-4} Torr. Once this pressure had been reached, the spacecraft was powered on (deployment) at 2:42pm on January 30th, 2015 by toggling the spacecraft's kill switch. The solar panel temperatures were stable at 21.5° C and started to climb at a rate of 0.05 to 0.06 degrees/minute as soon as the spacecraft power was enabled. The batteries were fully charged at deployment and the workmanship cold case (-29° C) temperature descent was started at 3:06pm. The base plate was set to -120° C and the shroud heater voltage to 70V.

At 10:10pm on February 3rd, the spacecraft automatically entered Phase B1, effectively initiating the experiment protocol. In this scenario, the experiment waited 96 hours before executing.

On the morning of February 8th, the experiment portion of the test concluded at 10:41am, and the team began to run a series of overnight thermal cycles. The heater was set to 85 V for 60 minutes followed by the heater being turned off for 30 minutes, simulating the thermal cycling of an orbit. During these tests the 8 hours of payload pumping power consumption was simulated by opening three valves. In addition, the power supply setting was adjusted from 10V at 1.35A to 9.5V at 1.42 A.

Prior to the end of the test, at 8:10am on February 12th, the chamber temperature set point was adjusted to 20 C. When the chamber reached room temperature, the spacecraft was powered down at 11:12am, and the vacuum pump was disabled.

Test Data
Below are the unprocessed charts from the TVPM master spreadsheet, showing the metrics recorded throughout the test. These charts serve as examples of as-run data recorded during the TVPM test process.

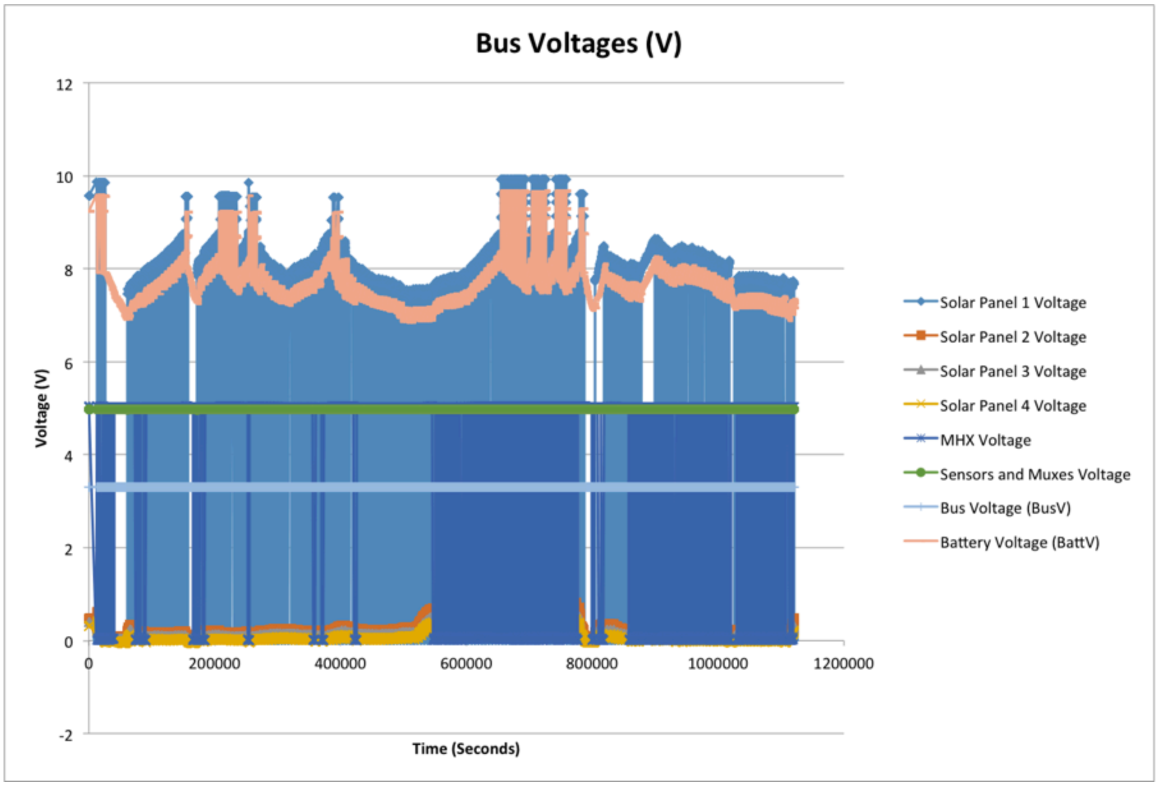


Figure 10 Bus Data: Voltages for Solar Panels, MicroHard Radio(MHX), Sensors

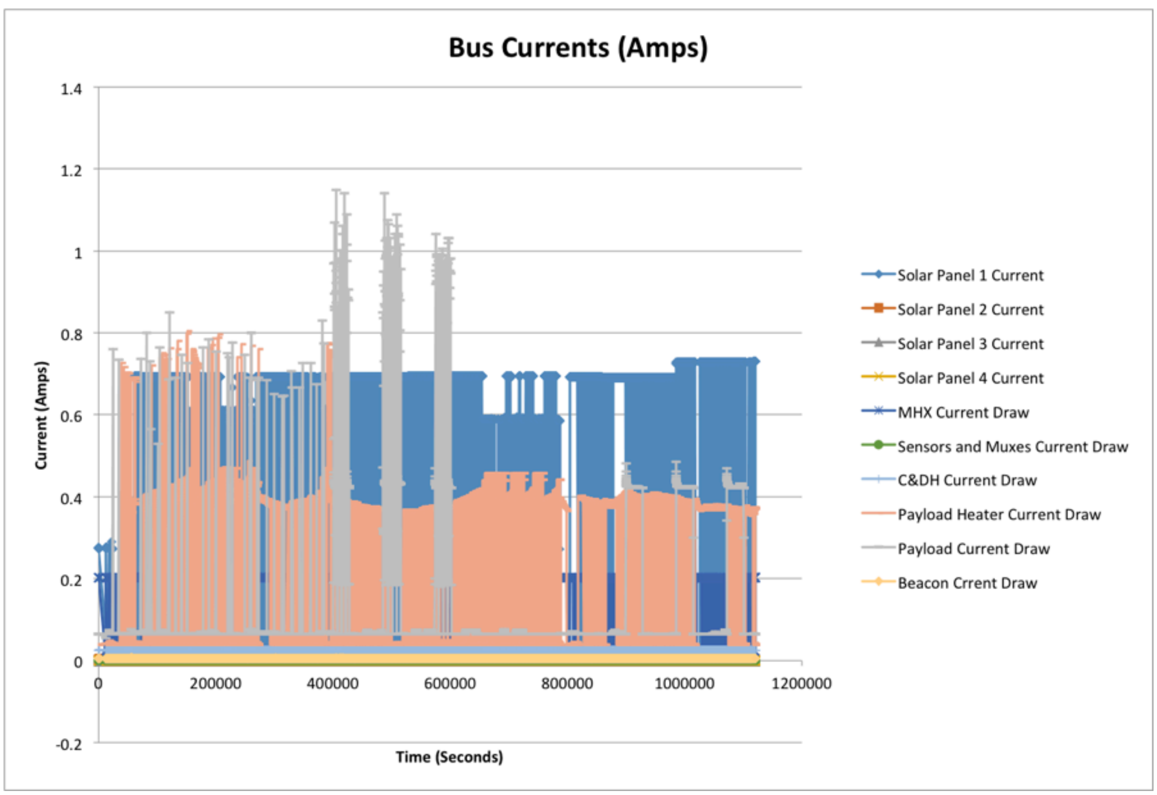


Figure 11 Bus Data: Current

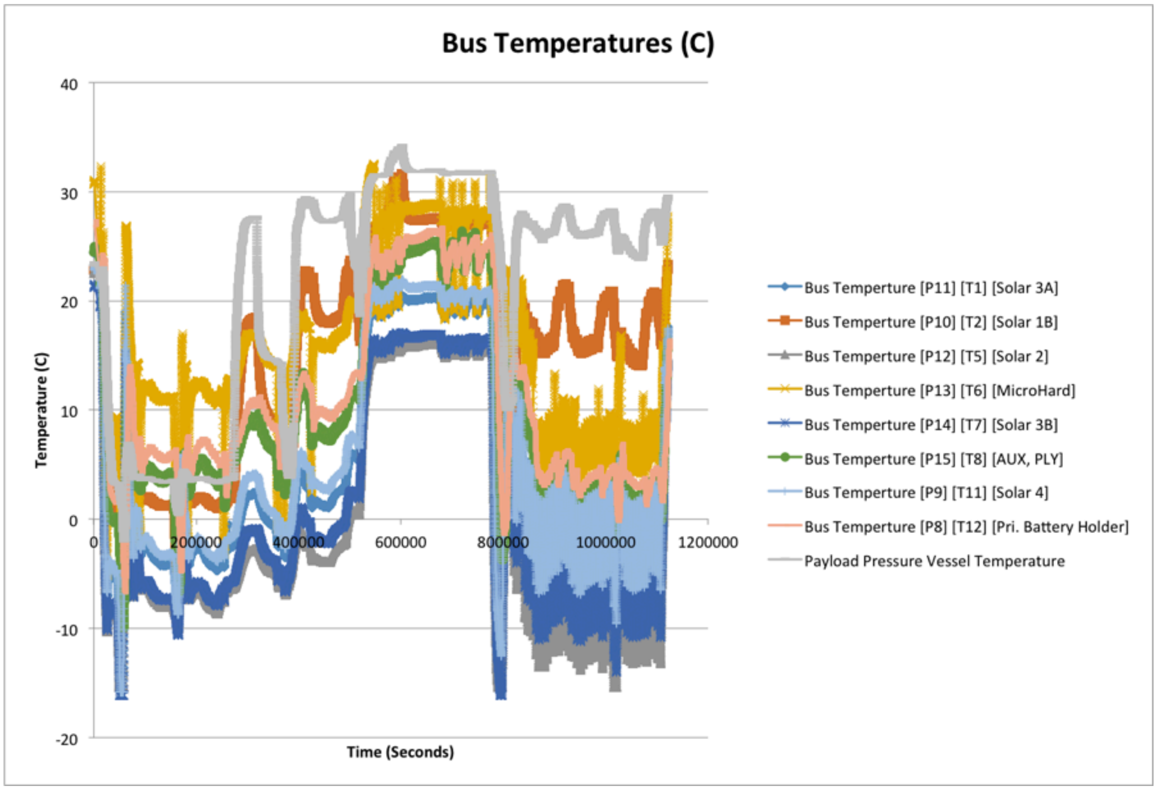


Figure 12 Bus Data: Temperatures

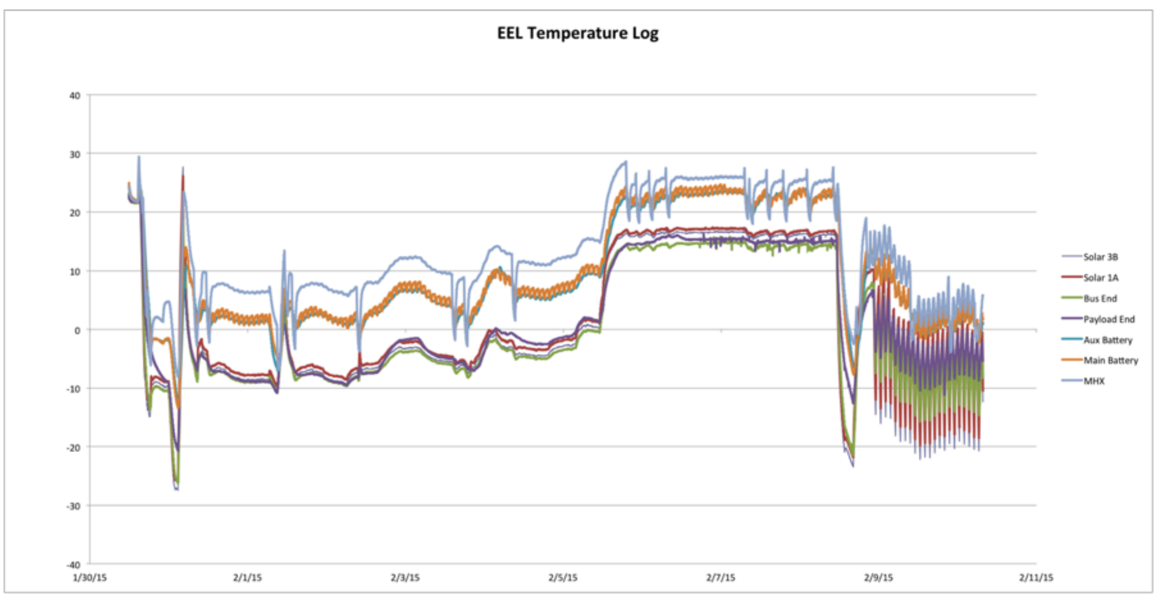


Figure 13 EEL Temperature Log

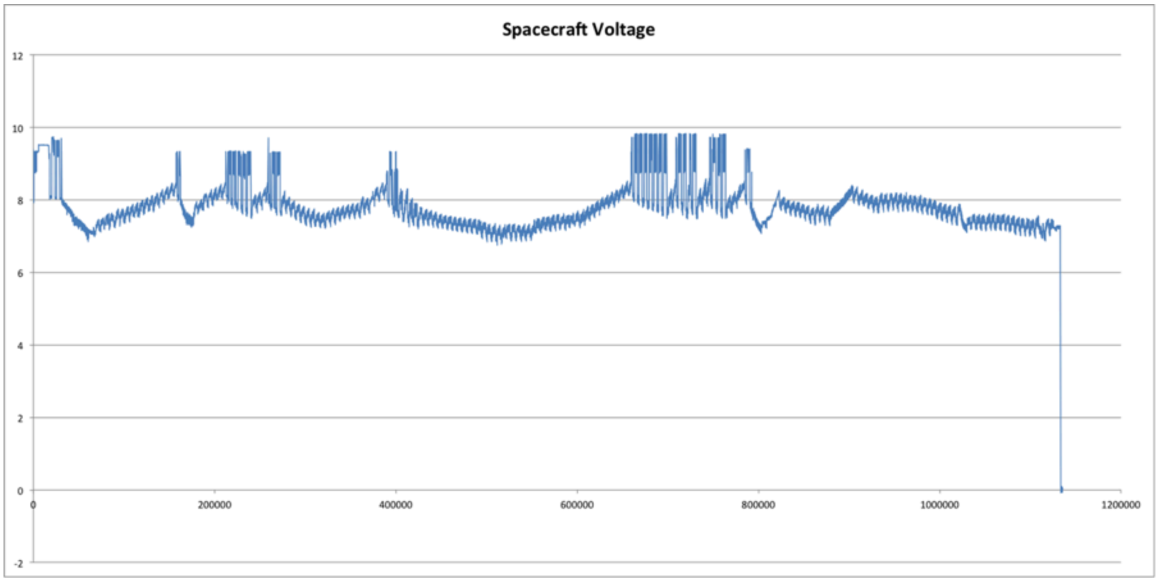


Figure 14 Full Spacecraft Voltage

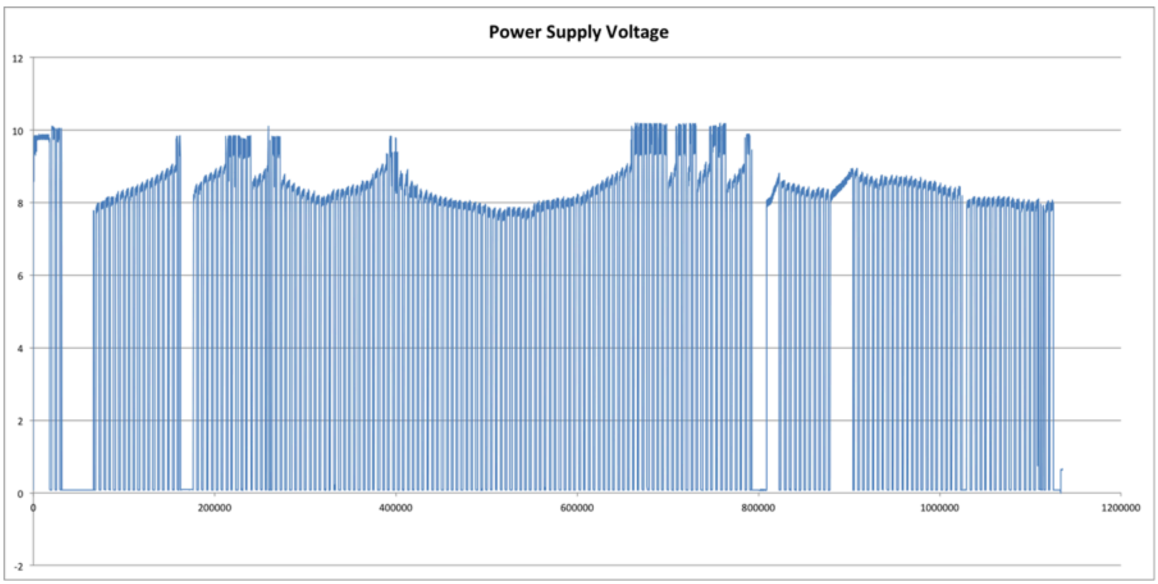


Figure 15 Power Supply Voltage

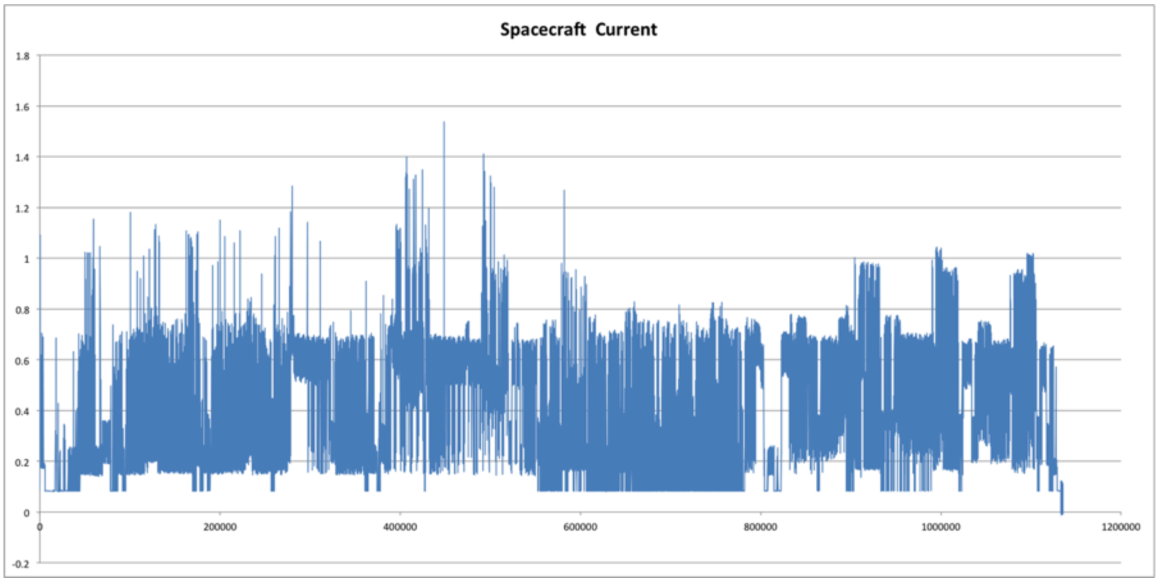


Figure 16 Full Spacecraft Current

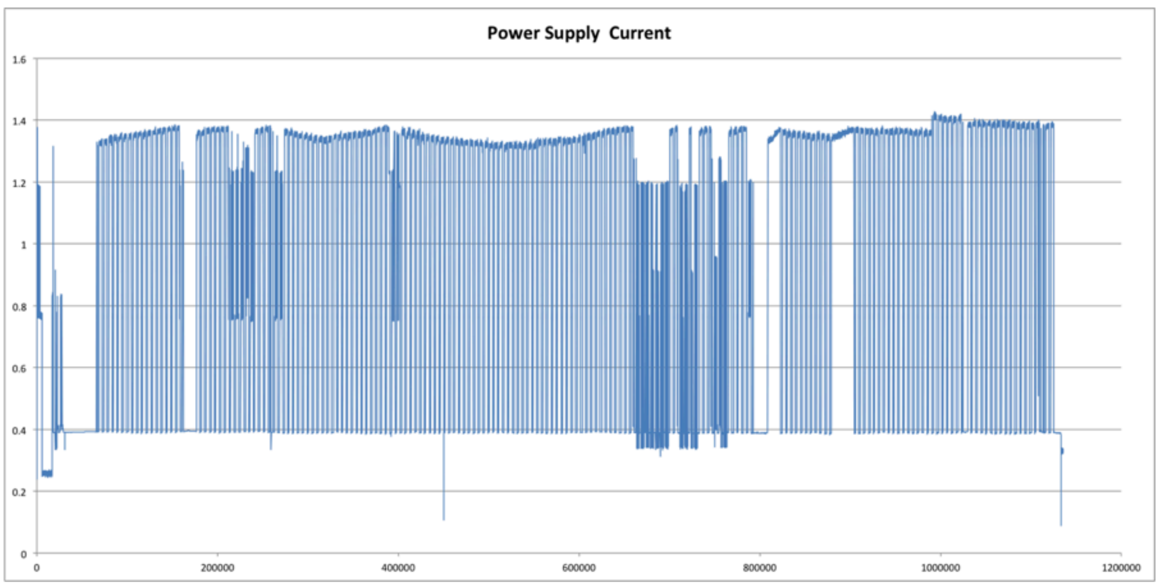


Figure 17 Power Supply Current

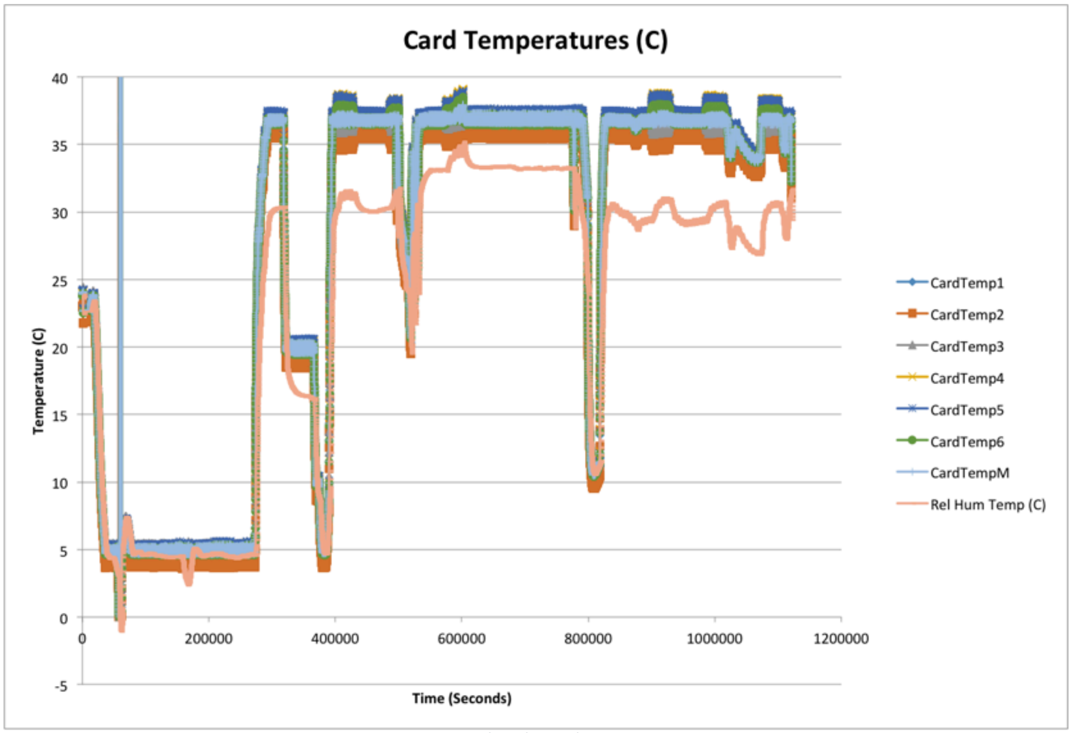


Figure 18 Payload Card Temperatures

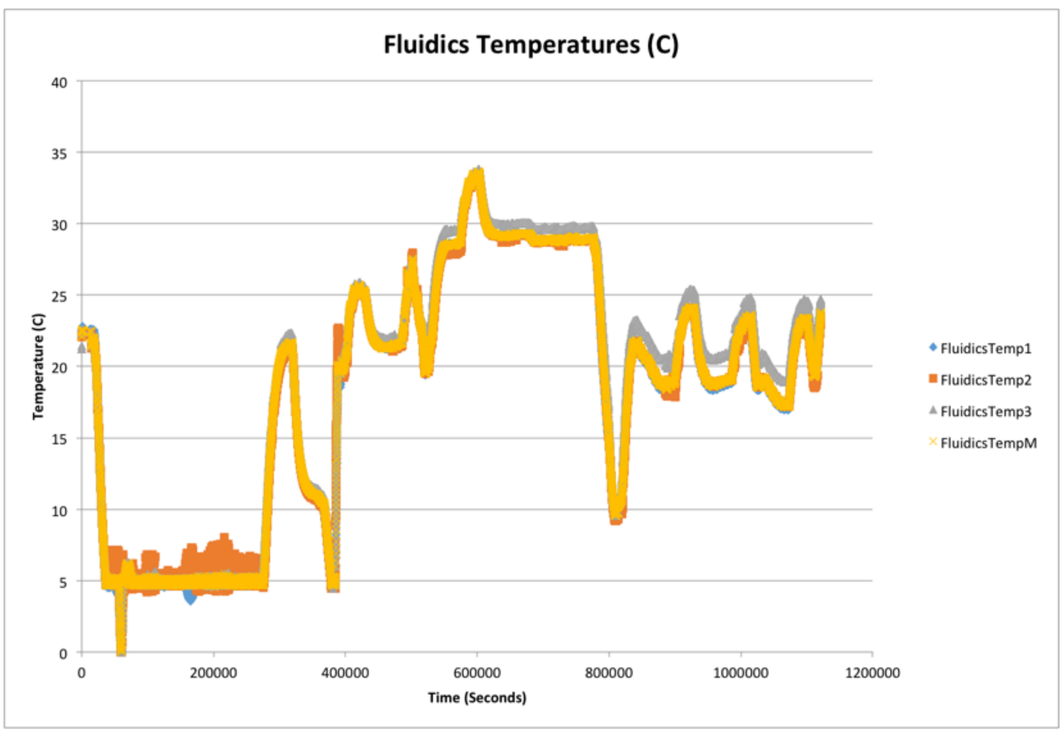


Figure 19 Payload Fluidic Temperatures

Model Results

The thermal model was run with updated improvements in geometry and dissipations. The results of the correlation are summarized on Table 5. In Figure 20, the deviation between predict and actual temperatures is shown. The average temperature prediction deviation is +1.2°C with a standard deviation of +/-3.4°C. It is desired to have predicts be within 5°C of actual temperatures. For this correlation, only three temperatures were outside the 5°C tolerance. They were all solar panel predicts, and they occurred on the hot case.

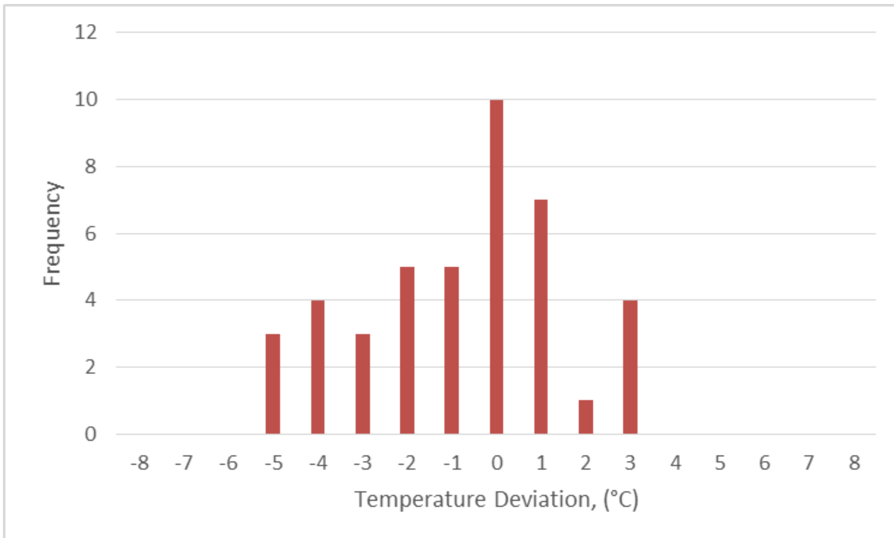


Figure 20 Prediction Deviation

Sheet:	Cold	Hot	Cold	Hot	Cold	Hot
Parameter:	Avg	Avg	Actual	Actual	T(P-A)	T(P-A)
Node/Group	*[°C]*	*[°C]*	*[°C]*	*[°C]*	*[°C]*	*[°C]*
Solar Panel						
Solar 1A	-8.6	14.1	-8	17.1	-0.6	-3.0
Solar 1B	-6.6	15.9				
Solar 2	-9.0	13.6	-12	13.3	3.0	0.3
Solar 3A	-8.8	13.9				
Solar 3B	-6.6	16.0	-9	16.5	2.4	-0.5
Solar 4	-6.5	16.1	-5	21.1	-1.5	-5.0
Solar - P11	-6.6	15.9	-3	20.1	-3.6	-4.2
Solar - P9	-6.5	16.1	-3	21.2	-3.5	-5.1
Solar - P12	-9.0	13.6	-8	15.9	-1.0	-2.3
Solar - P14	-6.6	16.0	-7	16.7	0.4	-0.7
Payload						
Payload Can Top	-2.8	21.7	-3	22	0.2	-0.3
Battery Cells						
Pri Cell	-0.1	21.8	2	26	-2.1	-4.2
Aux Cell	0.6	22.6	3	25	-2.4	-2.4
Bus PCB						
CD&H	-4.3	17.7				
EPS	6.2	28.3				
Backplane	-2.4	19.6				
MicroHard	10.9	27.5	11	28	-0.1	-0.5
Chamber and Heater Enclosure						
Chamber - Top	-91.2	-79.3	-86	-74	-5.2	-5.3
Enclosure - Top	-21.0	3.9	-20	5	-1.0	-1.1
Enclosure - Bot	-5.1	21.7	-7	22	1.9	-0.3
Enclosure - Side 1	-16.2	10.5	-12	12	-4.2	-1.5
Enclosure - Side 2	-15.5	11.5	-15	15	-0.5	-3.5

Table 5 Correlation Results

TVPM Conclusions

The most substantial impact to model prediction was the revision of power dissipations, discussed above. Of key importance, the battery pack temperature is vital to the spacecraft power management. The power per cell (each pack has four cells, and the spacecraft has two packs) was updated from 1.36W to .067W. A tighter agreement between predicted and actual temperatures is desired. Unfortunately, this was not a thermal test, per se. The sparse temperature instrumentation was enough to make an assessment of the spacecraft thermal performance, but it was not adequate to truly calibrate the thermal model. Still, it was enough to instill confidence in the model and to how the spacecraft will operate in flight.

Lessons Learned from TVPM

Drawing from lessons learned in this test, the team felt compelled to recommend that future Ames projects use thermal cycling as opposed to cold/hot soak dwell periods to evaluate the power budget of a spacecraft for a high-altitude mission. It is likely that since previous missions haven't been as cold as EcAMSat is predicted to be, the discrepancy went unnoticed, but the power consumption of a spacecraft in a cold soak at the average temperature is greater than the actual power consumption due to thermal cycling. Cold/Hot soaks may give inaccurate power consumption metrics.

Shock

Externally induced shock was used to perform shock tests on the EcAMSat test article in each of the three orthogonal axes. For this test, GEVS Protoflight Qualification levels were selected. The SRS(Shock Response Spectrum) requirement is 1.4 X Limit Level with the test performed once in each orthogonal axis.

Test Plot Observations

• Shock input levels were reasonably consistent from test to test.
• No anomalies were noted in the X, Y, and Z axis outputs.
• The test conditions achievable on the EEL facility's IMPAC 2424 shock machine fell short of target test parameters as shown in Figure 5. The actual level was 50% of the target value at 650 Hz and dropped to 25% at 8 kHz

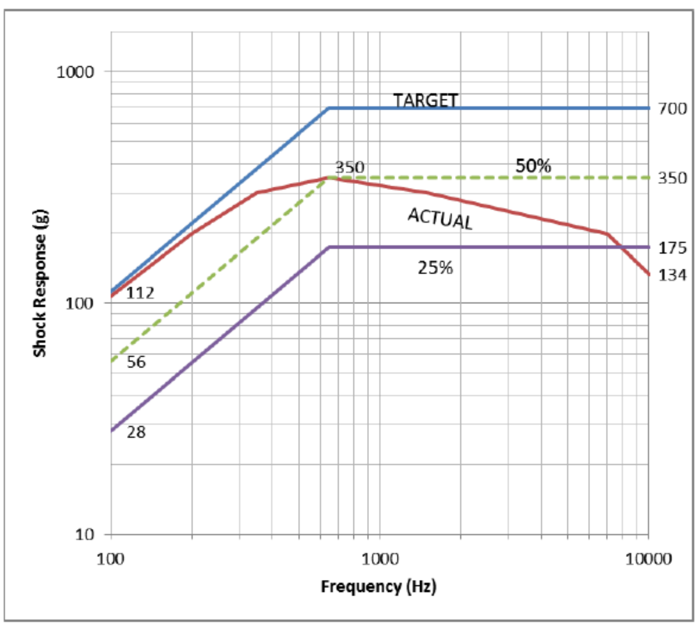


Figure 21 Target and Actual SRS

• The test article passed functional and mechanical inspection tests prior to shock testing.
• No anomalies were noted during shock testing.
• The test article passed functional and mechanical inspection tests following shock testing.

Vibe
GEVS Protoflight Qualification levels were used to test for random vibration levels on each of the three orthogonal axes.

Vibration Test Plot Observations:
- Shaker input levels stayed within the set warning limits at all frequencies for all test runs.
- Input GRMS = 14.1 for all random vibration test runs.
- No anomalies were noted in the X, Y, and Z axis outputs.
- No anomalies were noted between pre-test and post-test sine sweep plots for each axis.

Level	0.25 G_{RMS}
Frequency	20-2000 Hz
Sweep Rate	4.0 oct/min
Duration	2.0 minutes

Table 6 Sine Sweep Parameters

• The test article passed functional and mechanical inspection tests prior to random vibration testing.
• No anomalies were noted during vibration testing. All random vibration test parameters were within prescribed limits.
• The test article passed functional and mechanical inspection tests following random vibration testing

RF Communication Testing

In January of 2015, EcAMSat underwent two communication system tests with the support of the NASA Ames RF Test Lab (RTL). This included an antenna frequency tuning test and an RF Link-Margin Test. The antenna tuning was performed in the lab utilizing the low loss cables and a Network Analyzer. The RF Link Test was performed with the spacecraft located at NASA Ames and the waypoint measurements taken in standard atmospheric conditions at a 15-mile distance using project supplied antennas and RTL low loss cables, adapters and Spectrum Analyzer. The 15-mile location was chosen because of its line of site relative to Moffett Field and to generate enough distance from the spacecraft in order to simulate environmental space-to-ground attenuation. These tests met the specified technical requirements to meet mission acceptance.

In October of 2017, EcAMSat returned to the RTL for a second round of RF Link Testing, but this time conducted within the labs RFI/EMI Shielded Enclosure to null out any environmental effects. This test was conducted to confirm the spacecraft comm system was correctly electrically configured prior to launch due to the disassembly and reassembly required to load biology. The chamber was designed to provide attenuation for MIL-STD-461 with performance levels of 56dB at 1KHz and 100dB from 200KHz to 10GHz. This semi-anechoic chamber was also equipped with absorber material that completely shrouds the interior walls and ceiling adding an additional
-15dB to -50dB of attenuation from 450MHz to 40GHz. This further guarantees complete environmental isolation for the highest degree of precision and accuracy, the chamber is completely isolated from facility with the use of a 30A power filter and an independent chamber dedicated ground rod.

During the 2017 test, measurements were conducted using the labs Omni Antenna specifically designed for receive frequencies from 25MHz to 6GHz. Calibration of the test configuration and measurements were conducted with the labs Network Analyzer + Spectrum Analyzer. In order to improve the accuracy of the measurements and to identify spurious signals, measurements of the spacecraft's transmissions were captured in short bands of 412.1MHz – 462.1MHz, 2.4GHz – 2.45GHz and 2.4GHz – 2.45GHz. Measurements at small bands around the known transmit frequencies allowed to better resolve the signals of interest at a much quicker measurement speed. This improved the overall test campaign allowing for more time to focus on signal analysis of the spacecrafts transmit frequencies, identify spurious signals and take extra measurements where necessary.

Campaign Conclusion

The EcAMSat environmental test campaign was deemed a success by center management. All test requirements were completed and key TPM's were achieved. EcAMSat proved that a tailored approach to NPR 7120.5 could be successful while also achieving a factor of cost savings and gaining NASA decadal survey science. The test campaign was instrumental in achieving full mission success.