# AlbertaSat

## Ex-Alta 2

# Hyperion Detailed Design Document

**Version 0.12**

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University of Alberta

Institute for Space Science Exploration and Technology

# Document Authentication

**Prepared by**

| Name | Position | Signature | Date |
|---|---|---|---|
| Erik Halliwell | Power Lead | | |
| Nick Sorensen | Power Lead | | April 21, 2020 |

**Reviewed by**

| Name | Position | Signature | Date |
|---|---|---|---|
| | | | |
| | | | |

The undersigned acknowledge that they have reviewed the project management plan, and they authorize the objectives, rules, and organization of the project as described in this document. Any changes to this project management plan will be coordinated with and approved by the undersigned or their designated representatives.

**Approved by**

| Name | Position | Signature | Date |
|---|---|---|---|
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# Record of Document Changes

| Version | Effective Date | Author(s) | Section(s) Changed | Description of Change(s) |
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| 0.10 | 2019/05/21 | E. Halliwell | all | creation |
| 0.11 | 2020/04/21 | N. Sorensen | all | added tonnes |
| 0.12 | 2020/06/27 | N. Sorensen | all | Added more information pertaining to the schematic designs of Hyperion. |

# Executive Summary

This document serves as a detailed design description, design review and change document for the Hyperion Solar Panel component project. For interfacing information between Hyperion and other boards, please see the Hyperion Interface Control Document (ICD) [R-01], as well as the ICDs of other subsystems within the Ex Alta 2 project.

Hyperion is the alternative name of the solar panels which will be flown on Ex Alta 2, AlbertaSat's second satellite, and it is required that a wing deployment is used for Ex Alta 2, in which two panels are deployed from the Port and Starboard sides. This document will aid in the development of the Hyperion solar panel boards by providing a location for planned tasks, completed tasks, in-progress/finalized designs, gathered information regarding components, tracking of reviews, and all other relevant pieces of information for the development of the solar panels. Note that all dates refer to the date of the review unless otherwise specified. This document also serves as a detailed design overview of the Hyperion Component.

Hyperion is an in-house set of solar panels that will be assembled by AlbertaSat using commercial off the shelf solar cells, and circuitry designed in house. The large power requirements of Ex-Alta 2 motivate Hyperion's design to have a "spacewing" configuration; Hyperion features three body mounted solar panels on the zenith, port, and starboard faces, with two extra solar panels extending out from the starboard and port sides to sit level with the zenith face of the cubesat. The cells on the body-mounted faces each have three parallel strings of two solar cells, while the deployable solar panels have three full parallel strings of two cells.

Current versions of the boards can be found in the AlbertaSat GitHub [R-05].

The various requirements and recommendations governing the design of Hyperion motivate the inclusion of various electronic components: all solar panels will contain 3 Texas Instruments LMT70 temperature sensors capable of sensing temperatures from -55°C to 150°C; each panel also has 3 Silonex photodiodes acting as sun sensors, capable of sensing light in the visible to infrared light range, and a dual-amplifier chip which acts as both a voltage and current sensor; additionally, each panel contains rectifier diodes for back current protection.

# Table of Contents

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# List of Tables

# 1 References

| Reference | Document Name | Description |
|---|---|---|
| R-01 | Hyperion Interface Control Document | This document outlines the different interfaces between Hyperion and other boards and components |
| R-02 | Hyperion Design Tracker | This document outlines the requirements and how said requirements are achieved on the Solar Panels. |
| R-03 | Solar Cell Trade Study | This is a trade study of potential solar cells. |
| R-04 | Systems ICD | Shows location planning for the temperature sensors. |
| R-05 | GitHub Repository | Published versions of all project documents. |
| R-06 | Solar Cells | Folder of associated solar cell documents. |
| R-07 | Ex-Alta 2 Power Budget | Power budget for Ex-Alta 2. |
| R-08 | Hyperion Testing Plan | Component Testing Plan for Hyperion. |
| R-09 | Failure Analysis | Solar Panel Deployment Failure Analysis |

Table 1: Reference table.

# 2 Introduction

The Hyperion project consists of  three body mounted solar panels on the Zenith, Port, and Starboard faces, with two extra solar panels extending out from the Starboard and Port sides to sit level with the Zenith face of the CubeSat. Each panel is designed to house six solar cells in a 6S1P orientation, nominally providing ~0.48 A at ~14.4 V at max power point. There are a variety of auxiliary electronic components on each panel, described further in this document.

# 3 Hardware Design

## 3.1 Requirements

Below are the requirements from SX2-SE-003 System Requirements Compliancy Table [R07]. For the system's compliance to these requirements, see the Requirements Compliance Section and Table X.

| Requirement Number | Design Specification/Requirement | Source/Justification |
|---|---|---|
| **Functional** | | |
| **System** | | |
| ASX2-SYS-7.050 | No pyrotechnics shall be on the satellite unless they are pre-approved by NanoRacks. Melt wire systems for deployables are permitted. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: Section 4.4.5 |
| ASX2-SYS-7.121 | All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris will be created. The CubeSat shall not have detachable parts during launch or normal mission operations. Any exceptions will be coordinated with NanoRacks and documented in the unique payload ICA. | CalPoly CubeSat Standard wording<br>Nanoracks Interface Definition Document: NR-NRCSD-S0003: 4.4.6-1<br>Nanoracks Interface Definition Document: NR-NRCSD-S0003: 4.4.1 |
| ASX2-SYS-7.140 | Vented fasteners shall be used for blind holes. | Cannot have trapped air. |
| ASX2-SYS-7.160 | All spacecraft components shall be electrically bonded per SSP 30245 to ensure the spacecraft is free from electrical shock and static discharge hazards. Typically, spacecraft components may be bonded by either nickel plating or chemical film treated faying surfaces or dedicated bonding straps. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: Req 4.4.11 |
| ASX2-SYS-7.210 | Stress corrosion resistant materials from Table I of MSFC-SPEC-522 are preferred. Any use of stress corrosion susceptible materials (Table II) shall be coordinated with NanoRacks and documented in the ICA. Any use of Table III materials shall be avoided. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: Req 4.4.10.1 |
| ASX2-SYS-7.220 | All components shall be marked with axis orientations where appropriate | Ensure components are properly installed. |
| ASX2-SYS-7.230 | All components shall be marked with hardware and firmware versions where appropriate. | Ensure correct component versions are used. |
| ASX2-SYS-7.240 | All components shall be designed to avoid floating voltages. | Prevents build up of charge. |
| **Electrical Power** | | |
| ASX2-SYS-3.010 | The EPS shall provide sufficient power at appropriate voltages to the SC subsystems as required by the SC operational modes. | The components of the satellite require power in order to function and it is the job of the EPS to accomplish this. |

| | | |
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| ASX2-SYS-3.040 | The power system shall not create magnetic fields that will cause destabilization of the SC. | Electrical current in wires creates magnetic fields, which can cause rotation of the satellite. This is undesirable and may not be compensated for by the ADCS, thus it must be avoided. |
| ASX2-SYS-3.070 | The CubeSat shall have a RBF or ABF feature that keeps the satellite in an unpowered state throughout handling and integration into the NanoRacks deployment pod. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: Req 4.2.1-5 |
| ASX2-SYS-3.080 | The RBF/ABF feature shall preclude any power from any source from operating any satellite functions except for pre-integration battery charging. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: Req 4.2.1-6 |
| ASX2-SYS-3.210 | All electrical power storage devices shall be internal to the CubeSat. | Nanoracks Interface Definition document, NR-NRCSD-S0003: 4.2.1-1 |
| **Mechanical** | | |
| ASX2-SYS-4.045 | The CubeSat rails shall be the only mechanical interface to the NRCSD in all axes (X, Y and Z axes). This means nothing can overlap the minimum width of the rails. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: 4.1.1-10 |
| ASX2-SYS-4.260 | The DFGM shall be mounted to the exterior of the satellite. | Must be as far away as feasible from magnetic fields created by the satellite. |
| ASX2-SYS-4.390 | The CubeSat shall be capable of being integrated forwards and backwards inside of the NRCSD (the +/-Z face could be deployed without issue). Note: In general, the deployables should be hinged towards the front of the deployer to mitigate risk of a hang-fire should the deployables be released prematurely while the CubeSat is still inside the NRCSD. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: 4.1.5-2 |
| **Performance** | | |
| **System** | | |
| ASX2-SYS-7.010 | The CubeSat shall survive within the temperature range of −40◦C to +65◦C from the time of launch until its end of life. | These are the temperatures the the Nanoracks Interface Control document: NR-SRD-029-7 recommends |

| | | |
|---|---|---|
| ASX2-SYS-7.020 | The CubeSat shall not use any material that has the potential to degrade in an ambient environment during storage after assembly for a length of 2 years. | The SC may have to sit in storage depending on the time from completion/delivery to launch. Degradation of SC due to storage time is not desirable. From QB50 Systems Requirements. |
| ASX2-SYS-7.060 | Satellites shall comply with NASA guidelines for selecting all non-metallic materials based on available outgassing data. Non-metallic materials shall have a Total Mass Loss (TML) less than or equal to 1.0% and Collected Volatile Condensable Material (CVCM) value less than or equal to 0.1%. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: Req 4.4.10.3 |
| ASX2-SYS-7.110 | The CubeSat shall be capable of withstanding pressure extremes from 0 to 104.8 kPa and depressurization/pressurization rate 1.0 kPa/second. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: Req 4.3.8 |
| ASX2-SYS-7.170 | The CubeSat shall be capable of withstanding the relative humidity environment for all mission phases leading up to deployment, which is between 25% to 75% relative humidity (RH) for ascent and on-orbit phases of flight. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: Req 4.3.7 |
| ASX2-SYS-7.250 | All SC systems shall operate in a vacuum of 10^-10 Torr. | The pressure experienced by a spacecraft in LEO is between 10^-10 Torr and 5 x 10^-8 Torr. Ref: LEO Spacecraft charging NASA handbook |
| **Electrical Power** | | |
| ASX2-SYS-3.020 | The SC must be able to supply at least 1.5 Wh during a worst case scenario orbit. TBC upon final component selection. | Power critical mode requires at least 1.8 Wh to ensure a positive power margin, but this number should go down with further analysis of operation of the board. TX2-PW-346-Ver 10.03 Ex-Alta 2 Power Budget: https://docs.google.com/spreadsheets/d/1vJ5BbYvJwwVbvuQ7yAY-GjK6VjirMhG3TSkO1tw3fBc/edit?usp=sharing |
| **Mechanical** | | |

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|---|---|---|
| ASX2-SYS-4.040 | The CubeSat rails and envelope shall adhere to the dimensional specification outlined in NR-NRCSD-S0003 Figure 4.1.1-1. | Nanoracks Interface Definition Document NR-NRCSD-S0003: 4.1.1-2 |
| ASX2-SYS-4.160 | The CubeSat shall have an approved secondary locking feature for any and all fasteners or subcomponents external to the CubeSat chassis that would not be held captive by the spacecraft structure should it come loose. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: 4.4.3 |
| ASX2-SYS-4.250 | The imager shall have a minimum clear half-angle FOV of 5 degrees surrounding the aperture. TBC | The imager requires a clear FOV in order to gather scientific data. |
| ASX2-SYS-4.270 | The exterior of the satellite shall accommodate the DFGM sensor head when stowed. The sensor head has the dimensions 38x32x28 mm. | The DFGM must be able to be stowed to fit within the required volume envelope. |
| ASX2-SYS-4.290 | All RBF/ABF and charging systems for the integration process must be located on one side of the satellite (+Y in the NRCSD frame) and in the access port locations shown in NR-NRCSD-S0003, Figure 3.3-1. | Nanoracks Interface Definition document: NR-NRCSD-S0003: 4.1.3-1 |
| ASX2-SYS-4.350 | The DFGM wiring bundles that come out of the boom shall connect directly to the DFGM PC104 board. | The DFGM on the end of the boom must be able to receive power from and exchange data with the DFGM PC104. |
| ASX2-SYS-4.380 | CubeSat deployable systems (solar arrays, antennas, payload booms, etc.) shall have independent restraint mechanisms that do not rely on the NRCSD dispenser. | Nanoracks Interface Definition Document: NR-NRCSD-S0003: 4.1.5-1 |
| **Spacecraft tests** | | |
| **General** | | |
| ASX2-TST-1.010 | Internal connectors shall not be disconnected after the completion of the spacecraft acceptance test program. | Final flight hardware configuration must be tested in the acceptance test program and no changes made after |
| ASX2-TST-1.020 | All remove before flight items shall be easily accessible and removable without any disassembly of the SC. | Disassembly is not allowed after acceptance testing. |
| ASX2-TST-1.040 | The SC shall incorporate a connector easily accessible to allow battery charging and basic functional verification of the SC during ground testing and pre-launch operations. | Functional tests require access to SC software after integration. |

| | | |
|---|---|---|
| ASX2-TST-1.050 | Each subsystem shall be tested independently prior to unit assembly. | Allows for easier correction of subsystem errors. |
| Mechanical | | |
| ASX2-TST-2.070 | The SC shall include handling points to permit lifting. | We will need to lift and move the SC during integration and test, so having designated points which are the safest to hold it will ensure minimal impact on SC operation and lifetime. |
| ASX2-TST-2.080 | The SC components shall fit inside or on the structure. | Everything must be attached. |
| ASX2-TST-2.090 | The SC shall fit inside the NRCSD without any contact except the outer rails. | NanoRacks requires this. |
| ASX2-TST-2.100 | Deployment mechanisms shall be verified to work by deployment tests. | Need to ensure the deployment mechanisms will work. There were some issues on Ex-Alta 1 with them getting caught. |
| ASX2-TST-2.140 | Every element shall be designed to have a positive Safety Margin. The Safety Margin is defined as: (Max allowable load or stress)/(Predicted design load or stress)-1 | Minimum required to be designed to not break. |
| Electrical | | |
| ASX2-TST-3.010 | All solder joints must be inspected after component assembly and before integration into larger assemblies. | CSA assembly technician stressed the important to ensure that solder joints have a good connection that will not fail during testing or mission. |

Table 2: Requirement table.

## 3.2 Recommendations

Below are the requirements from SX2-SE-003 System Requirements Compliancy Table [R07]. For the system's compliance to these recommendations, see the Recommendations Compliance Section and Table X.

| Requirement Number | Design Specification/Requirement | Source/Justification |
|---|---|---|
| Functional | | |
| System | | |

| | | |
|---|---|---|
| Recommendation 4.110 | All electronic assemblies and electronic circuit boards should be conformally coated. | Protection from ground contamination, debris or particles, and the formation of metal whiskers. From QB50 Systems Requirements. |
| Recommendation 4.120 | Magnetizable material should not be used for CubeSat parts. | This is because magnetizable materials may impact ADCS performance. From QB50 requirements |
| Recommendation 4.140 | All soldered components should have the ability to be manually reworked. | Manual rework provides the ability to fix bad solder joints and other solder errors. |
| **Performance** | | |
| **System** | | |
| Recommendation 4.115 | All components used on in-house designs should have an operational temperature rating of at least -40◦C to +105◦C. | |

Table 3: Recommendations table.

## 3.3 Design Choices and Component Selection

### 3.3.1 Solar cell trade study

Please refer to the Solar cell trade study folder and any associated documents for more information on possible solar cells and their implementation on Hyperion. In summary, Spectrolab's XTJ-Prime Solar Cells have been selected as Hyperion's power generation component. Tables 4 and 5 summarize the electrical and mechanical characteristics of the Spectrolab XTJ-Prime cells.

| Model | Thickness [um] | Width [cm] | Length [cm] | Area [cm$^2$] | Mass [g] |
|---|---|---|---|---|---|
| **Spectrolab XTJ-Prime** | 225 | 6.91 | 3.97 | 26.62* | 2.24 |

Table 4: Mechanical characteristics of the XTJ-Prime solar cell.

| Category | Parameter | Condition | Typical Value |
|---|---|---|---|
| **Power Generation** | **$P_{MP}$ (W)** | Max Power Generation | 1.18 |
| | **$I_{SC}$ (A)** | Short-Circuit Current | 0.51 |

| | | | |
|---|---|---|---|
| | **$V_{OC}$ (V)** | Open-Circuit Voltage | 2.75 |
| **Temperature Effects** | **$V_{MP}/\Delta T$ (mV/°C)** | Change in Max-Power Voltage per Degree Celsius | -6.2 |
| | **$I_{MP}/\Delta T$ (uA/°C)** | Change in Max-Power Current per Degree Celsius | 217.76 |
| | **$I_{SC}/\Delta T$ (uA/°C)** | Change in Short-Circuit Current per Degree Celsius | 299.42 |
| | **$V_{OC}/\Delta T$ (mV/°C)** | Change in Open-Circuit Voltage per Degree Celsius | -5.6 |

Table 5: Electrical Characteristics of the XTJ-Prime Cells.

These cells were majorly selected due to their favourable efficiencies, as well as their small size. The configuration of these cells on the satellite was designed so that they produce a worst case power orbit greater than the power required to run the satellite in Standby mode, and preferentially greater than the power required for the satellite to function in the defined science modes. These powers are given in Table 5; we see that we must produce over 3.78 W in a worst case, but would preferentially generate approximately over 5 W in a worst case orbit, and over 6 W in a nominal/best case orbit.

| Mode | Power Critical | Power Safe | Standby | Detumbling | DFGM Science | Imaging | Mixed Science |
|---|---|---|---|---|---|---|---|
| **Power (W)** | 0.16 | 1.75 | 3.78 | 4.76 | 6.02 | 5.68 | 6.14 |

Table 5: Table of power consumption over different modes.

We considered a wide range of solar panel configurations, and the design that resulted was that of a wing-deployment design (see Figure 1). There are 5 solar panels, each with 6 cells. As the satellite is earth-facing, there are 3 body mounted panels: one on Zenith, one on Port, and one on Starboard. There are two deployable panels, and when stowed one is mounted on the port side, and the other is mounted on the starboard side. When deployed, the panels sit parallel to Zenith. The configuration of the cells are shown in Figure 2, where each solar panel has 6 cells in a 6S1P format, producing nominally 14.4 V at 0.48 A at MPPT. A schematic visualization of this can be seen in Figure 3.

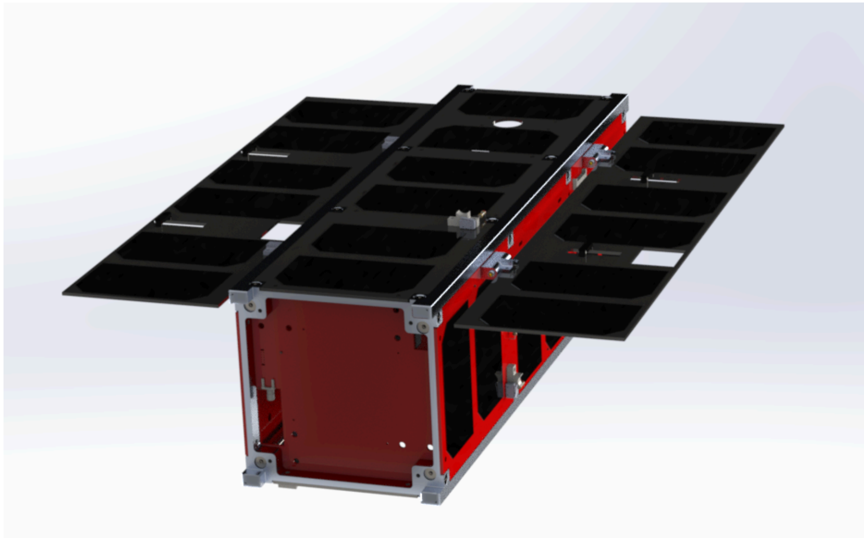


Figure 2: Render of solar panel wing-configuration for Ex-Alta 2.

This configuration of cells, assuming Nadir pointing, produces the power shown in Table 6. Deployment failure analysis has been performed on this design and it has been shown that power generation is adequate at all reasonably possible failure configurations [R-09]. Note that these values meet our requirements given the power consumptions specified earlier.

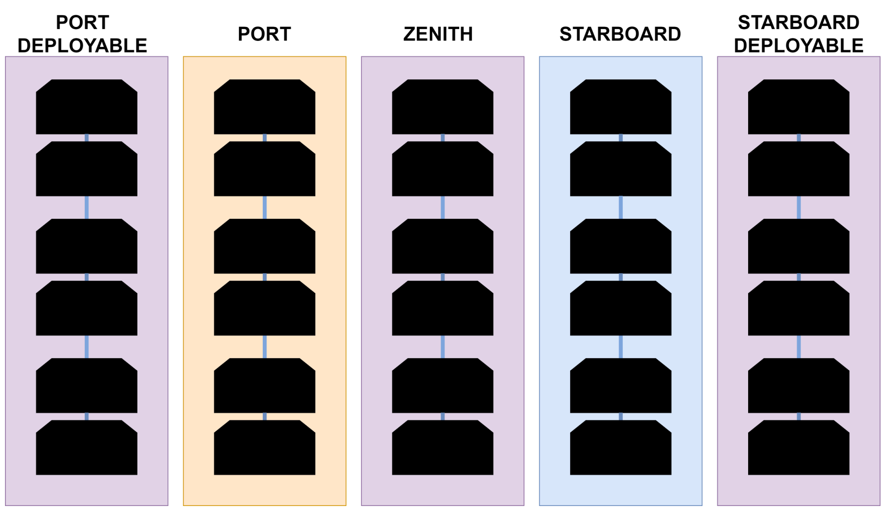



Figure 3: Configuration of solar cells on Ex-Alta 2. All cells are configured as 6S1P and each panel operates at MPP at ~14.4 V and ~0.48 A.

| Case | Worst Case | Nominal Case | Best Case |
|---|---|---|---|
| **Power Generation (W)** | 5.11 | 6.09 | 7.31 |

Table 6: Table of power generated over worst, nominal, and best case orbits.

| Category | Parameter | Condition | Min | Typical | Max |
|---|---|---|---|---|---|
| **Solar Panel Output** | $V_{MPPT}$ (V) | MPP Array Voltage * | - | 4.812 (Zenith)<br>9.624 (Other) | - |
| | $I_{MPPT}$ (A) | MPP Array Current Per Panel | - | 0.479 | - |
| | $P_{AVG}$ (W) | Orbital Average Power (Entire Array) | 3.40 | 3.91 | 5.19 |
| **Operation** | $P_{USE}$ (mW) | Array Idle Power Consumption | - | 60 | 75 |
| | $T_{OP}$ (°C) | Operating Temperature * | -40 | - | 125 |
| **Storage** | $T_{STO}$ (°C) | Storage Temperature | -40 | - | 125 |
| **Mass** | $M_{BM}$ (g) | Mass - Single Body Mounted Panel | 72 | 82 | 92 |
| | $M_{DEP}$ (g) | Mass - Single Deployable | 66 | 76 | 86 |

Table 8: Summary characteristics of Hyperion-2U. *The HW Set MPPT voltage on the EPS will be set to 0.3 V per cell less than at max power point.

### 3.3.2 Temperature Dependence

The efficiency and power generation capabilities of the solar cells are dependent upon the temperature of the solar array, and that is further dependent upon the satellite orbit. This is because different orbits will expose the satellite to different amounts of solar irradiance with differing duty cycles. Thermal analysis has determined that the solar cells would reach 70°C if exposed to irradiance equaling the solar constant (1.361 kW/m^2) at steady state. This forms the maximum temperature the solar array could experience, though the actual maximum temperature the solar array would experience is likely much lower than this. This is because the SC experiences eclipse for a significant portion of most orbits. All calculations and numbers cited in the rest of this document assume an operating temperature of 28°C, at which the XTJ-Prime operational characteristics are listed [R-25]. This section demonstrates how temperature could affect the power generation of the satellite over the determined range of temperatures.

The XTJ-Prime solar cells list the temperature coefficients as follows:

| Parameter | Condition | BOL | EOL (10 year LEO orbit) |
|---|---|---|---|
| **$\Delta V_{OC}$ /ΔT (mV/°C)** | Open Circuit Voltage | -5.6 | -5.8 |
| **$\Delta J_{SC}$ /ΔT (μA/(cm^2 °C)** | Short Circuit Current | 10.0 | 10.0 |
| **$\Delta V_{MP}$ /ΔT (mV/°C)** | Maximum Power Voltage | -6.3 | -6.4 |
| **$\Delta J_{MP}$ /ΔT (μA/(cm^2 °C)** | Maximum Power Current | 5 | 6.5 |

**Table 6.1: Temperature Coefficients of the XTJ-Prime Cells (15°C ~ 125°C).**

These characteristics can be extended to the following figures, which show the dependence of power and efficiency on the temperature. Note, the solar absorptance is 0.88 and the emittance is 0.85 for these cells.

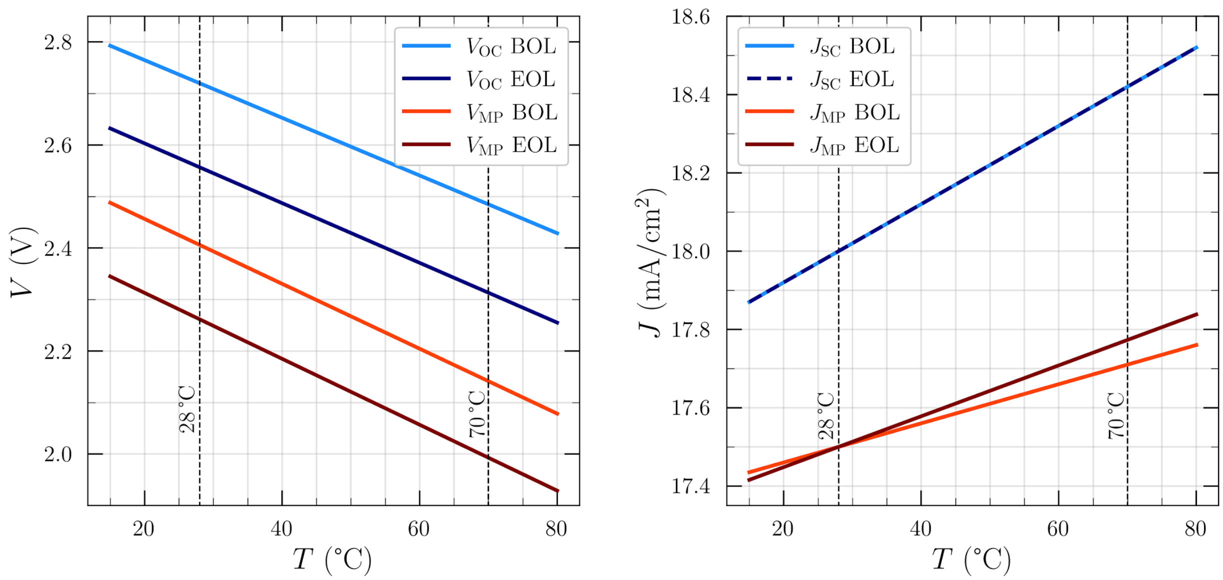



**Figure 24.1: Electrical characteristics of the cells at different states and at BOL and EOL.**

The above figure shows how the open circuit and max power voltages decrease with time (EOL vs BOL) and temperature, and how the short circuit and max power current densities increase with time and temperature. Note that the voltage decreases more than the current increases over the same temperature range, so it is expected that the power and efficiencies would decrease with increasing temperature. Note also that the conclusions made from this analysis deal with the BOL values, as the EOL values are pertinent to a 10-year mission, and the NS satellites are expected to only survive one or two years. It is evident that there would be little degradation over this period of time.

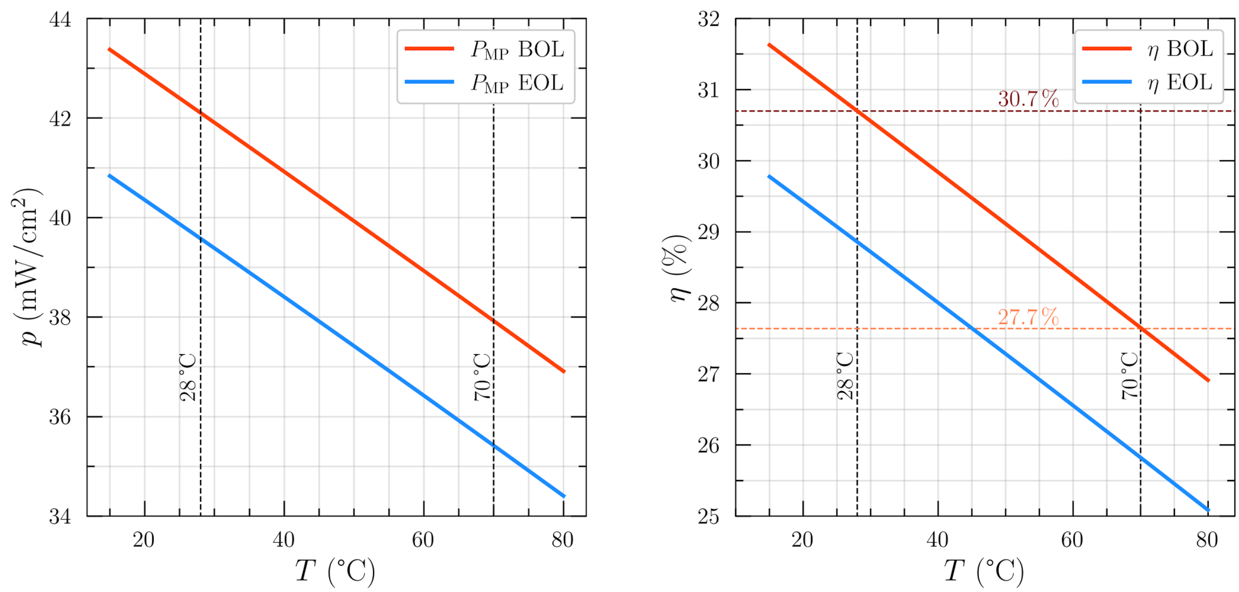



**Figure 24.2: Power and efficiency characteristics of the cells at different states and at BOL and EOL.**

As predicted, the efficiency of the cells decreases over time and with increasing temperature, though not by very much. Between 28°C and 70°C (the maximum expected temperature), we calculate a decrease in power density from only 42.1 ~ 37.9 mW/cm^2 which constitutes only a 9.9 % decrease. As well, we see only a 4% decrease in maximum power efficiency (30.7% ~ 27.7%). As seen in the power budgeting section later on in this document, this decrease in power in a worst case scenario is accounted for by the design. In conclusion, there is negligible risk introduced to the power budget by the heating of the solar cells through solar irradiance.

### 3.3.3 Solar Cell Application

The solar cells are applied to the substrate using a combination of polyimide tape and double-sided Kapton tape, as well as silver epoxy for the electrical connections. The precise procedure of application for the cells is described in the TX2-PW-429 Ver. 0.10 Solar Cell Adhesion document (R-08). The substrate that these cells are applied to can be seen in the following figures, where the cells are first adhered, then electrically connected through the application of silver epoxy. The large vias in the substrate are to electrically connect the positive terminals of the solar cells, and the large pads adjacent to the cells are to electrically connect the negative terminals. Note the large cutout under each of the cells - this is to provide room for the built-in bypass diode in the solar cell. The numerous small vias underneath each solar cell are placed in order to mitigate the risk of air bubbles forming when applying the kapton and polyimide tape - such non-idealities can produce considerable risk in vacuum.

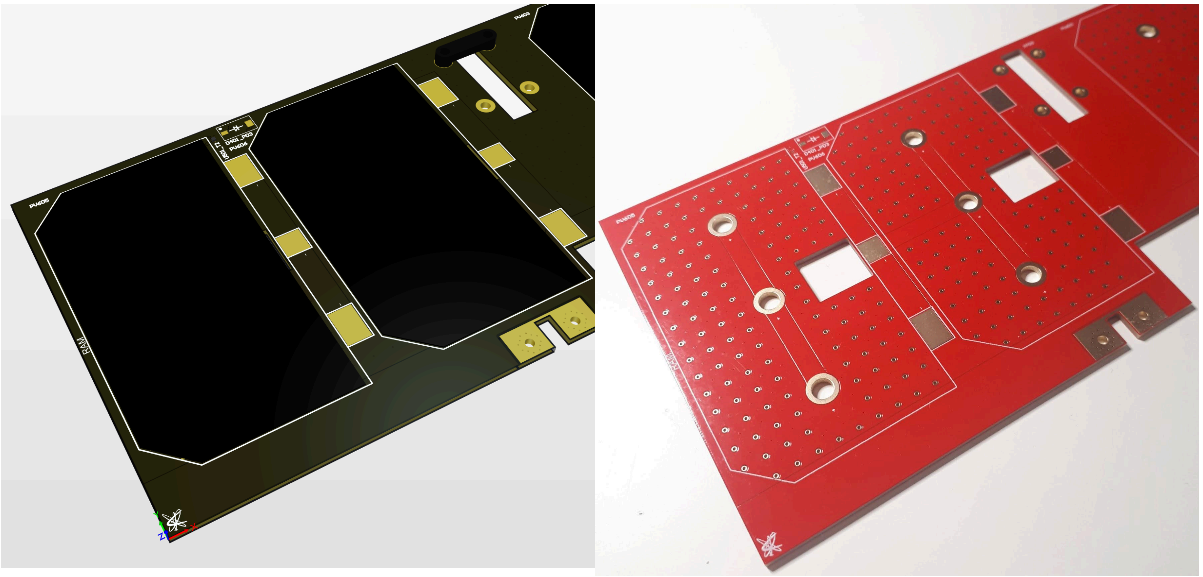


**Figure 25: Rendering of the Hyperion Starboard Deployable with attached solar cells (left), and photograph of the PCB substrate without solar cells (right)**

### 3.3.4 Temperature sensor

The temperature sensor selected for the solar panels is the LMT70YFQR made by Texas Instruments (TI). It can sense from -55°C ~ 150°C (+/- 0.05%) and is extremely small in size (0.88 mm x 0.88 mm). Each sensor is switched using an 8-Bit ADC which is in turn interfaced with the OBC. The LMT70YFQR outputs a voltage corresponding to the measured temperature: at 150℃, the chip outputs 303 mV, and at -55℃, the chip outputs 1375 mV, with an approximately linear dependence (~5 mV/℃). Each chip consumes 30 uW nominally.

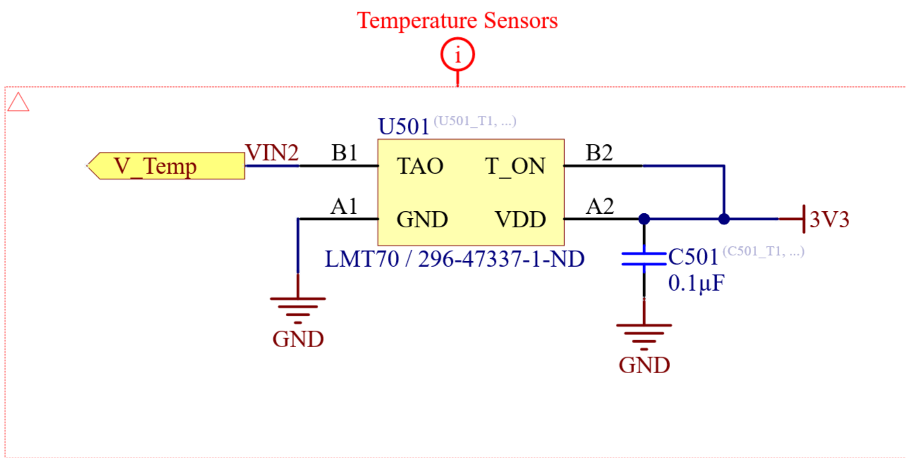



**Figure 26: Schematic design of the LMT70FQR. For more information please see the LMT70FQR Datasheet (R-09).**

Initial tests show that the temperature sensors produce an analog voltage equal to that expected at a certain temperature.

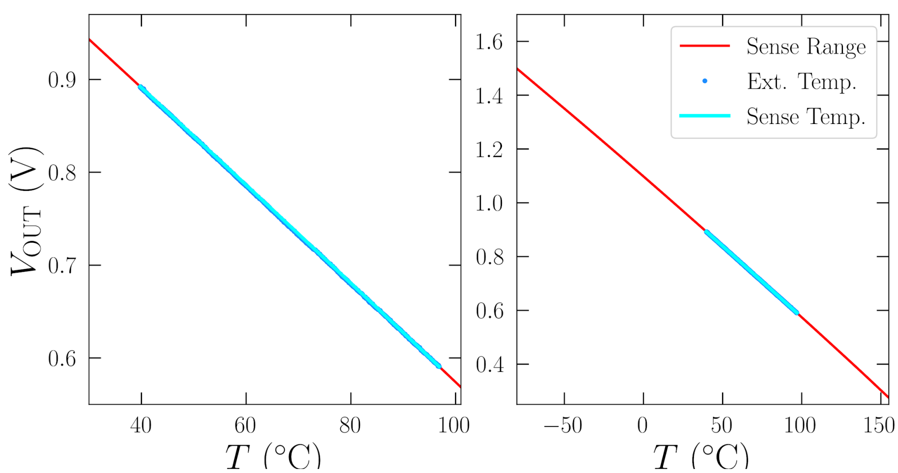



**Figure X: Measured analog output of the temperature sensor which matches the trend reported by the LMT70FQR.**

There are three sensors on each panel, distributed as shown in the component planning document, or as seen in Figure X. Each sensor is switched using an 8-Bit ADC, which is in turn interfaced with the OBC.

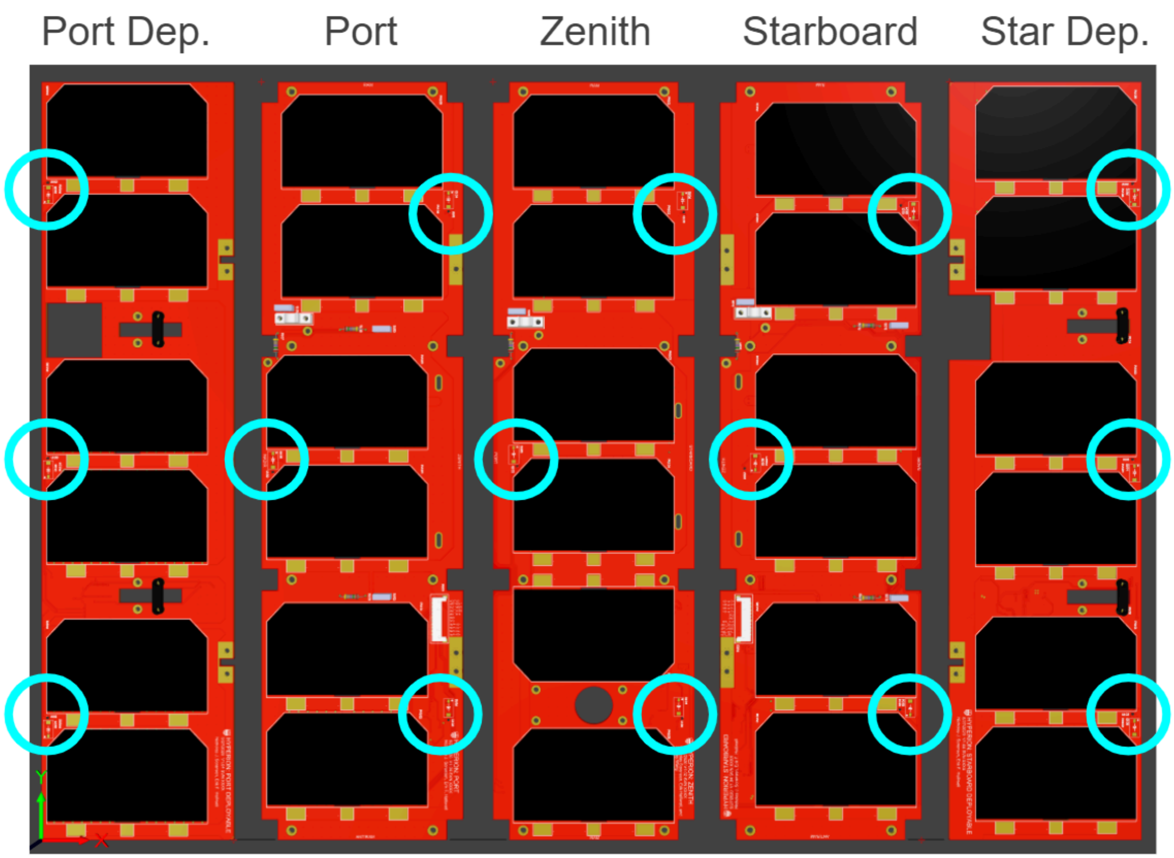



**Figure X: Temperature sensor placement on Hyperion.**

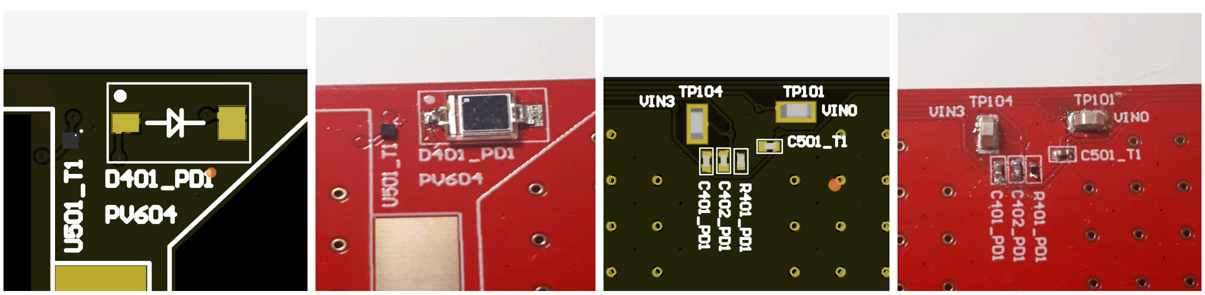



**Figure X1: Left: Front Sensor Rendering. Left-Middle: Front Sensor Photo. Right-Middle: Back Sensor Render. Right: Back Sensor Photo**

The impetus for putting temperature sensors on boards on CubeSats is to determine the nominal thermal characteristics of boards, and whether or not they are operating correctly. As per Figure X, sensors were placed in order to gather accurate information about the actual temperature of the boards, rather than the temperature of the chassis or attaching hardware. This means that body-mounted panels like Zenith, Starboard, and Port all have their temperature sensors as far from the rails as possible, while being distributed laterally as distinctly as possible. As for the deployable panels, all temperature sensors are placed oppositely to the hinges. There are two sensors, on port, placed somewhat close to the hinges to measure the temperature coupling between the body-mounted panels and the deployable panels.

### 3.3.5 Sun sensor

The photodiode selected as the sun sensor is the Advanced Photonics PDB-C160SM (R-10). It was chosen for its surface mountability, its wide sensing range (120°), and its active wavelength range (400-1100 nm) which is maximal in NIR (850 nm). See Figure X1 for renders and photos of the sun sensors.

Sun sensors are important components in determining the attitude of CubeSats, and are instrumental in the function of ADCSs. In both designs, there are two series of photodiodes on the solar panels: those used for housekeeping which interface with the OBC, and those that are wired to the ADCS. They provide rough orientation information for attitude adjustments. Each sun sensor consumes approximately 200 uW.

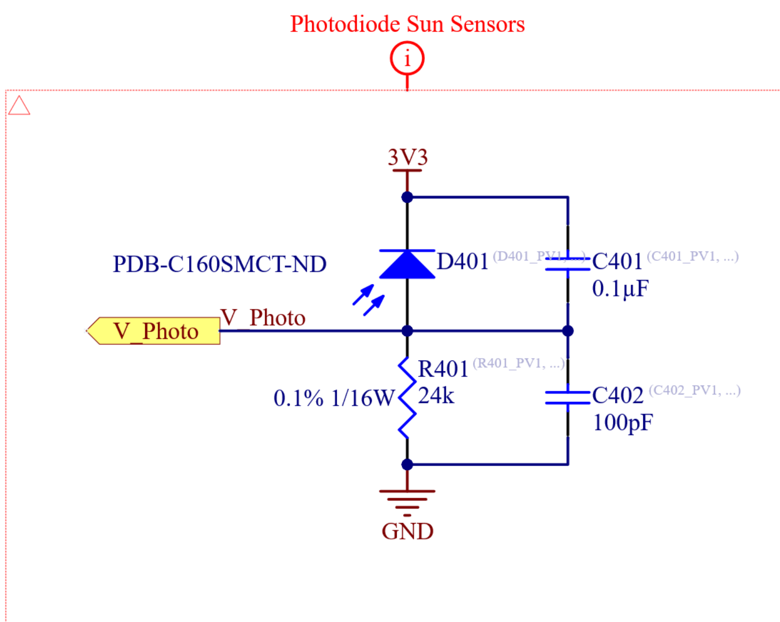



Figure 29: Schematic design of the PDB-C160SMCT-ND. For more information please see the PDB-C160SMCT-ND Datasheet (R-10).

When illuminated, the circuit shown in the above figure outputs a voltage between 0~2.5 V depending on the incident intensity, and each sensor is switched using an 8-Bit ADC, which is in turn interfaced with the OBC.

### 3.3.6 Voltage Sensor

The hyperion voltage sensor buffers the output voltage produced by the solar cells which is then switched by the ADC and interfaced to the OBC. The sensor uses an OPA2333PIDSGR (R-11) operational amplifier to buffer the output voltage while effectively isolating and dividing the produced voltage and the sensed analog voltage. As six XTJ-Prime solar cells in series maximally produce 16.32 VOC at BOL, we scale the voltage by 0.154 so that the buffer maximally outputs 2.50 V, matching the electrical limits of the ADC. Each amplifier consumes around 60 uW nominally.

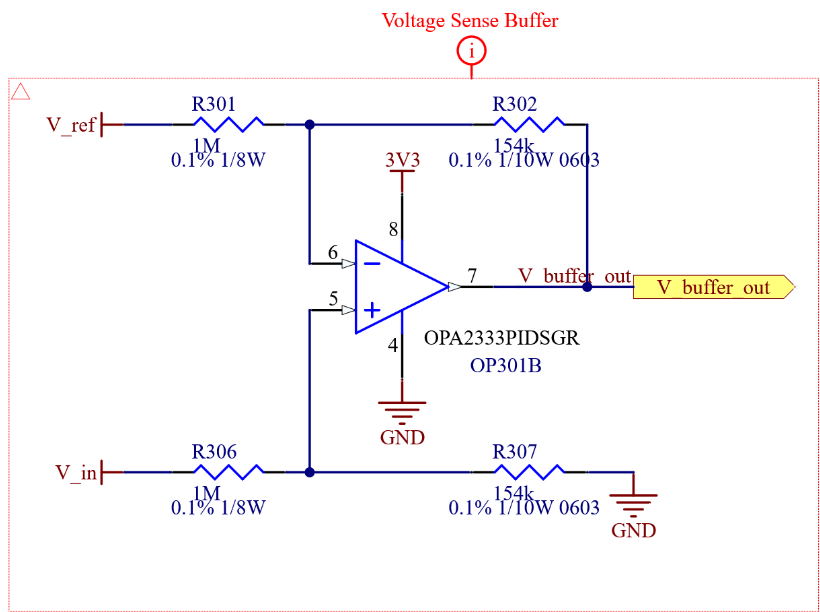



**Figure 30: Schematic design of the Voltage Sense Buffer.**

Some initial measurements show that the buffer works as intended (see the below figure).

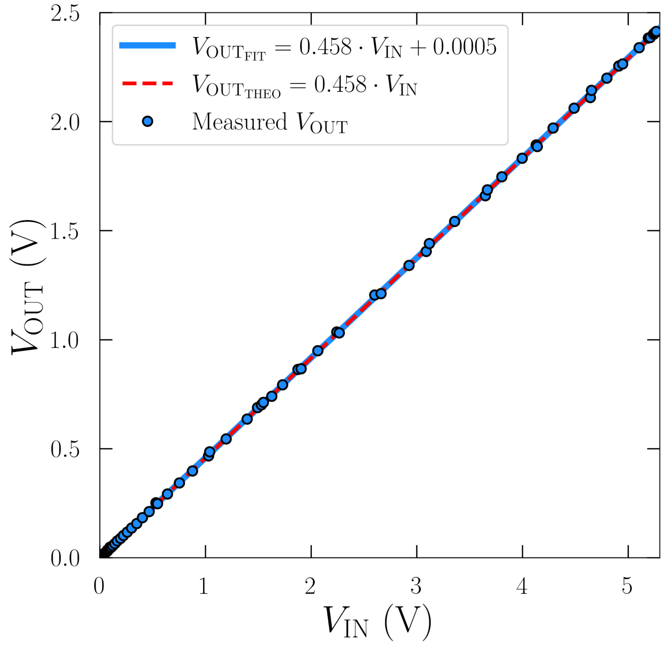



**Figure 31: Measured analog output of the voltage sense buffer.**

### 3.3.7 Current sensor

The hyperion current sensor measures the current produced by the solar cells and outputs this information as a voltage between 0 and 2.5 V. This is then switched by the ADC and interfaced to the OBC. The sensor uses an OPA2333PIDSGR (R-11) amplifier and consists of a simple differential amplifier configuration in conjunction with a sense resistor. The selected resistive elements result in a differential gain of 417, and as a 6S1P set of solar cells is expected to maximally produce 0.479 A, the selected resistors transform a current of 0.6 A into an analog voltage of 2.500 V, which approximates the ADC electrical limitations.

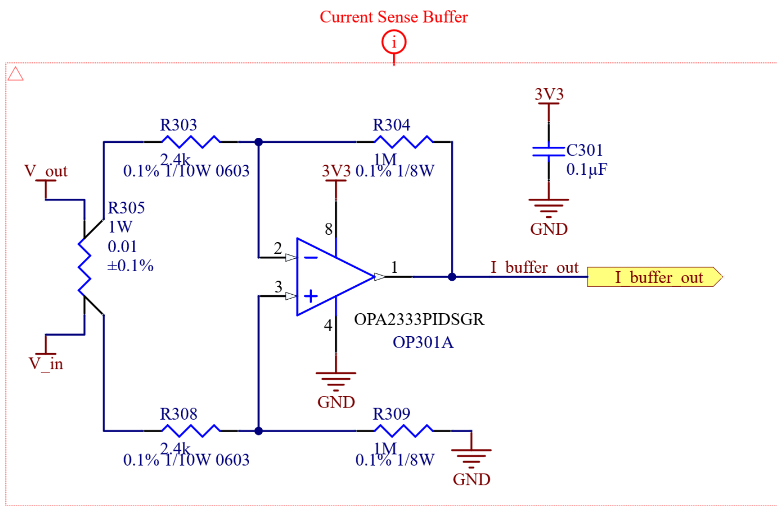



**Figure 32: Schematic design of the Current Sense Buffer.**

Some initial measurements show that the current sensor works as intended (see the below figure).

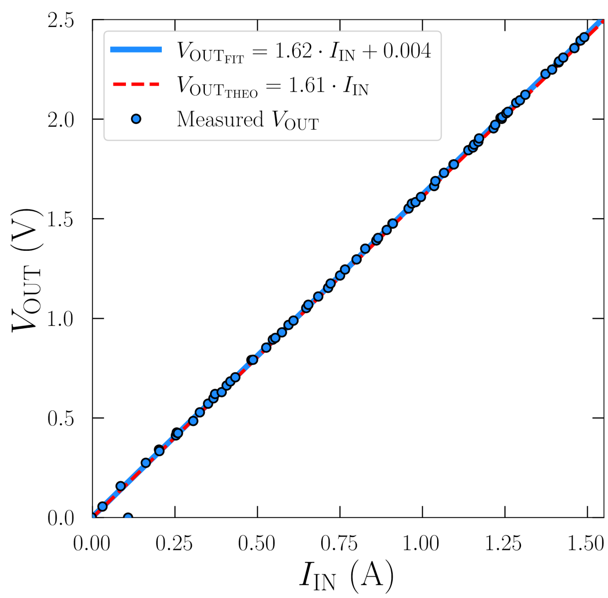



**Figure 33: Measured analog output of the current sensor.**

### 3.3.8 ADC

The ADC selected is the 12-Bit AD7291TCPZ-EP I2C (R-12), which takes up to 8 analog inputs and has an internal temperature sensor. It operates at 3.3 V with a wide range of addresses, so it is applicable to control and output the 8 Hyperion analog sensors: 3 sun sensors, 3 temperature sensors, a voltage sensor and a current sensor. The specifics of addressing and other more technical details will be discussed in following sections specific to each satellite. Each ADC consumes 10 mW nominally.

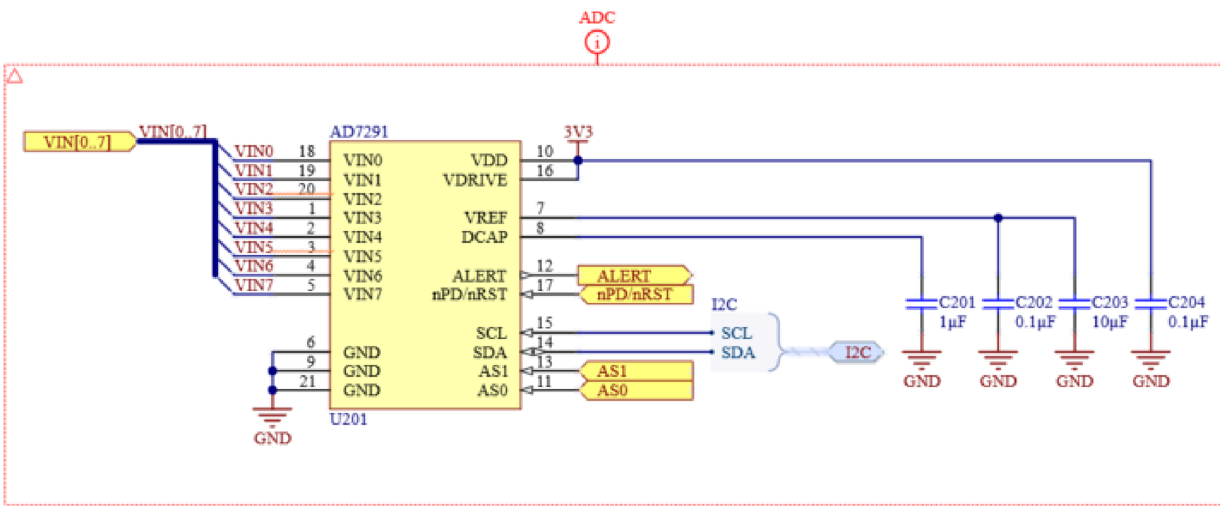



**Figure 34: Schematic design of the AD7291TCPZ-EP. For more information please see the AD7291TCPZ-EP Datasheet (R-12).**

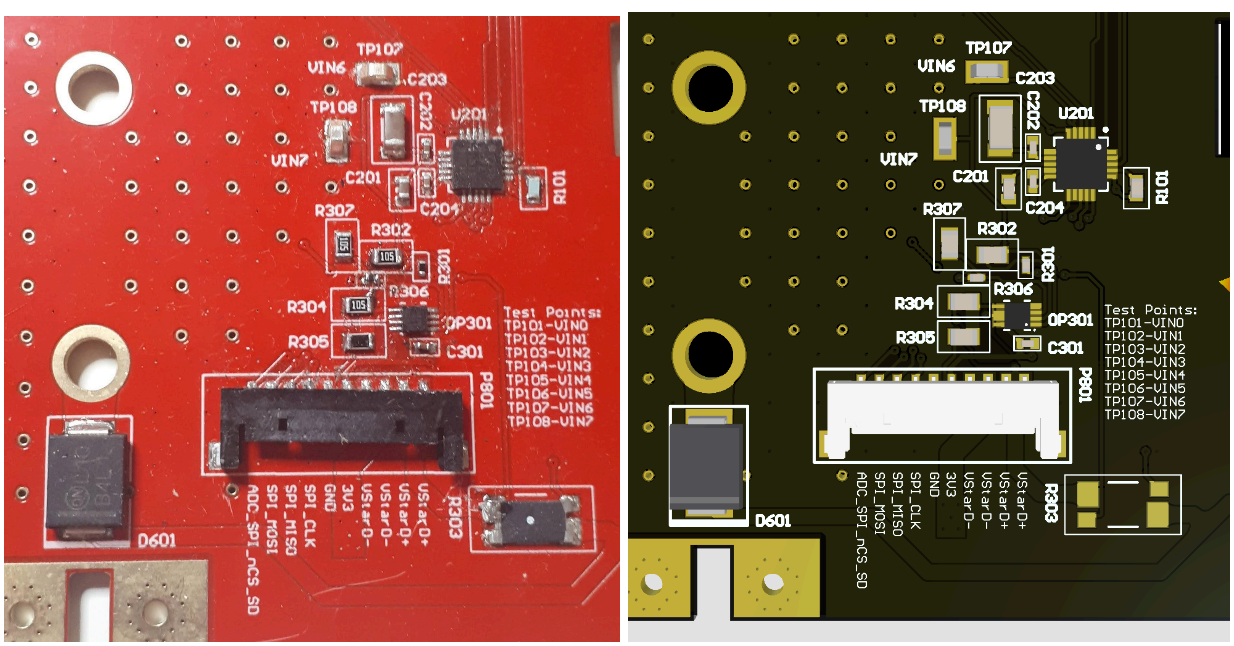



**Figure 35: Photo (left) and rendering (right) of the ADC and voltage/current sensing applications aboard Hyperion.**

### 3.3.9 Deployment Hinges

The discussion of the design of the hinges can be seen in the TX2-MC-127 Ex-Alta 2 Icarus Design CDR Document (R-23), and this document will discuss how they are used to connect the solar panels. Beneath are two drawings of the footprints that the hinges are mounted on. They are electrically and thermally connected to the deployable grounds, and to ensure electrical isolation (single point of grounding through the EPS), the hinges are electrically isolated but thermally grounded to the body-mounted panels.

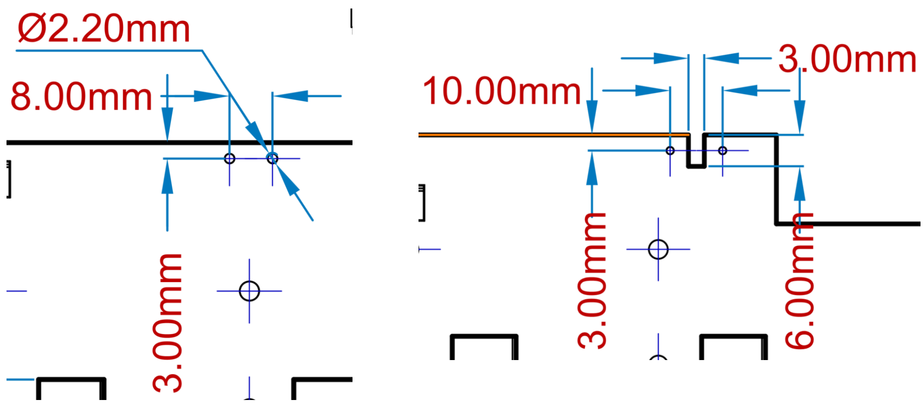



**Figure 36: Starboard hinge substrate (base) mechanical drawing (left); starboard deployable hinge substrate (hinged) mechanical drawing (right).**

### 3.3.10 Deployment Mechanism

The deployment mechanisms to release both the stowed antennae and payloads, as well as the deployable panels, are analogous. When the command to deploy is initiated, the OBC switches 5 V to power the burnwire resistors which are 10 Ω Flame Resistant Metal Film Resistors (CPF110R000FKEE6, R-13). The dissipated power will raise the temperature of the resistor to around 200°C which will burn the Dyneema thread (Berkley Solutions Braid (BSBFS10-22), melting point of ~130 °C) wrapped around the resistor. This will then free the component from a stowed position via a spring within the hinge (in the case of the deployable panels) deploying it. Note that as the power required to heat the resistors properly will need to be fine-tuned during AIT, we have included the possibility of adding SMT resistors in series with the burnwire resistors in order to allow for power tunability.

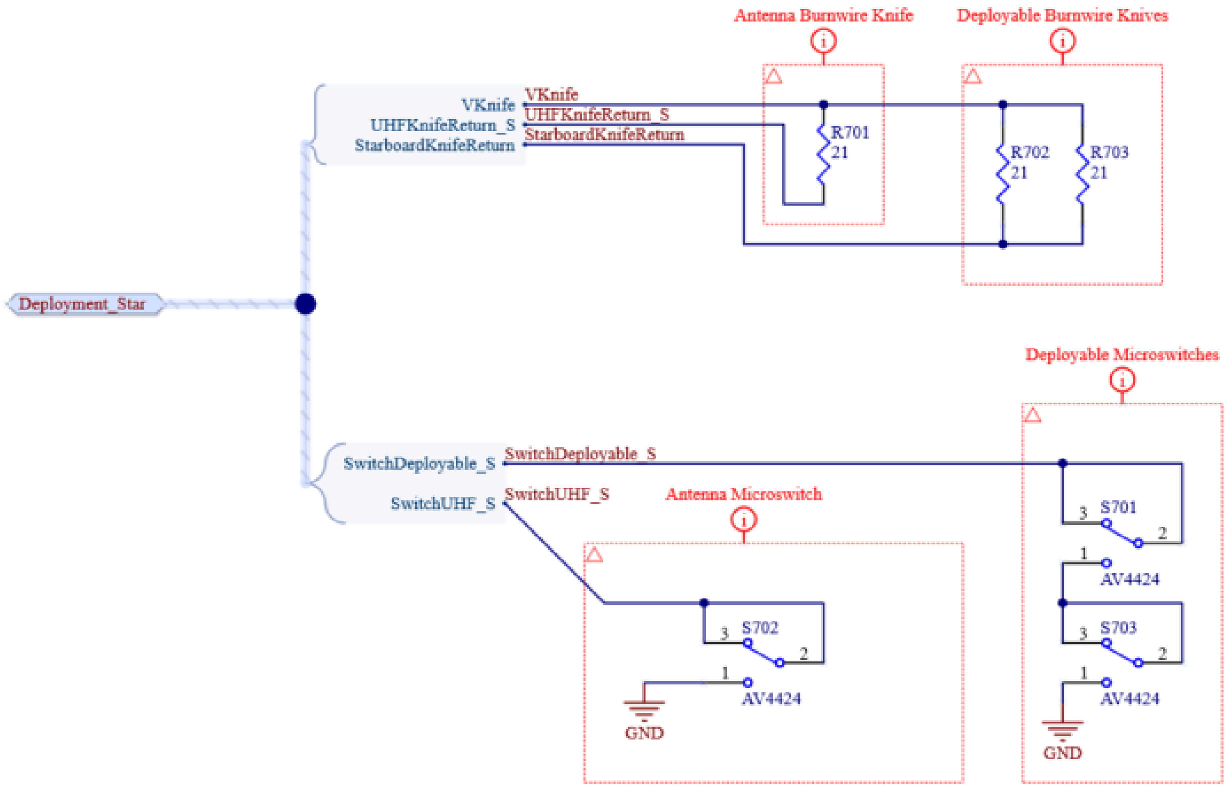



**Figure 38: Schematic design of the burnwire deployment mechanisms.**

In order to sense the deployment state of the component, the design uses SPDT switches (PANA-AV4424) (On-Momentarily) that are connected to ground when depressed. These signals are interfaced by the OBC that determines the state of the deployment mechanism. When the switches are both depressed the signals are pulled high by the OBC.

In order to meet the height requirements, the deployment switches and burnwire mechanisms were designed into the body-mounted panels and to make electronic harnessing easier, and all electrical mechanisms were designed into the body-mounted panels with appropriate cutouts. This allowed us to avoid designing any countersunk components into the panels. Renders of some of the appropriate cutout burnwire designs are shown below for both the UHF antenna burnwires and the deployable panel burnwires.

The filament will be kept as close to normal to the board as possible, and the resistors will likely be wrapped around twice in order to minimize the possibility of getting stuck once burnt (unlike what is shown in the figures).

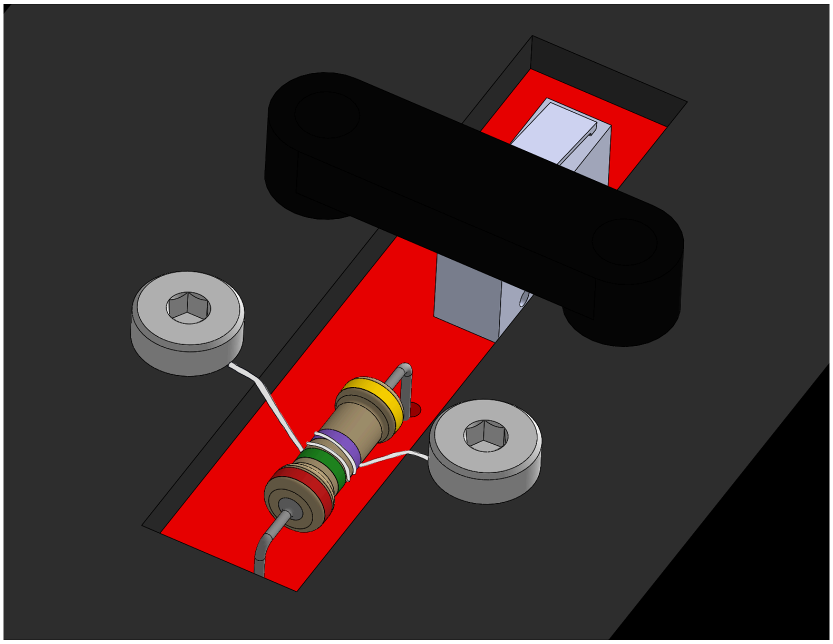
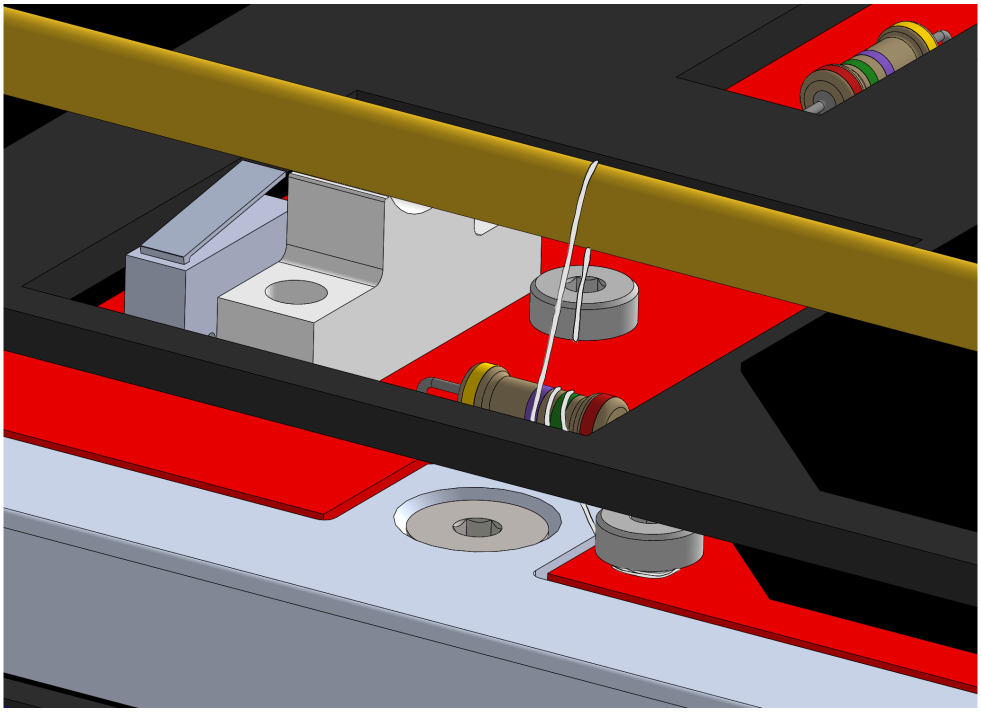


**Figure 39: Render of the solar panel burnwire mechanism. There are two of these switch/burnwire mechanisms on each deployable panel.**



## 3.3.11

### 3.3.12 Headers

The headers used on the panels must have a low profile and have a wide range of pin-allocations. This restricts selection to single row headers. Additionally, the inclusion of solar cells restricts us to the use of surface mounted headers as through-hole headers would restrict their placement too greatly. stress can be mitigated by placing through-stack pads on the mounting pads for the headers. The PicoLock Header manufactured by Molex has been selected. It has a low profile (2mm) which is required to fit in between the body mounted and deployable panels. It is additionally used on the likely COTS Nanoavionics EPS, so this shows heritage and reduced the variability in headers throughout the satellite. Other candidates included the Picoblade series, but these proved to be too tall for our application (3.4 mm).

## 3.4 Software Interface

The Hyperion boards interface with Athena via the I2C ADCs aboard each panel as shown in the following figure. Note that the Arke and Nadir Panels are also included in this figure as they have the same ADC with similar sensing requirements.

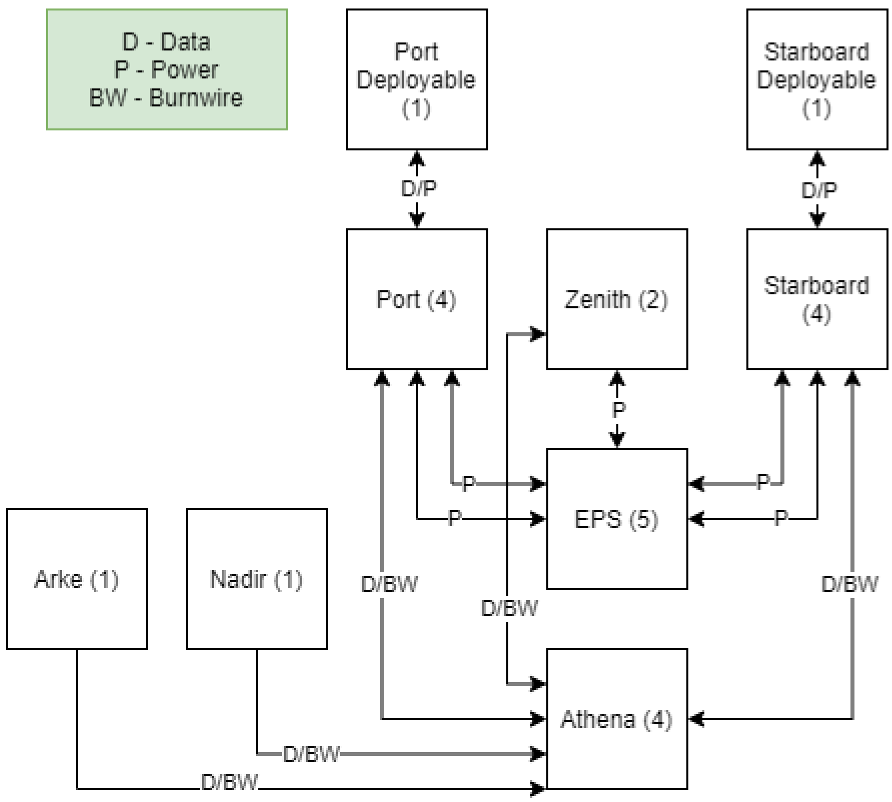
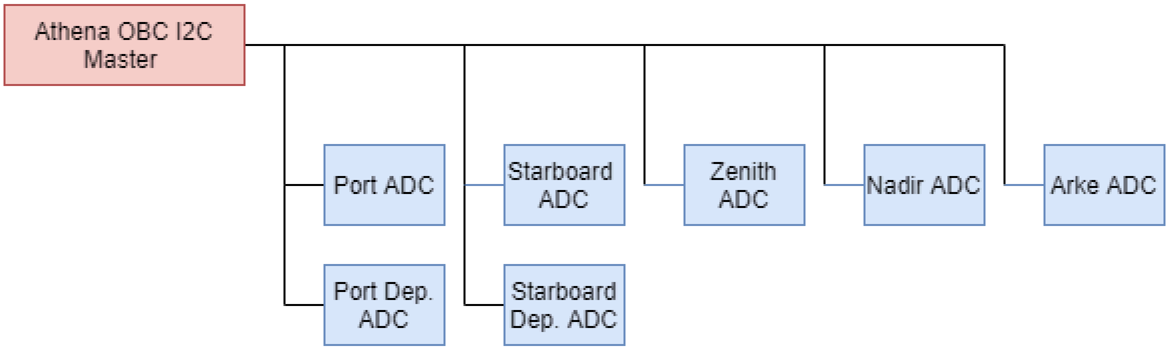



Figure X: Block Diagram for harnessing between the Athena OBC and the Hyperion, Nadir, and Arke Panels.

A distinction must be made, however, between the ADC designations for the different devices as they are not all the same. The following figure shows how the I2C line connects the Athena OBC master to the 7 slave devices, or the solar panels in addition to Arke, the UHF Phasing board, and the Heracles Nadir panel. The solar panel I2C drivers aboard Athena need to control and communicate with these as well.



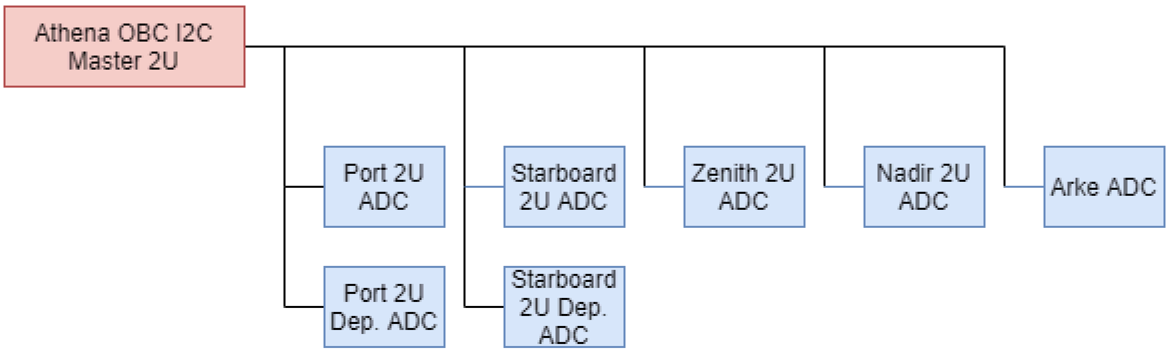




Figure X: I2C connections for both Ex-Alta 2 (top) and the 2U satellites (bottom).

The ADC has several capabilities that are detailed in the datasheet: these include the 12 bit A-D conversion of 8 channels (VIN0 to VIN7), as well as temperature sensing among others. The main functionality of the ADCs is to convey the signals provided by 4 different sensors on the panels: temperature sensors, photodiodes, voltage sensors, and current sensors. The design of these components is given in the above section, as well as what the sensed voltages mean.

The voltage channel distributions for each ADC are given in the following figures.

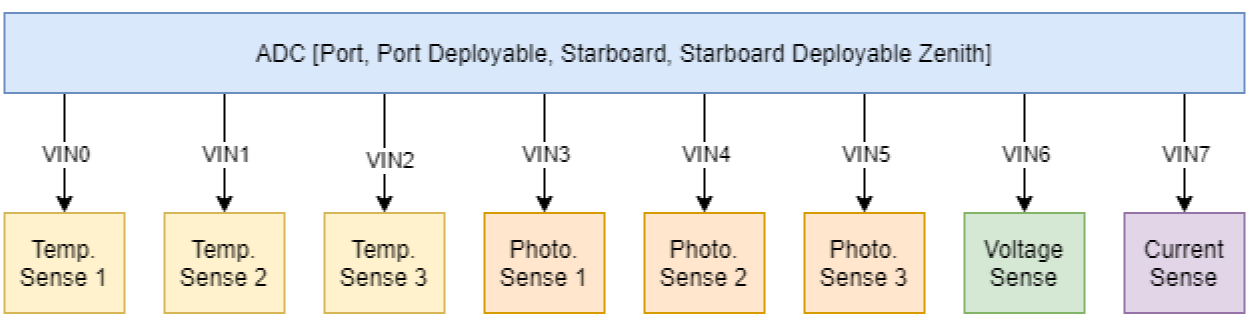



Figure X: The voltage channel allocations for Port, Port Deployable, Starboard, Starboard Deployable, and Zenith are all the same and are given here.

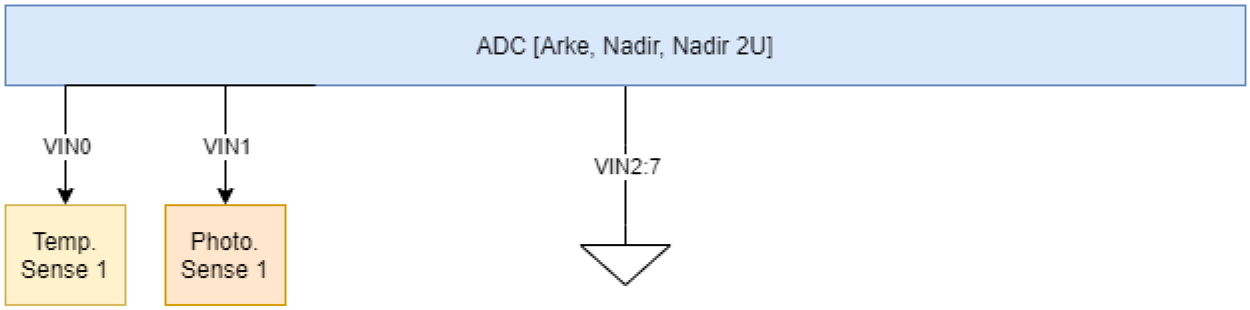



Figure X: The voltage channel allocations for Nadir, Nadir 2U, and Arke are all the same and are given here.Only VIN0 and VIN1 have meaningful voltages - the others are grounded.

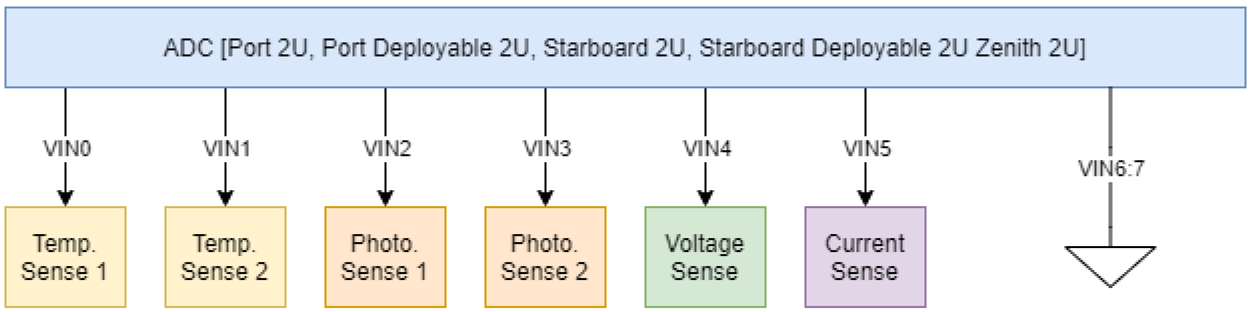



Figure X: The voltage channel allocations for Port 2U, Port Deployable 2U, Starboard 2U, Starboard Deployable 2U, and Zenith 2U are all the same and are given here.

The OBC software should be able to in a single command choose the panel and the channel that is being converted, as well as the capability for ADC temperature conversion.

The I2C addresses for each of the panels are given in the following table:

| Panel | AS1 | AS0 | Slave Address (A6 to A0) | |
|---|---|---|---|---|
| | | | Binary | Hex |
| Port | H | H | 010 0000 | 0x20 |
| Port Deployable | H | NC | 010 0010 | 0x22 |
| Starboard | H | L | 010 0011 | 0x23 |
| Starboard Deployable | NC | H | 010 1000 | 0x28 |
| Zenith | L | NC | 010 1110 | 0x2E |
| Port 2U | H | H | 010 0000 | 0x20 |
| Port Deployable 2U | H | NC | 010 0010 | 0x22 |
| Starboard 2U | H | L | 010 0011 | 0x23 |
| Starboard Deployable 2U | NC | H | 010 1000 | 0x28 |
| Zenith 2U | L | NC | 010 1110 | 0x2E |
| Arke | L | H | 010 1100 | 0x2C |
| Nadir | L | L | 010 1111 | 0x2F |
| Nadir 2U | L | L | 010 1111 | 0x2F |

Table X: I2C addresses for the panel ADCs. AS1 and AS0 are input pins that will set the address

| Designator | Type | Unit | Ex-Alta 2 | ARI/YU |
|---|---|---|---|---|
| 00 | float | °C | NADIR TEMP 1 | NADIR TEMP 1 |
| 01 | float | °C | PORT TEMP 1 | PORT TEMP 1 |
| 02 | float | °C | PORT TEMP 2 | PORT TEMP 2 |
| 03 | float | °C | PORT TEMP 3 | Reserved |

| | | | | |
|---|---|---|---|---|
| 04 | float | °C | PORT TEMP ADC | PORT TEMP ADC |
| 05 | float | °C | PORT DEP TEMP 1 | PORT DEP TEMP 1 |
| 06 | float | °C | PORT DEP TEMP 2 | PORT DEP TEMP 2 |
| 07 | float | °C | PORT DEP TEMP 3 | Reserved |
| 08 | float | °C | PORT DEP TEMP ADC | PORT DEP TEMP ADC |
| 09 | float | °C | STAR TEMP 1 | STAR TEMP 1 |
| 10 | float | °C | STAR TEMP 2 | STAR TEMP 2 |
| 11 | float | °C | STAR TEMP 3 | Reserved |
| 12 | float | °C | STAR TEMP ADC | STAR TEMP ADC |
| 13 | float | °C | STAR DEP TEMP 1 | STAR DEP TEMP 1 |
| 14 | float | °C | STAR DEP TEMP 2 | STAR DEP TEMP 2 |
| 15 | float | °C | STAR DEP TEMP 3 | Reserved |
| 16 | float | °C | STAR DEP TEMP ADC | STAR DEP TEMP ADC |
| 17 | float | °C | ZENITH TEMP 1 | ZENITH TEMP 1 |
| 18 | float | °C | ZENITH TEMP 2 | ZENITH TEMP 2 |
| 19 | float | °C | ZENITH TEMP 3 | Reserved |
| 20 | float | °C | ZENITH TEMP ADC | ZENITH TEMP ADC |
| 20 | float | % | NADIR PD 1 | NADIR PD 1 |
| 21 | float | % | PORT PD 1 | PORT PD 1 |
| 22 | float | % | PORT PD 2 | PORT PD 2 |
| 23 | float | % | PORT PD 3 | Reserved |
| 24 | float | % | PORT DEP PD 1 | PORT DEP PD 1 |
| 25 | float | % | PORT DEP PD 2 | PORT DEP PD 2 |
| 26 | float | % | PORT DEP PD 3 | Reserved |

| | | | | |
|---|---|---|---|---|
| 27 | float | % | STAR PD 1 | STAR PD 1 |
| 28 | float | % | STAR PD 2 | STAR PD 2 |
| 29 | float | % | STAR PD 3 | Reserved |
| 30 | float | % | STAR DEP PD 1 | STAR DEP PD 1 |
| 31 | float | % | STAR DEP PD 2 | STAR DEP PD 2 |
| 32 | float | % | STAR DEP PD 3 | Reserved |
| 33 | float | % | ZENITH PD 1 | ZENITH PD 1 |
| 34 | float | % | ZENITH PD 2 | ZENITH PD 2 |
| 35 | float | % | ZENITH PD 3 | Reserved |
| 36 | float | mV | PORT VOLTAGE | PORT VOLTAGE |
| 37 | float | mV | PORT DEP VOLTAGE | PORT DEP VOLTAGE |
| 38 | float | mV | STAR VOLTAGE | STAR VOLTAGE |
| 39 | float | mV | STAR DEP VOLTAGE | STAR DEP VOLTAGE |
| 40 | float | mV | ZENITH VOLTAGE | ZENITH VOLTAGE |
| 41 | float | mA | PORT CURRENT | PORT CURRENT |
| 42 | float | mA | PORT DEP CURRENT | PORT DEP CURRENT |
| 43 | float | mA | STAR CURRENT | STAR CURRENT |
| 44 | float | mA | STAR DEP CURRENT | STAR DEP CURRENT |
| 45 | float | mA | ZENITH CURRENT | ZENITH CURRENT |

Table X: Housekeeping allocation.

## 3.5 Requirement Compliance

This section outlines the system's compliance to the requirements outlined at the start of this document.

| Colour Scheme | Compliant | Semi-Compliant | Not Compliant. |
|---|---|---|---|

| Requirement Number | Design Specification/Requirement | Implementation |
|---|---|---|
| **Functional** | | |
| **System** | | |
| ASX2-SYS-7.050 | No pyrotechnics shall be on the satellite unless they are pre-approved by NanoRacks. Melt wire systems for deployables are permitted. | There are burnwire implementations for the deployables. These are permitted explicitly. |
| ASX2-SYS-7.121 | All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris will be created. | No actual parts are fully released from the solar panel design. All parts remain attached to the satellite in some form. |
| ASX2-SYS-7.140 | Vented fasteners shall be used for blind holes. | All fasteners attached to Hyperion will be vented. |
| ASX2-SYS-7.160 | All spacecraft components shall be electrically bonded per SSP 30245 to ensure the spacecraft is free from electrical shock and static discharge hazards. Typically, spacecraft components may be bonded by either nickel plating or chemical film treated faying surfaces or dedicated bonding straps. | All components are appropriately nickel plated. |
| ASX2-SYS-7.210 | Stress corrosion resistant materials from Table I of MSFC-SPEC-522 are preferred. Any use of stress corrosion susceptible materials (Table II) shall be coordinated with NanoRacks and documented in the ICA. Any use of Table III materials shall be avoided. | All screws used will likely be 300 series steel, which is on Table I. |
| ASX2-SYS-7.220 | All components shall be marked with axis orientations where appropriate | Each panel is marked with their orientation with regard to Zenith, etc. |
| ASX2-SYS-7.230 | All components shall be marked with hardware and firmware versions where appropriate. | There is appropriate dating and versioning on each panel. |
| ASX2-SYS-7.240 | All components shall be designed to avoid floating voltages. | All applied voltages are decoupled to ground. |
| **Electrical Power** | | |
| ASX2-SYS-3.040 | The power system shall not create magnetic fields that will cause destabilization of the SC. | Parasitic magnetic fields and current loops are designed against. |

| | | |
|---|---|---|
| ASX2-SYS-3.070 | The CubeSat shall have a RBF or ABF feature that keeps the satellite in an unpowered state throughout handling and integration into the NanoRacks deployment pod. | All powering is interfaced through Athena, which is actuated by the EPS, which is switched by RBF pins. |
| ASX2-SYS-3.080 | The RBF/ABF feature shall preclude any power from any source from operating any satellite functions except for pre-integration battery charging. | The solar panels do not function when the RBF feature is not used. |
| ASX2-SYS-3.210 | All electrical power storage devices shall be internal to the CubeSat. | There are none, barring the EPS, and this is internal to the CubeSat. |
| **Mechanical** | | |
| ASX2-SYS-4.045 | The CubeSat rails shall be the only mechanical interface to the NRCSD in all axes (X, Y and Z axes). This means nothing can overlap the minimum width of the rails. | No overlap occurs.with the minimum width of the rails. The deployable panels overlap the tabs, but not the rails. |
| ASX2-SYS-4.260 | The DFGM shall be mounted to the exterior of the satellite. | This is taken into account with the dimensions of Nadir. |
| ASX2-SYS-4.390 | The CubeSat shall be capable of being integrated forwards and backwards inside of the NRCSD (the +/-Z face could be deployed without issue). Note: In general, the deployables should be hinged towards the front of the deployer to mitigate risk of a hang-fire should the deployables be released prematurely while the CubeSat is still inside the NRCSD. | Hyperion allows the CubeSat to be integrated into the NRCSD in either direction. The deployables do not follow the recommendation in the note, as the deployable magnetometer is hinged in both directions |
| **Performance** | | |
| **System** | | |
| ASX2-SYS-7.010 | The CubeSat shall survive within the temperature range of −40◦C to +65◦C from the time of launch until its end of life. | All components are chosen to fit within this range. |
| ASX2-SYS-7.020 | The CubeSat shall not use any material that has the potential to degrade in an ambient environment during storage after assembly for a length of 2 years. | All materials are chosen for their space durability. |
| ASX2-SYS-7.060 | Satellites shall comply with NASA guidelines for selecting all non-metallic materials based on available outgassing data. Non-metallic materials shall have a Total Mass Loss (TML) less than or equal to 1.0% and Collected Volatile Condensable Material (CVCM) value less than or equal to 0.1%. | Components are tested to have a TML less than or equal to 1%. |

| | | |
|---|---|---|
| ASX2-SYS-7.110 | The CubeSat shall be capable of withstanding pressure extremes from 0 to 104.8 kPa and depressurization/pressurization rate 1.0 kPa/second. | We will employ a vacuum chamber in order to test these characteristics. |
| ASX2-SYS-7.170 | The CubeSat shall be capable of withstanding the relative humidity environment for all mission phases leading up to deployment, which is between 25% to 75% relative humidity (RH) for ascent and on-orbit phases of flight. | The solar panels are testing in a humidity-controlled environment. |
| ASX2-SYS-7.250 | All SC systems shall operate in a vacuum of 10^-10 Torr. | Systems will be tested in HV, or approx 10^-6 Torr. |
| **Electrical Power** | | |
| ASX2-SYS-3.020 | The SC must be able to supply at least 1.5 Wh during a worst case scenario orbit. TBC upon final component selection. | Worst case scenario orbit produces > 6Wh. |
| **Mechanical** | | |
| ASX2-SYS-4.040 | The CubeSat rails and envelope shall adhere to the dimensional specification outlined in NR-NRCSD-S0003 Figure 4.1.1-1. | The envelope adheres to the dimensional guidelines. |
| ASX2-SYS-4.160 | The CubeSat shall have an approved secondary locking feature for any and all fasteners or subcomponents external to the CubeSat chassis that would not be held captive by the spacecraft structure should it come loose. | All fasteners have locking washers. |
| ASX2-SYS-4.250 | The imager shall have a minimum clear half-angle FOV of 5 degrees surrounding the aperture. TBC | When deployed, the deployable panels are completely out of range of the FOV of the imager. |
| ASX2-SYS-4.270 | The exterior of the satellite shall accommodate the DFGM sensor head when stowed. The sensor head has the dimensions 38x32x28 mm. | Hyperion is is made to fit with the DFGM plate. |
| ASX2-SYS-4.290 | All RBF/ABF and charging systems for the integration process must be located on one side of the satellite (+Y in the NRCSD frame) and in the access port locations shown in NR-NRCSD-S0003, Figure 3.3-1. | All located on Nadir. |
| ASX2-SYS-4.350 | The DFGM wiring bundles that come out of the boom shall connect directly to the DFGM PC104 board. | Does not interfere or interface with Hyperion at all. |
| ASX2-SYS-4.380 | CubeSat deployable systems (solar arrays, antennas, payload booms, etc.) shall have independent restraint | There are independent burnwires for all deployable mechanisms. |

| | | |
|---|---|---|
| | mechanisms that do not rely on the NRCSD dispenser. | |
| **Spacecraft tests** | | |
| **General** | | |
| ASX2-TST-1.010 | Internal connectors shall not be disconnected after the completion of the spacecraft acceptance test program. | After the spacecraft acceptance tests, nothing needs to be disconnected. |
| ASX2-TST-1.020 | All remove before flight items shall be easily accessible and removable without any disassembly of the SC. | All are directly available from Nadir. |
| ASX2-TST-1.040 | The SC shall incorporate a connector easily accessible to allow battery charging and basic functional verification of the SC during ground testing and pre-launch operations. | Nadir panel provides this interface for Ground Charging. |
| ASX2-TST-1.050 | Each subsystem shall be tested independently prior to unit assembly. | Hyperion will be tested independently before other integration. |
| **Mechanical** | | |
| ASX2-TST-2.070 | The SC shall include handling points to permit lifting. | There are none on Hyperion. |
| ASX2-TST-2.080 | The SC components shall fit inside or on the structure. | Hyperion is designed to fit on Icarus. |
| ASX2-TST-2.090 | The SC shall fit inside the NRCSD without any contact except the outer rails. | Hyperion does not contact the NRCSD. |
| ASX2-TST-2.100 | Deployment mechanisms shall be verified to work by deployment tests. | There are rigorous tests to verify this. |
| ASX2-TST-2.140 | Every element shall be designed to have a positive Safety Margin. The Safety Margin is defined as: (Max allowable load or stress)/(Predicted design load or stress)-1 | Safety margins have been accounted for. |
| **Electrical** | | |
| ASX2-TST-3.010 | All solder joints must be inspected after component assembly and before integration into larger assemblies. | There are testing procedures for solder joint inspection. |

## 3.6 Recommendation Compliance

This section outlines the system's compliance to the recommendations outlined at the start of this document.

| Requirement Number | Design Specification/Requirement | Implementation |
|---|---|---|
| **Functional** | | |
| **System** | | |
| Recommendation 4.110 | All electronic assemblies and electronic circuit boards should be conformally coated. | Likely implemented on all boards. |
| Recommendation 4.120 | Magnetizable material should not be used for CubeSat parts. | No materials used on the panels are ferromagnetic/ferrimagnetic. |
| Recommendation 4.140 | All soldered components should have the ability to be manually reworked. | All pads are large enough to account for this. |
| **Performance** | | |
| **System** | | |
| Recommendation 4.115 | All components used on in-house designs should have an operational temperature rating of at least -40◦C to +105◦C. | Mostly. |

# 4 Testing

See the Hyperion Testing Plan for more information [R-08].

# Acronyms and Abbreviations

Update this to reflect the acronyms and abbreviations used in this report.

| | |
|---|---|
| **ADCS** | Attitude Determination and Control System |
| **CAD** | computer aided design |
| **Comms** | Communications (as in Comms team) |
| **CSDC** | Canadian Satellite Design Challenge |
| **CSDCMS** | CSDC Management Society |
| **cubesat** | cube satellite |
| **C&DH** | Command and Data Handling (as in C&DH team) |
| **Ex-Alta** | Experimental Albertan |
| **HEPA** | high-efficiency particulate arrestance |
| **PDM** | product data management |
| **UHF** | ultra high frequency |
| **VHF** | very high frequency |