SSC17-IX-05

CubeSat Sensor Platform for Reentry Aerothermodynamics

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ABSTRACT

Even with over 50 years of flying to and returning from Earth orbit, scientists and engineers still lack sufficient data to validate chemical reaction rate models for nonequilibrium reentry flows. This leads to increased mission weight and cost due to the need for more substantial thermal protection system margins. Reaction rates are more accurately determined with flight data than with ground-based testing due to the difficulties in reproducing high enthalpy, low-density flows on the ground. Of the handful of missions that have attempted to gather the necessary information, none have successfully provided science-grade data for a non-ablative vehicle at orbital velocities. Deorbiting CubeSats are ideally suited to collect the uncontaminated reentry data needed to validate atmospheric reentry models. A student team at Purdue University, as part of the Student Aerothermal Spectrometer of Illinois and Indiana (SASSI[2]) project, has developed a CubeSat sensor platform to take advantage of the natural reentry experienced by all CubeSats. The sensor platform will measure bulk flow properties as well as ambient conditions. Once combined with chemical species information from onboard spectrometers, this data will enable scientists and engineers to determine the chemical reaction rates needed to validate their models.

INTRODUCTION

As the use of Earth orbit by both national space programs and private companies continues to increase, so too will the need for more accurate atmospheric reentry models. In February of 2017, the Indian Space Research Organization (ISRO) launched 104 satellites at one time, shattering the previous world record of 38 set in 2014[1], and a number of companies including PlanetLabs, Spire, OneWeb, and SpaceX have plans to place thousands of small satellites into drag-heavy orbits below 400 kilometers.[2] The International Space Station (ISS) has been resupplied by private companies SpaceX and Orbital ATK 17 times in six years, with an additional six missions planned before 2018.[3] In April of 2016, Bigelow Aerospace attached the Bigelow Expandable Activity Module (BEAM) to the ISS to demonstrate future private space stations, and SpaceX and Boeing are scheduled to begin delivering astronauts to the ISS in 2018. Blue Origin and Virgin Galactic are promising sub-orbital space tourism for the masses and United Launch Alliance is now awarding free rides to Low Earth Orbit for University CubeSats. This increase in the utilization of space by private companies is happening while NASA and its partner agencies are developing the next generation of vehicles that will take humans further into the solar system than ever before and then bring them back home. The one thing that each of these missions has in common is that at some point, every single one of them will reenter Earth's atmosphere.

More accurate models of the reentry conditions these spacecraft will encounter will allow Thermal Protection Systems (TPS) to become safer and more cost-effective while providing improved estimates of mission lifetime for commercial satellites. Reentry TPS is a single point of failure, so engineers are understandably conservative with their designs. The Apollo heat shield made up over 10 percent of the command module weight and never used more than 20 percent of the available ablator.[4] While substantial TPS material improvements have been made since the Apollo era, improving the confidence in current reentry models has the potential to produce a significant reduction in current margins. This reduction is especially valuable to both private companies concerned with safety and cost, and to deep space missions where every pound of TPS is one less pound dedicated to the mission.

Ground Testing

The only methods for doing ground based testing are through use of an arc-jet, like the one at the Johnson Space Center Atmospheric Reentry Materials and Structures Evaluation Facility (ARMSEF), or a light gas gun like the one in the NASA Ames Hypervelocity Free-Flight Gun Development Facility (HFFGDF). Arc-jet testing can reproduce the high enthalpies encountered during re-entry, but it does so at lower velocities. Therefore, while it creates a representative amount of heat flux, it does not accurately reproduce the nonequilibrium chemistry that affects heat loads, ablation rates, and aerodynamic coefficients. The light gas gun works by propelling a projectile into a representative gas. Maximum velocities of just over 11 km/s have been attained using this method, however 8 km/s remains the typical peak velocity used, and is primarily limited to impact research.[5] This method can simulate the physics and chemistry needed to validate reentry models, but attaining the necessary velocities can damage the facility, making it costly to obtain speeds approaching the minimum energy reentry velocities of 7-8 km/s. Additionally, obtaining data from this method is often challenging due to the almost instantaneous nature of the tests, which are often contaminated by the light gas used to propel the projectile. As a result of these limitations, many scientists and engineers are looking to actual flight data to validate their models.[6]

Past Missions

In 1962, NASA Langley began Project FIRE to attempt to understand reentry conditions before the start of the Apollo program. While these missions reached velocities over 11 km/s, they were not equipped to look at a wide spectral range, and therefore were not able to provide the high-fidelity chemistry information that is currently required. Apollo 4 and 6 provided additional reentry data for both shallow and steep reentry profiles, but were not capable of detecting flow chemistry. During the Shuttle era, investigations into the phenomenon called "shuttle glow" provided additional insight into the nonequilibrium chemistry around the orbiting vehicle. It is believed that the orange glow that was visible along the leading edges of the shuttle was caused by the recombination of O with NO on the surface; however, "The shuttle glow in the infrared region of the spectrum is not well understood and requires measurements at higher spectral resolution to identify the emitters definitively".[7] Throughout the subsequent years, there have been no missions that have reached orbital reentry velocities while collecting satisfactory chemical reaction rate data of a flow uncontaminated by an ablative heat shield. It should be noted that in addition to onboard sensors, remote imaging of a reentering spacecraft can provide additional insight into the nonequilibrium chemistry in reentry flows, albeit with reduced resolution. This method was used for the returning sample of the Stardust mission in 2006[8] and several space shuttle missions.[9]

PROJECT OVERVIEW

To provide high-quality reentry flow data, a student team at Purdue University has developed a standardized CubeSat sensor platform that can be combined with spectrometers into a single U (10x10x10cm) of any spacecraft that will be reentering the atmosphere. This platform will provide an unprecedented amount of atmospheric data that can be used to improve atmospheric models and enable a better understanding of the physical processes that occur to satellites, asteroids and other spacecraft encountering an atmosphere.

To validate this platform, Purdue has been selected to participate in a NASA Undergraduate Student Instrument Project (USIP) along with the University of Illinois Urbana-Champaign (UIUC) to design, build and launch the Student Aerothermal Spectrometer Satellite of Illinois and Indiana ($SASSI^2$). This mission will make use of a common 3U spacecraft bus provided by UIUC with a GlobalStar radio to provide constant coverage and allow data transmission during the final hours of the mission. The Purdue Sensor Payload (PSP) will validate the use of a sensor platform capable of measuring the aerothermal heat flux while collecting pressure data in a series of specially designed settling chambers to provide the flow dynamic pressure, velocity, and ambient atmospheric conditions. When combined with the chemical species data from the spectrometers provided by UIUC, the sensor platform will enable the complete characterization of the reentry flow and surrounding atmosphere.

Science Requirements

Atmospheric flows can be categorized based on the Knudsen number given by Equation 1:

$$Kn = \frac{\lambda}{L} \tag{1}$$

where λ is the mean free path, or the average distance traveled by a moving particle between collisions, and L is the characteristic length of the object in the flow. Knudsen numbers less than 0.01 are considered continuum flows, values greater than 10 are considered free molecular, and anything in between is classified as transitional. At an altitude of 200 kilometers, the Knudsen number of the flow around a 1U CubeSat is approximately 5,000. This means that particles interact with the CubeSat several orders of magnitude more frequently than with each other. Therefore, particle to particle collisions can be assumed to be negligible when determining bulk flow properties. In this flow regime, each particle must be modeled separately rather than as a continuous fluid. As particles reflect off the CubeSat and into the oncoming flow, they create a diffuse bow shock with translational temperatures over 20,000 K as seen in Figure 1. However, the low density of the flow results in relatively low amounts of heat flux to the spacecraft. As the CubeSat descends through the atmosphere, the reflected particles begin colliding more frequently with the incoming particles, resulting in chemical reactions in the bow shock. During the Shuttle era, chemical reactions on the surface of the orbiter were a cause of concern due to the additional energy they imparted to the vehicle. However, unlike the exothermic recombination of O_2^+ that occurred on the surface of the Space Shuttle[9], the dissociation of Nitrogen and Oxygen in the bow shock of a reentry vehicle is endothermic and thus decreases the net heat transfer to the spacecraft. Reducing the uncertainty of the rates at which these chemical reactions occur is the primary objective of the SASSI^2 mission.

Modeling in the free molecular flow regime is done with a Direct Simulation Monte Carlo (DSMC) method, which uses a stochastic approach to model flow properties. This approach produces surface properties that are in strong agreement with both theory and lower enthalpy ground testing, but this method has not yet been validated for flow chemistry. For this project, the team used the DSMC solver SPARTA, developed by Sandia National Laboratories, to determine an optimized design for the sensor platform.

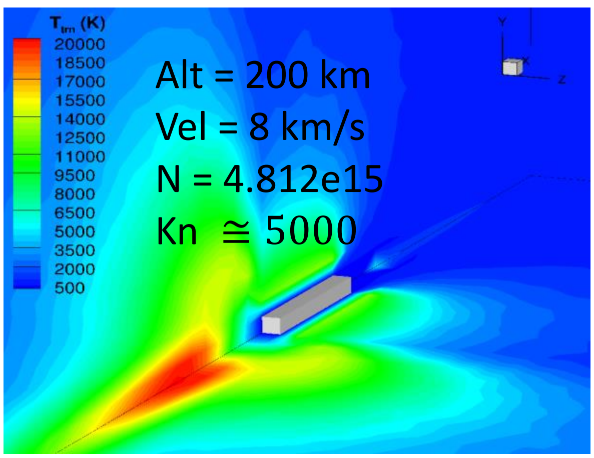

Figure 1: DSMC Simulation of Flow Translational Temperature. Translational temperatures in the bow shock can reach over 20,000 K, however, the low density results in less than 0.5 (W/cm^2) of aerothermal heat flux at 200 km. [10]

The sensor platform requirements shown in Table 1 flow down from the primary objective of determining the chemical reaction rates in the diffuse bow shock. The sensor performance requirements are derived from these science requirements, along with the results from initial simulations using DSMC. To accurately determine the freestream velocity, two or more pressure ports at independent angles to the flow in a manner similar to pitot static tubes are required. An accurate aerodynamic velocity measurement will further reduce the uncertainty in the chemical reaction rates. Additional angled ports enable the determination of other flight parameters, including orientation with respect to the flow.

Table 1: Sensor Platform Requirements

Sensor Platform Requirements	
SP-R1.	The platform shall collect Stagnation properties during atmospheric reentry to determine flow bulk number densities.
SP-R2.	The platform shall determine the freestream velocity of an atmospheric reentry flow.

Concept of Operations

The Concept of Operations for the SASSI^2 mission has been divided into five altitude-dependent phases to maximize the scientific data being collected.

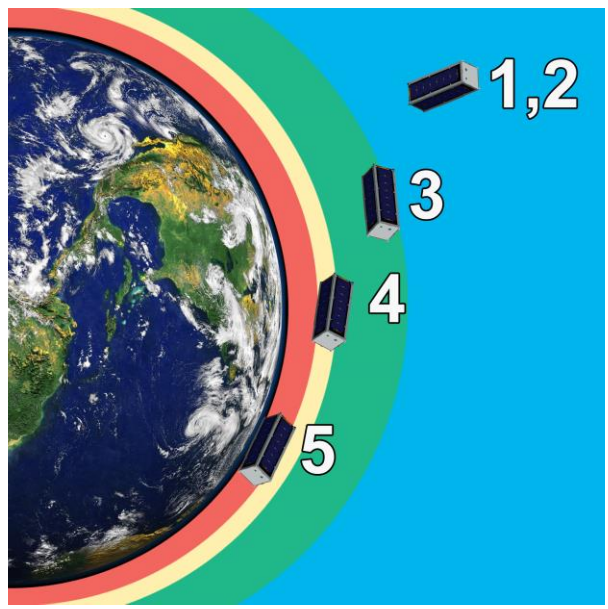

Figure 2: Concept of Operations. After initial checkout and orbit degradation, phases 3-5 collect and transmit science data at increasing rates.

After a 45-minute period of radio silence in accordance with Do-No-Harm requirements, phase 1 will stabilize the CubeSat and begin communications with the ground.

Phase two serves as a subsystem and instrumentation checkout phase to ensure reliable data collection and proper calibration. After initial checkout, the CubeSat will orient itself perpendicular to the velocity vector as seen in Figure 2, maximizing drag and thus minimizing overall mission lifetime. An additional advantage of this orientation is that it will make the CubeSat gravity gradient stabilized until the orbit degrades to an altitude of 200 kilometers.

Phase three is the first science phase of the SASSI[2] mission. This phase is marked by the maximum altitude at which our sensors will measure meaningful data. During this phase, the CubeSat will reorient to the ram direction, placing the spacecraft body x direction along the velocity vector. Spectral data collection will occur during eclipse using a visible light calibrated spectrometer, while the PSP runs on a duty cycle determined by the bus to remain power positive. Phase three runs until the craft reaches an altitude of 150 km.

Phase four continues science data collection. However, in this phase, ultraviolet spectral data is collected during eclipse using one of the two UV calibrated spectrometers. PSP data collection remains unchanged for this phase. Phase four is completed once the CubeSat reaches an altitude of approximately 130 km.

Phase five is the final science phase and concludes the mission. After phase four, the CubeSat will cease to duty cycle the PSP and will run both UV calibrated spectrometers on board to collect as much data as possible through the thickest atmosphere that the spacecraft will survive.

SENSOR PAYLOAD

To meet mission objectives, the PSP must measure atmospheric pressure and temperature data, as well as interface with the Illinisat Bus. To do this, the PSP team has developed a system consisting of three pressure ports with inlets at independent angles to the flow feeding to independent settling chambers with Pirani gauges. A heat flux sensor to determine flow temperature and an avionics system for command and data handling complete the sensor platform. The PSP was designed to fit into a 2U payload space alongside three spectrometers and a GlobalStar radio provided by the University of Illinois. The 2U payload space is shown in Figure 3.

The PSP subsystem development is divided into two main groups, the Science Payload and the Payload Support System. The Science Payload is comprised of the pressure sensor suite and the heat flux sensor assembly. The Payload Support System consists of the avionics system, the thermal management system, and the hardware mounting systems. These systems are designed to ensure the nominal operation of the Science payload, and proper communication and power distribution with respect to the rest of the SASSI[2] bus. A detailed description of the development and state of the PSP Science Payload follows.

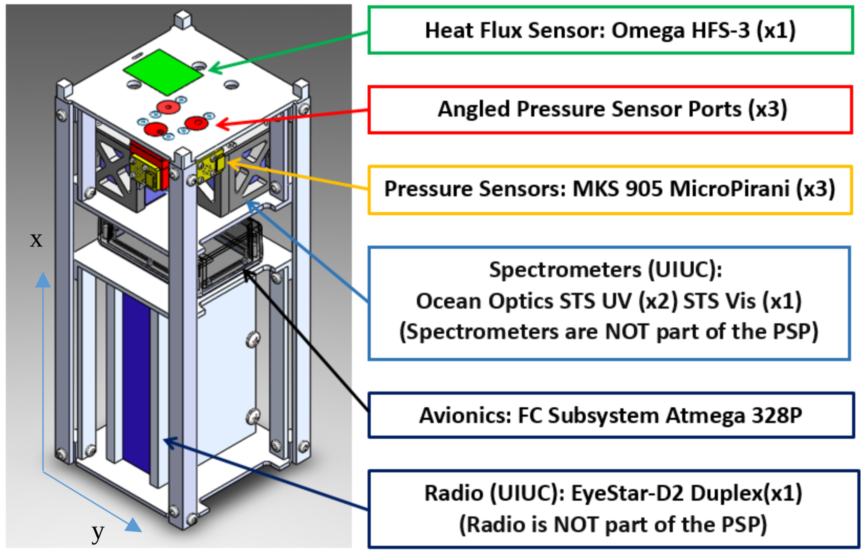

Figure 3: SASSI[2] mission 2U payload space. All sensors fit in the front ¾U leaving the remaining payload space for avionics and the GlobalStar Radio.

PRESSURE SENSOR SUITE

Mission Requirements

The primary system-level science requirements of collecting flow stagnation properties during atmospheric reentry and determining freestream velocity require that pressure data be obtained during flight. Orbital velocity is not sufficient for this measurement due to the over 100 m/s changes in wind speeds that can occur throughout a single day and alter the freestream conditions.[11]

Free stream velocity is calculated using the difference in pressure between the three pressure ports configured in a pseudo-pitot-static probe. Unlike a conventional pitot-static probe in continuum flow, the pressure ports' angle of incidence with the surface in free molecular flow will cause some particles to bounce out of the port through diffuse collisions, thus lowering the pressure in the port settling chamber.

On the satellite, the three pressure ports are configured with one orthogonal to the ram face, referred to as the stagnation port, and two ports angled at 20 and 30 degrees offset from the stagnation port in perpendicular planes. In the angled ports, the particles collide diffusely with the inlet port walls, causing some to exit the port before entering the settling chamber. This occurs in a manner that produces a predictable pressure drop according to the inlet angle. The stagnation port allows the particles to enter a settling chamber before colliding. This creates a difference in the steady state number density and pressure between each settling chamber. This pressure differential can then be used to calculate the flow velocity, angle of incidence, and density.

Due to the free molecular nature of the flow, the sensors must have ports on the ram face of the satellite to collect measurable data. Because the free-molecular conditions affect the entirety of the front face similarly, the only additional location requirement is that the port inlets be placed at least ten port diameters away from any protrusions on the ram face. This requirement prevents interactions with parts of the flow that are not representative of freestream conditions.

Approximate knowledge of the pressures encountered across the mission flight regime was necessary before the sensor selection process could begin. Molecular flux relations[10] were used in conjunction with a DSMC solver called SPARTA[12] to determine bounds on the expected stagnation pressures. These pressure approximations have some uncertainty due to the unknown reaction rates of high enthalpy flows, though the variance of the pressure due to the uncertainty remains within the same order of magnitude. The stagnation pressure of 7.7 mPa at an altitude of 200 km and a velocity of 7 km/s, and 322 mPa at 130 km altitude and 7 km/s were used to develop the requirements of the science mission. To fulfill these requirements, the pressure sensor must detect changes in flow stagnation pressure from 10 mPa to 420 mPa with a resolution of 1 mPa or smaller.

Sensor Selection

There are four commercially available types of sensors capable of measuring vacuum pressures within the mission range: Pirani gauges, capacitive gauges, hot-cathode gauges, and cold-cathode gauges. Pirani gauges are the most viable option for this mission, primarily due to their low power requirement, but also because their Micro-Electro-Mechanical System (MEMS) package provides for a small size and low mass.

Sensor Performance

The MKS Instruments 905 MicroPirani meets both the minimum requirements and desired performance, reading the lowest required pressure of 1 mPa up to 100 kPa. The sensor also has a resolution of 1 mPa, which meets the minimum requirement.

While two pressure ports with inlets at independent angles to the flow are required in to determine the free stream velocity, it was decided to add a third port to the sensor platform in order to determine angle of attack. This will allow for higher fidelity velocity information from the pressure data. Without this third pressure port, any deviations from the velocity vector would produce lower pressure readings than would otherwise be recorded at a given altitude and velocity.

Using three ports provides greater velocity accuracy, as well as adds a level of redundancy to the system. A fourth port was considered to provide yaw information, however it was abandoned due to limited space on the ram face of the CubeSat. Any follow-on missions will likely revisit this decision to enable full characterization of the freestream velocity vector.

Satisfy Sensor Requirements

Each of the three settling chambers is equipped with a MicroPirani. Each sensor requires 45 mA of current at 5 Volts DC, or approximately 0.225 Watts per sensor. The spacecraft bus provided for the mission primarily uses a 3.3-Volt rail to power electronics. Therefore, a switching converter was required to power and communicate with the MicroPirani. Data is transferred digitally through a multiplexer via a 5 Volt TTL UART connection to the payload avionics unit. This allows the unit to communicate with all three sensors and the bus at the correct operating voltage using a single pin.

The MicroPiranis have a narrow operating temperature and therefore cannot be exposed directly to the high temperature flow. The sensors work by heating a small nickel filament to 15 K greater than the surrounding sensor. As heat is conducted away from the sensor through particle collisions, power is required to maintain the temperature difference. A higher number of collisions requires more power, corresponding to a higher pressure. Knowledge of the molecular mass of the chemical species in the gas being measured allows the chamber pressure to be uniquely determined. If the filament were directly exposed to the high-energy particles in the flow, the particles would impart their energy to the filament rather than take energy away, and render any sensor outputs invalid.

A settling chamber with a narrow inlet prevents high-energy flow from hitting the sensor directly. Further, the sensor is placed on the side of the chamber to ensure that particles bounce on a cold surface before interacting with the sensor. Particles colliding with the wall of the settling chamber are scattered diffusely, transferring energy to the wall and reducing the gas temperature. The settling chamber design was optimized using SPARTA, with plans to complete low-velocity tests for validation.

This reduction in temperature comes at a cost; pressure in the settling chamber is predicted to drop about one order of magnitude relative to the freestream. This lowers the minimum pressure the sensor must read to 1 mPa, which is on the same order of magnitude as the sensor's resolution. While cooling the flow makes it measurable, the measurement becomes less accurate at the highest altitudes. Fortunately, as the satellite decreases in altitude and gains velocity, the stagnation pressure increases exponentially and accuracy improves quickly during the science mission.

Despite the 56% concentration of monatomic oxygen predicted by SPARTA and the presence of a silicon substrate in the sensor, the sensor is expected to operate without corroding. This is because silicon dioxide, a non-reactive material, is used as a protective layer over the critical nickel filament, and the gold wire is also non-reactive with monatomic oxygen. Figure 4 shows the layout of the sensors, along with the materials used.

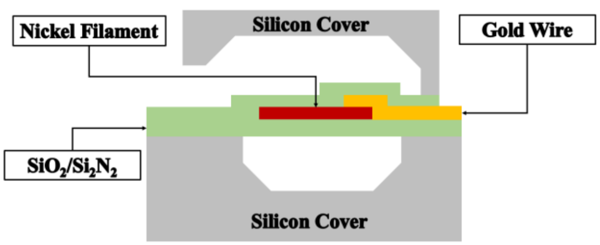

Figure 4: MKS 905 MicroPirani Sensor Detail. The colors show the layout of the materials within the sensor and assist in determining the sensor performance.

Testing

The primary goal of testing was to develop a sensor output model for given flight conditions. This model provides a correction factor for variations in voltage, species, boot times, and duty cycling.

Voltage Drift

During flight, power voltage conversion may fluctuate. To characterize this effect on the sensor output, the sensor was put under varying voltage conditions in the lab. The MKS905 MicroPirani was tested between the voltages of 4.3 V and 5.1 V while at a pressure of approximately 10 Pa. The sensor ceased to give readings below 4.6 V at 50 mA of current, and continued to function up to 5.1 V at 50 mA of current. Voltages above this level were not tested to prevent damage to the sensors.

There was no statistically significant change in the pressure reading as the voltage was adjusted from the upper to the lower limit. Pressure readings continued to be consistent until stopping completely when the voltage became too low.

Species Dependence Testing

The sensor determines pressure based on molecular collisions. The pressure reading is therefore dependent on the chemical species being measured. The sensor comes with settings to determine the pressure of N_2, O_2, H_2O, standard air, Ar, He, and H_2. However, after the gas setting is specified, any change in the chemical species interacting with the sensor will cause the sensor reading to change independently of the true pressure. Chemical species information provided by the onboard spectrometers will be used alongside DSMC simulations and empirical data to determine the true pressures inside the settling chambers.

The species in the upper atmosphere vary widely from those present at sea level. To account for the sensors' species dependence, the sensor has been calibrated for helium, argon, and nitrogen while set to measure nitrogen. Using these variations in species data, an empirical fit was developed with the aid of Gambosi[10], and Jousten[13]. This fit distinguished $\frac{M}{M_{ref}}$ and $\frac{\gamma+1}{\gamma-1} * \frac{\gamma_{ref}-1}{\gamma_{ref}+1}$, as meaningful non-dimensional values that isolate species-dependent pressure behavior, where M is molecular mass, γ is the ratio of specific heats, and the subscript "ref" denotes the value the sensor is set to read. With these non-dimensional parameters identified, a fit was developed using a planar logarithmic regression.

$$\frac{P_{actual}}{P_{mks}} = \left(\frac{M}{M_{ref}}\right)^{0.5} * \left(\frac{\gamma+1}{\gamma-1} * \frac{\gamma_{ref}-1}{\gamma_{ref}+1}\right)^{-1} \quad (2)$$

Theoretically, the fit should conform to Equation 2, but because of factors unique to the sensor, such as the shape and combination coefficient, the fit is skewed. Ultimately, the empirical fit shown in Equation 3 will be used instead of the theoretical fit because it was developed using real-world sensor data.

$$\frac{P_{actual}}{P_{mks}} = 1.01 * \left(\frac{M}{M_{ref}}\right)^{0.0849} * \left(\frac{\gamma+1}{\gamma-1} * \frac{\gamma_{ref}-1}{\gamma_{ref}+1}\right)^{-0.9251} \quad (3)$$

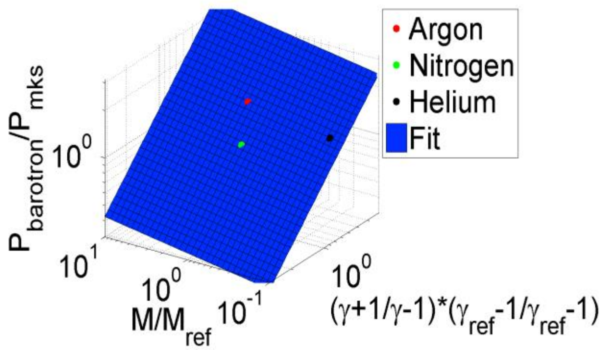

Figure 5: Empirical Fit for Species Dependence. Empirical fit determined from minimum number of data points requiring additional species testing to evaluate fit quality.

The planar fit for the domain was determined from three data points, resulting in data that is perfectly constrained to the domain as seen in Figure 5. More testing is required to explore the quality of the fit. This will be done through varying M_{ref} and γ_{ref} by setting the MicroPirani to measure different gases. Once this testing is complete, the final empirical fit will be used to determine the true pressure of the gas mixture in the settling chambers.

The measurements to determine the empirical fit occurred between 10 Pa and 1 mPa, however the Barotron sensor used as a reference pressure became prone to error below about 100 mPa. This data was therefore discarded during analysis. The MicroPirani exhibited a highly linear correlation with a variance of less than 0.01 on a log-log scale between the sensor readings and correct pressure. The empirical fit developed in the measured pressure range can be extrapolated for the entirety of the MicroPirani measurement range because this linear trend continues over many orders of magnitude.

Boot Up Cycle Testing

During flight, the MicroPirani sensors will be periodically switched on and off to save power. Since the sensors will be cycled over the course of the mission, it was important to understand any lag time between when the sensor is supplied power and when it first provides valid data.

During testing, it was found that the maximum boot-up response time was 1.29 seconds. This means that for the duty cycle for the pressure sensors must remain on at least 1.3 seconds in order to take pressure measurements. As seen in Figure 6, there is no pressure ramp as the sensor cycles on and off. The sensor tested was in a nitrogen environment and read the same pressure for each of the cycles tested. Placing the sensor in a duty cycle was found to have no effect on pressure measurement.

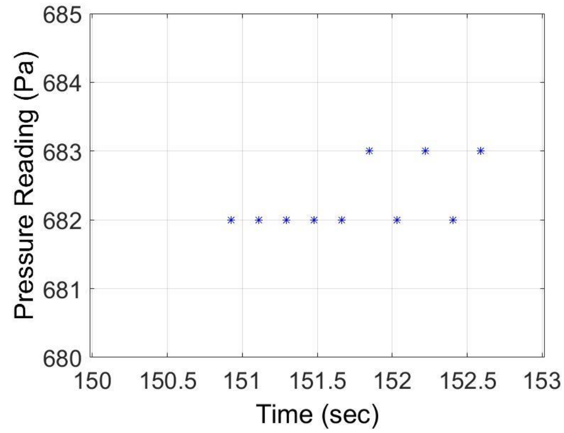

Figure 6: Typical Sensor Boot Cycle. Over 10 cycles, the sensor took a maximum of 1.29 seconds to begin collecting data once commanded and shows no ramp response during boot-up.

Remarks

The entire pressure sensor suite has a mass of just over 100 grams and a volume of less than 30 cm^3 while drawing only 0.75 watts. This compact design, low mass, and low power consumption makes it capable of being added to almost any mission, especially due to the fact that the sensors would not draw any power until the spacecraft began to reenter the atmosphere, at which point almost all other primary missions would have ended. Additionally, the off-the-shelf sensors and settling chambers that can be made by any local machine shop provide a cost-effective means of collecting valuable atmospheric reentry data.

HEAT FLUX SENSORS

Mission Requirements

To meet the system and component level requirements of the mission, the temperature sensor suite must detect changes in flow temperature below 200 km. Determining flow temperature is not a trivial task, as there is no way to directly measure flow temperature due to the low density and high enthalpy of the flow. Instead, the temperature sensor suite must monitor the heat flux applied to the ram face of the CubeSat and convert that to flow temperature. Preliminary DSMC simulations in SPARTA showed that at altitudes between 200 and 100 km, the flow will impart between 35 and 100,000 Watts per square meter (W/m^2) respectively, to the ram face. The sensor will monitor the change in this applied energy with a minimum resolution of 3333 W/m^2 (equivalent to 5 K) for the duration of the science mission.

Determination of Baseline Sensor

To monitor the heat flux applied to the CubeSat, two different approaches were examined. The first involved developing the relationship between the temperature of a thermocouple and the heat flux applied to the sensor area. This approach required the development of a calibration curve, and sensor temperature was reliant on more factors than the heat flux applied. This resulted in levels of error unacceptable for the final mission. The second approach used a dedicated heat flux sensor that was found to meet all performance requirements.

Sensor Performance

The heat flux sensor selected for the mission was the Omega HFS-3. The specifications of this sensor are listed in Table 2.

Table 2: HFS-3 Performance

Sensor	Nominal Sensitivity $(\frac{\mu V}{W/m^2})$	Max Heat Flux $(\frac{W}{m^2})$	Response Time (sec)	Thickness (mm)
Omega HFS-3	0.951	94638	0.6	0.18

This sensor was selected because it meets the system requirements, is commercial off-the-shelf, is vacuum rated, and is currently used to measure heat flux ranges like those expected in the mission. One requirement that the sensor is not capable of meeting is measuring the maximum heat flux applied to the CubeSat. However, DSMC simulations show that the CubeSat will encounter the sensor's maximum heat flux at an altitude of 100.45 km, only 450 meters from the assumed communications blackout altitude.

The HFS-3 sensor consists of a thermopile with 54 junctions suspended inside a thin layer of Kapton polyimide film. This film is extremely thin, allowing for efficient energy transfer through the junctions and into the surface of the CubeSat where it is mounted. As energy from the flow passes through the sensor, a temperature gradient is created, causing the thermopile junctions to produce a voltage. A heat flux can then be determined by monitoring the voltage output and converting it using the sensor's nominal sensitivity. This heat flux value is then input into the DSMC models along with the corresponding altitude and velocity models to determine the flow temperature.

Driven Requirements of Support Systems

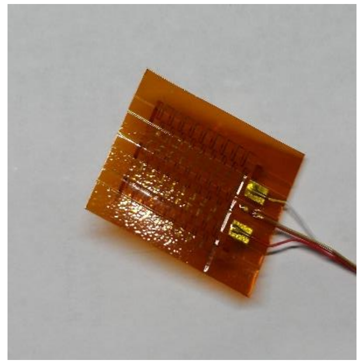

Figure 7: HFS-3 is a commercial off the shelf heat flux sensor used in vacuum freeze-dryers [14]

For data collection, the sensor's voltage must be converted from an analog to a digital signal. This is done by interfacing directly with the Analog to Digital Converter (ADC) built into the avionics board. Due to the low voltage output of the sensor, it is necessary to amplify the voltage as it enters the ADC.

During the flight, it is necessary to use epoxy between the sensor and the front plate to ensure proper thermal contact between the sensor and the CubeSat's ram face. While the sensor is rated for temperatures ranging from -200 $°C$ to 150 $°C$, the epoxies used to mount the sensor are only rated for -55 $°C$ to 250 $°C$. Based on initial thermal modelling of the CubeSat, the low end operating temperature will not be reached, so thermal management solutions are unnecessary.

Heat Flux Sensor Calibration Test

Due to the mission's wide range of expected heat flux values, it is essential to validate and, if necessary, calibrate the original equipment manufacturer (OEM) provided models relating the sensor output voltage to heat flux for the entire range of expected heat flux values. Doing so ensures that the data being collected during the mission is accurate and representative of the heat flux being applied to the CubeSat. The sensor calibration was performed using a method similar to industrial freeze drier heat flux sensor calibration tests.

A maximum applied temperature differential of 68$°C$ was determined from plugging in the values of heat flux calculated using DSMC, a γ of 0.12 W/(°C *m), and the thickness of the sensor into Equation 4

$$\dot{q} = \frac{\gamma * \Delta T}{t} \tag{4}$$

where $\dot{q}$ is the heat flux through the material, γ is the thermal conductivity of the material, ΔT is the temperature difference between the hot and cold sides of the material, and t is the material's thickness. Subjecting the sensor to these known ΔT values and comparing the resulting heat flux to the theoretical heat flux shows whether the sensor requires calibration.

The setup, shown in Figure 8, includes: a cold block/sink

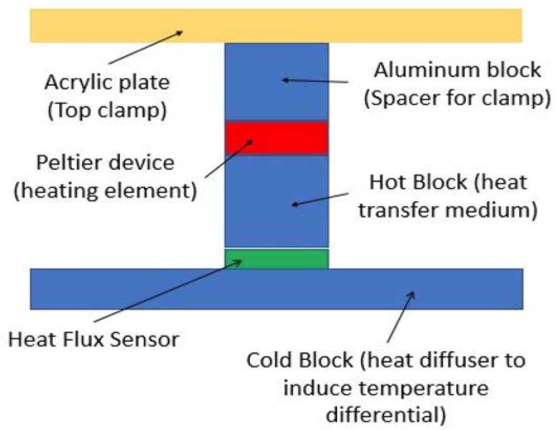

Figure 8: Heat Flux Sensor Calibration Test Setup. Clamping the sensor between a hot and cold plate allow the heat flux measured by the sensor to be calibrated against the applied heat flux.

used to induce a temperature differential, the heat flux sensor, a hot block used to provide a medium for heat transfer, a Peltier device as a heating element, a second aluminum block for spacing, and an acrylic plate to clamp the setup together.

When the heat flux from the flow is combined with the flow pressure, the flow temperature can be determined. The flow temperature is the last key piece that must fall into place before the reaction rates in the diffuse bow shock can be determined. The off-the-shelf sensor selected provides a robust method for measuring heat flux in low-density environments that has been repeatedly used in industry.

CONCLUSION

Thermal protection systems on spacecraft do not contribute to mission objectives. They do not aid in research or provide services that make going to space worthwhile. Yet these systems account for a significant portion of a reentry vehicle's mass and cost, and they are a vital piece of any mission that is designed to plunge into an atmosphere. Students at Purdue University have developed a standardized CubeSat sensor platform that can be applied to almost any mission and provide crucial data that will allow increased confidence and reduced margins in thermal protection systems. This platform has been selected by NASA to fly a demonstration mission on a common CubeSat bus built by the University of Illinois. The Student Aerothermal Spectrometer Satellite of Illinois and Indiana will provide the flight experience needed to demonstrate the sensor payload as a cost-effective approach for collecting the atmospheric reentry data needed to validate reentry models and reduce TPS mass and cost.

Acknowledgements

We would like to thank David Wilcox and NASA Undergraduate Student Instrument Project agreement NNX16AK73A for providing funding to pursue this project, as well as the Purdue School of Aeronautics and Astronautics for providing the lab space and much of the equipment we have used. We would like to thank the Purdue Aerospace Sciences Laboratory for their excellent workmanship and punctuality as well as T.N. Thompson and Qiming Wang of Millrock Technologies, Inc. for providing advice on heat flux sensors and sensor calibration procedures. We would also like to thank C&R Technologies for providing a student license of their Thermal Desktop software, and Advanced Circuits, Inc. for their generous printed circuit board contributions to the project.

We want to especially thank the entire Purdue undergraduate team on SASSI[2] project of whom we could not be prouder and who have made working on this project thoroughly enjoyable. We also want to acknowledge and thank Prof. Zach Putnam, Prof. Deborah Levin, Prof. Alexander Gosh, Kaushik Ponapalli, Nick Zuicker and the rest of the UIUC team for everything they have done and continue to do for this project. We are incredibly grateful to Devon Parkos, our graduate student mentor, who has proved invaluable to this project every step of the way. Further, we would like to acknowledge Dr. Anthony Cofer for lending his plethora of vacuum testing and spacecraft technology knowledge to the project. We would like to thank Andrew Strongrich and Prof. Alina Alexeenko for the original concept of a pressure-based freestream sensor. Finally, we thank Prof. Alexeenko and Prof. David Spencer for their continued efforts of mentoring the project team at Purdue and without whom this mission would not be possible.

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