2019 Spaceport America Cup Conference

Podium Presentations

Design, Analysis and Testing of a Passive Ramjet Inertial Stabilization Mechanism

Team 01 Project Technical Presentation to the 2019 SA cup

Jeremey Waldron and Thomas Susi
The George Washington University, Washington, DC, 20052

INTRODUCTION

The Passive Ramjet Inertial Stabilization Mechanism, henceforth referred to as PRISM, is an aerodynamically driven gyroscope which aims to counter the natural pitching and yawing moments of a sounding rocket. The concept was originally developed for the Intercollegiate Rocket Engineering Competition, and strict requirements had to be met in order for PRISM to be accepted. PRISM must: integrate into a standard six inch diameter rocket, be a passive system (i.e. not able to be controlled after launch), limit the pitch and yaw of a rocket to within ten degrees, not add significant drag to a standard rocket, and must withstand the physical forces of launch and landing. Based on these requirements, the following design was conceived. It is made up of a forward hardpoint to direct the airflow through the nose cone to outlet holes, a turbine fan blade mounted on a steel shaft assembly, and an electronics bay located in a standard six inch rocket coupler. During flight, airflow is naturally forced through the hollow nose cone, the turbine blade extracts energy from that airflow, and the turbine blade-shaft assembly spins as a result. The spinning shaft provides the torque, or rotational force, to counter the pitch/yaw of a rocket through a phenomenon known as gyroscopic precession. After conducting a ground spin test, a flight test was then carried out with PRISM mounted on a standard six inch rocket; the rocket flew at about 400 miles per hour and PRISM spun up to 8,000 revolutions per minute, and the preliminary data shows that the rocket hovered at zero degrees in both the pitch and yaw directions. The results are promising for this pioneering system.

DESIGN PROCESS

PRISM was designed with a few goals in mind outside of the requirements previously laid out. It was to provide maximum airflow to the turbine assembly, maximize torque produced, minimize frictional losses, and minimize instabilities in drag and pressure. It consists of a commercial-of-the-shelf nosecone with its own nose cut off to allow airflow in. A forward hardpoint was created to both house the shaft system and direct airflow into the nosecone. A shaft with a turbine fan sits internally to this nosecone on two bearings, one at the hardpoint and one at the couplers's top bulkplate. This shaft is free to spin as air enters the nosecone and runs through the turbine blade. The turbine itself is positioned on the shaft directly beneath the inlet area to ensure fluid flow passes through it. The air is then directed by an internal cone to four outlet slots in the fiberglass nosecone. Additionally, a data collection system sits in the PRISM coupler to collect acceleration, pressure, and rotation data. The data collection system has access to the end of the shaft assembly and uses an infrared tachometer to collect rotation rate data. This design was settled on after months of iterative design and digital prototyping.

The system got its start with sizing a large commercial electric ducted fan to then develop a flow path around. Once we settled on a 90-mm aluminum fan, we moved to preliminary internal flow design, focusing on avoiding choked flow. The forward hardpoint design was key to both avoiding choked flow and excess drag given shocks forming. We designed the hardpoint to slot directly onto a commercial nosecone that had been cut open at the top. It allows flow to enter the nose of the rocket and houses one of the two bearings from the shaft assembly. The inlet area of this hardpoint was then scaled up for the exit ports on lower on the sides of the nosecone. This increase in size by a factor of 1.5 was to guarantee flow could not choke as it flowed through the nosecone. Then the internal area was reduced and optimized to allow flow to exit without large pockets of stagnant air forming. This was done with an internal cone directing air from the turbine blade out of the rocket. Finally, the shaft was sized to both add significant weight and allow for the infrared tachometer to take data from inside the coupler. The entire process focused on machinability and ease of assembly while not compromising our goals stated previously.

ANALYSIS AND TESTING

Analysis on PRISM was done continuously from the project's conception, all the way through its first flight. First the concept of gyroscopic precession was researched. This process is the result of a torque forming along an axis as a mass has its rotation rate accelerated. This creates a direct counter moment to the natural motion of the rocket.[1] The fundamentals of gyroscopic precession have been previously used in control moment gyroscopes (CMG). These rotate a mass, generally using an electric motor, to provide a direct torque on the system via the CMG support structure to orient the rocket in a desired direction. Douglas Havenhill patented a direct torque CMG to be used in satellites, and research into his design heavily influenced the initial design of PRISM.[2]

The system was put through a multitude of CFD analysis tests before manufacturing to optimize and check the feasibility of the system. This focused on finding discrepancies in pressure around static ports and massive shifts in the center of pressure that could cause instability. This analysis was done with SolidWorks Flow Simulations coupled with Openrocket simulations to validate this testing. A typical ogive nosecone was analyzed in SolidWorks and the data received from this process was validated with Openrocket to start. The data collected found the location of the center of pressure in the nosecone and the drag produced. Then PRISM was run through the same simulation and the same data was collected. By comparing this data we could analyze how our rocket would react to this new system. The findings showed a 65% increase in drag with PRISM and the center of pressure shifting 2-in towards the nose. The drag increase was large but something we expected and could work around. The center of pressure location ended up making our rocket more stable overall. Both these results however were only for the nosecones alone, not the whole rocket airframe, so in the end these findings would be negligible.

After this research and analysis the system was manufactured. It was first spun-up on the ground using compressed shop air directly blown into the inlet area. This was to test the system spun freely and data could be collected. This was a success as the system logged rotation rates up to 3,000-RPM. Then PRISM was flown on a test flight. It flew on a Aerotech M1500 to about 4500-ft at speeds reaching 400-mph. The results data showed PRISM spinning up to 8,000-RPM and rotations for the rockets pitch and yaw hovered around 0º during ascent, meaning the rocket flew straight. The system was recovered successfully with no significant damage to the hardware or electronics. Post-processing this data showed an average torque of about 70-Nm being produced by the system during flight. This is promising data as the system will be pushed further at IREC when it is launched to 10,000-ft.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Overall, the first iteration of the Passive Ramjet Inertial Stabilization Mechanism accomplished all of its functional requirements and initial test results look promising for IREC and beyond. The first test flight with PRISM achieved a stable, straight low altitude flight. Upon further inspection, some strengths and weaknesses appear obvious. PRISM was easy to manufacture, was not expensive to produce, is easily integrated into any six inch rocket (and can be modified to fit into any rocket), and is a mechanically simple system. On the other hand, PRISM does produce excess drag effects with its current inlet geometry, is not ideally optimized to produce the maximum amount of torque possible, and is difficult to disassemble completely. This being said, we expect PRISM to perform well at IREC. It will encounter higher speeds, forces, and altitudes in this flight but should not only withstand these forces, but work even more effectively with more airflow passing through the nosecone.

With careful optimization this system would prove particularly useful for finless highspeed rockets, where static stabilization effects (i.e. fins) are minimal. Minor improvements to the inlet and outlet sections could allow the system to operate at high speeds and minimize the shock effects encountered. Using internal gyroscopic stabilization, the aerostructure would only require minimal control surfaces to maintain stability. Coupled with an aerospike, high-speed flow would be sonically decelerated and then enter a turbine PRISM system. Similar flow turbine designs could be used to power aerodynamic driven pump systems on hybrid and liquid rockets. By mitigating the need for rocket spin stabilization, PRISM allows rockets to act more flexibly for hosting payloads with specific observation or orientation-keeping requirements. Traditionally low altitude sounding rockets use spin stabilization to maintain a straight flight. However, spin is not always compatible with sensitive or optical payloads which require steady flight and or a specific flight orientation. This is where our system could be implemented.

REFERENCES

[1] Keshtkar, Sajjad; Moreno, Jaime A; Kojima, Hirohisa; Uchiyama, Kenji; Nohmi, Masahiro; Takaya, Kesiuke. "Spherical Gyroscopic Moment Stabilizer for Attitude Control of Microsatellites". Acta Austronautica Volume 143. Feb 2018: 9-15. ScienceDirect, 5 Oct. 2018.
[2] VEEM Ltd. "How Gyrostabilizers Work." VEEM Gyro , 20 Jan. 2019.

Paraformaldehyde/Epoxy Resin Hybrid Rocket Engines

Team 02 Project Technical Presentation to the 2019 SA Cup

Agata Zwolak and Bartosz Wyciszkiewicz
AGH University of Science and Technology, Cracow 30-059, Poland

INTRODUCTION

AGH Space Systems is a student scientific association, which was established in 2014 at AGH University of Science and Technology in Cracow, Poland. The members work on space technologies in a broad sense, among others experimenytal sounding rockets. The hybrid rocket engines have been investigated since the begining of AGH Space Systems. Motivation of the team was to develop their own fuels beacuse the team always try to do everything themselves and to gain skills which would help in future rockets designing. All previous hybrid-propellant rockets designed by AGH Space Systems used low regression rate fuels, because of simplicity of manufacturing and safety, but high regression rate fuels are characterized by higher generated thrust and lower consuption of oxidizer, what allows to minimize the mass of the propellants. AGH Space Systems members have carried out research on new types of fuels, which could be applied in next rocket engines.

RESEARCH OVERVIEW

Many hybrid fuels both high and low regression rate were checked, such as polypropylene, polyoxymethylene, polyamide, epoxy resin and waxes. Additives like carbon black, aluminum powder or various oxidizers were examined to improve combustion stability and efficiency. This year, idea was born to create fuels based on epoxy resin matrix and various concentration of paraformaldehyde (PF). There was a prediction that PF would increase fuel regression rate and decrease optimal O/F ratio. The PF temperature of decompossition is much lower than epoxy matrix so fuel's burning surface should be extended due to porosity which expected to be constant during engine work. High concentration of oxygen should be reducing agent for optimal O/F ratio. Samples of 10, 20, 30, 40 and 50% of PF by mass were tested in special small scale rocket engine which was designed and constructed for this project. Hot-flow tests were made in small scale due to logistics and economic considerations. Gaseous nitrous oxide was chosen as an oxidizer. Thanks to the broad range of tasks electronics systems are capable of the team was able to use them during hot-flow tests. Variety of samples based on epoxy matrix with different concentration of paraformaldehyde were made and combused in several values of oxidizer mass flux rate to gain fuel regression rate equation. Performance parameters like C-star, specific impulse, combustion chamber pressure were also measured to select the best composition.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

The results were compared with well-known fuels such as HDPE and nylon. Moreover the samples were subjected to thermal and chemical analysis, also hardness was measured just to make sure that effect of PF addition will not be negative for mechanical durability. The most promising sample was chosen and will be enlarged to test the effect of scale. First step is to test the best paraformaldehyde/epoxy resin fuel in B3 engine (which propelled Panda3 rocket last year). It allows to both check the performance of this motor with new fuel and test, if it is possible to manufacture this fuel in a larger scale. The results of this work will be use in preparation of actual and next projects such as Prototype's egine B4.

Analysis of Aerodynamic Heating of High Powered Rocket Nose Cone

Team 08 Project Technical Presentation to the 2019 SA Cup

Archie Scott III and Eddie Mikalsen
Ohio University, Athens, OH, 45701

INTRODUCTION

For the 2019 Intercollegiate Rocket Engineering Competition (IREC) the Ohio University Rocket Design and Engineering Team will be launching project The Russ Space Buss. This project will be competing in the 10,000 foot altitude category and will be fitted with a sensor to measure the temperature increase of the nose cone tip during flight. This experiment was decided upon in order to gather information for future projects and investigation of alternative nose cone materials.

EXPERIMENTAL DESIGN

In order to collect data pertaining to the aerodynamic heating of the nose cone tip, an resistive thermal detector (RTD) will be placed in the nose cone. By centering this sensor at the rockets forward most leading edge, the team intends to gather information on the increase of temperature due to aerodynamic drag and skin friction. The RTD used will be the PT-100 and accompanying amplifier purchased from Adafruit. These two devices will be connected to a Teensy 3.2 microcontroller with a micro SD card for data storage. Successful launch and recovery of the Russ Space Buss will provide the first sample of data for processing. Following the competition, this system can be integrated into future projects to gather multiple samples of data.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

After the sensors, amplifier, and computer program written for this competition have been verified, future recommendations would be to increase the number of RTDs used on the vehicle. Additional RTDs could be used to measure additional points of interest within and on the rocket. Such as the fin leading edges, surfaces of the airframe and internal points such as temperature increase within the motor tube.

Estimation & Optimization of an Airbrake Lookup Control Scheme Through Rocket Performance Simulations

Team 17 Project Technical Presentation to the 2019 SA Cup

Paul Buzuloiu, Manit A Ginoya and Andrew Zavorotny
University of Ottawa Rocketry (uORocketry), Ottawa, Ontario, Canada, K1N 6N5

INTRODUCTION

In the previous year, uORocketry presented a SRAD, transversely actuating airbrake mechanism along with a model predictive control (MPC) scheme for the purposes of achieving a chronically accurate apogee. Upon the realization that limited computation capabilities would impede the effectiveness of such a scheme, alternative solutions were determined; the most appealing of which is the underlying topic of this extended abstract: development of an in-house flight dynamics and rocket performance simulation model, enabling the determination of non-obvious relations between performance metrics, software-in-the-loop control algorithm testing and, in the future, experimentation with expansion on currently known analytical models of aerodynamics, electrical systems, thermal systems and propulsion methods.

Through the use of this tool it became possible to simplify the airbrake control scheme from MPC to a lookup table method; a relation of current altitude, current apparent velocity corresponding to an ideal airbrake deployment percentage was determined. This relation was implemented in a software-in-the-loop model of the avionics and simulated to observe the performance of flight under multiple wind speed conditions and launch angles. By performing this study, the team achieved both an implementable control solution to use for the airbrake system, as well as an opportunity to test and demonstrate the effectiveness of the developed performance modeling and simulation tool.

AIRBRAKE MECHANICAL DESIGN & CFD RESULTS

The optimized mechanical design was almost entirely carried over, except for the internal changes that made the assembly easier to integrate into the rocket body. The summary of the mechanical design is briefly presented here for convenience: The airbrake system is positioned directly above the motor mount. Its drag inducing surfaces include three identical aluminum leaves that deploy perpendicular to the relative airflow . When fully deployed, a surface area of 36.8-cm^2 per leaf (110.4-cm^2 in total) is introduced. This results in a range of drag from 219.9-N down to 32.6-N, corresponding to speeds between 250-m/s and 100-m/s respectively. The thickness of each leaf is 3mm which requires a slot to be cut into the rocket body. All three leaves are actuated simultaneously by a single 5.8-kg×cm, metal geared servo motor in order to prevent asymmetric deployment. The servo is connected directly to the SRAD avionics system.

FEA analysis was conducted to ensure the structural integrity of the rocket body with the addition of the leaf gaps. The drag characteristics presented above were determined by completing a set of CFD computations using ANSYS ® Fluent. These drag characteristics were implemented in the performance model and simulation suite developed by the team to perform the airbrake control scheme study.

AIRBRAKE CONTROL SCHEME STUDY

The reasoning and methodology for the airbrake control scheme are presented here: The airbrakes are meant to deploy only during the coast phase of flight. As such, there is a coupling between altitude and apparent velocity during the control loop. Following this line of reasoning, there must be some relation between current apparent velocity, current altitude and airbrake deployment percentage that results in an exact apogee of 3048-m (10000-ft).

To find this relation, the custom rocket performance model was simulated by sweeping ranges of constant commanded airbrake percentages and constant apparent velocities for which to deploy at. More specifically, the range of airbrake percentages was from 0.1 to 1.0 (normalized to 1) and the range of deployment velocities was from 50-m/s to 250-m/s. The data collected from these simulation runs were filtered such that only those that resulted in an apogee within the error threshold of 10-m were kept. By comparing and minimizing the apogee error of each filtered run for the specified ranges, the final relation between current altitude, current velocity and ideal airbrake deployment percentage was determined.

This relation was then tested by implementing a software-in-the-loop model of the avionic. As the developed rocket performance model is modular, this change happened quickly and concurrently to the development of the constant commanded deployment model of the avionics. In addition to implementing the determined relation as the control scheme, the rate of actuation was also limited to match the hardware limitations. The results of these proofing runs showed promise in achieving the end goal of a so-called chronically accurate apogee, and thus this implementation will be tested at SA Cup 2019 on uORocketry's launch vehicle, Jackalope I.

PHYSICAL IMPLEMENTATION OF AIRBRAKE LOOKUP CONTROL

The physical implementation of the lookup control scheme was shaped heavily by the hardware used for the SRAD avionics system. As the control scheme had been simplified from a full model predictive control method to a lookup method, it was decided that an ATMega 328P would suffice as the central computing element for the SRAD avionics. This allowed the team to quickly implement and test all algorithms slated for flight. The ATMega 328P has a clock speed of 16-MHz and storage in the form of EEPROM with size 1-kB. As the 30-kB of useable flash memory may need to be expanded for future development of this avionics platform, the lookup data was encoded and stored in the EEPROM. The avionics system is also equipped with a full suite of redundant sensors which include two 9 DOF (degrees of freedom) inertial measurement units (IMU), two barometric altimeters, one GPS module and a radio for air-to-ground communications. While the central computing element of the avionics was simplified, the quality of sensors was increased to help accurately estimate current altitude and apparent velocity. A run through the control loop is as follows: the sensors are first polled, the current altitude and apparent velocity quantities are used to find the index of the ideal airbrake deployment percentage using binary search, the deployment percentage is converted to a pulse width modulated (PWM) signal to send to the actuation servo which drives the airbrakes to the desired position.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

It is in the nature of many engineering practices, and especially rocketry, that some aspects are difficult to optimize or even estimate given resource constraints. However, many times these difficult aspects are crucial to achieving the desired performance. For this reason, uORocketry has developed arocket performance model. To test and demonstrate the effectiveness of this ideology, the team decided to construct the non-obvious relation between current altitude and apparent velocity to ideal airbrake percentage deployment. This relation will act as the new robust control scheme for the custom airbrakes presented in the previous year. While the developed models and simulations may get close to the truth in a meaningful way, reality is the only true model; and as such, Spaceport America Cup 2019 will act as a means to collect data and measure the variations between model and reality. This data will then be used to augment the developed models to increase their accuracy. This cycle is planned to continue concurrently with the development of future generations and provide incremental improvements to flight performance.

REFERENCES

[1] Genevieve, B., "Development of a Hybrid Sounding Rocket Motor," Master's Thesis, School of Engineering, University of KwaZulu-Natal, South Africa, Submitted, December 2012

[2] James, R. L., "A Three-Dimensional Trajectory Simulation Using Six Degrees of Freedom with Arbitrary Wind," NASA TN D-641, 1961

[3] Mason, W., Applied Computational Aerodynamics, http://www.aoe.vt.edu/~mason/Mason_f/CAtxtTop.html , pp. A-27–A-28. [accessed 14-Mar-2009].

[4] Whorton, M., Hall, C., and Cook, S., "Ascent Flight Control and Structural Interaction for the Ares-I Crew Launch Vehicle," AIAA 2007-1780, April 2007.

[5] Zipfel, P. H., "Modeling and Simulation of Aerospace Vehicle Dynamics," AIAA Education Series, 1998.

Comprehensive Testing Procedures for Robust Avionics

Team 21 Project Technical Presentation to the 2019 SA Cup

Mei Qi Tang
McGill University, Montreal, Quebec, H3A0E9

INTRODUCTION

Testing procedures and standards are a key aspect to developing reliable avionics, and have been lacking in the design process for the McGill Rocket Team (MRT) until this year. It is very difficult to predict the rocket's behavior in-flight, and even more difficult to simulate that behavior on the ground. As such, it is important to develop a wide variety of tests to both guide the avionics design, and to validate the final product. These sort of tests are the building blocks of avionic system development, and are required to ensure the success of continuously more complex SRAD systems. The testing procedures presented divide into two categories: configuration and integration testing. Configuration testing is meant to drive design choices for the most optimal performance of an individual subsystem. On the other hand, integration testing validates the design choices of an individual subsystem with respect to its neighbouring subsystems through interference analysis, aiming to optimize the overall system performance. Both types of tests were applied to RF Radios and COTS Antennas to validate the telemetry design for MRT's 2019 SAC submission – *Caladan*.

CONFIGURATION TESTING

Configuration testing was done entirely indoors, inside an anechoic chamber, which allowed repeatable measurements to be taken in an ideal test environment without any outside interference or RF reflective sourroundings. First, a preliminary exploration on the ideal placement of radios in the rocket was conducted. It was already known that carbon fibre, a material used for many structural elements in Caladan, is somewhat conductive and therefore will act as a ground plane reflecting signals1. However, due to assembly and other design considerations, the COTS patch antennas had to be in proximity to the carbon fiber material. Therefore, preliminary configuration tests in the anechoic chamber were conducted to study the effect of antenna orientation and placement with varying distances to the carbon fiber material. These preliminary tests were not only easy to setup, but were also designed to be remote-controlled with automated data recording and post-processing. These allowed for efficient preliminary optimization of the most performant configuration of antenna placement around RF reflective materials.

Once the top desired configurations were selected, the team then conducted more precise, but also more time consuming, antenna radiation pattern tests using NSI-MI Technologies Near-Field test equipment in the anechoic chamber. With the rocket's flight behavior in mind, the configuration with a radiation pattern that showed the highest potential of maintaining a stable link was selected.

INTEGTRATION TESTING

Although a best-performing, or at least functioning, configuration is important, it does not prove that the subsystems won't interfere with each other or that the system will work under real flight conditions. To gain that confidence, MRT ran indoor spectrum analyzer tests, as well as short-range and long-range outdoor tests. By displaying the magnitude power level of an input signal over a specified frequency range, the spectrum analyzer gave insight on potential interference effects such as intermodulation between the radios. To see how the system would perform closer to in-flight settings, the AV bay was lastly studied outdoors. Due to simpler logistics and setup, the preliminary tests occurred at a short range (15 to 100 meters), simulating long range with added attenuators. Once these preliminary short range tests showed the ability of the system to communicate at the imposed "long distance", a true long-distance test was carried out. Separating the receivers and transmitters for all of our telemetry systems by a distance of first 10-km (about apogee distance), and subsequently 15-km (potential recovery distance), signal strength, throughput, and interference were measured and a final system validation was performed. The distance tests were carried out with different orientations of the rocket, including a spin test where the rocket spun around its primary axis for several minutes, to account for the different orientations expected in flight. This concluded the testing procedures with a confirmation that, not only are the on-board radios and antennas positioned in the optimal spot, but also that they perform as expected in the field, under different applied stressors.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

These testing procedures will continue on to serve as the basis of all of MRT's avionics development in the future. The need to have quantitative analysis to back up design decisions for rocket electronics cannot be overstated, and is absolutely necessary when looking ahead at creating any type of SRAD system. In tangent with testing, MRT has been developing SRAD antennas over the past year. This is the goal of MRT's short-term follow-on work, and the tests, now proven to work on COTS modules, will be used to improve SRAD modules in the coming year to the point where a reliance on COTS will no longer be necessary.

REFERENCES

[1] Mckenzie, A. B., and White, S., "Characterization Of Electrical Conductivity Of Carbon Fiber/Epoxy Composites With Conductive Afm And Scanning Microwave Impedance Microscopy," University of Illinois, 2015.

Reliable Modeling and Experimental Validation of Staging Events for High Power Rockets

Team 24 Project Technical Presentation to the 2019 SA Cup

Dan Fries, Benjamin Jensrud, Robert C. Wilson and Matthew Chicky
Georgia Institute of Technology, Atlanta, Georgia, 30318

INTRODUCTION

We present an analytical, one-dimensional model based on five ordinary differential equations (ODEs) to predict the behavior of two rocket stages after ignition of the separation charge, for the case of linear sliding motion along the stage coupler. Input parameters include aerodynamic drag coefficients, connecting forces between the stages, masses, dimensions, state-of-flight and altitude of the rocket at separation initiation, leaks in the rocket body and properties of explosives used. The ODEs are solved to yield changes in the velocity of the two stages with time, forces acting on the inside of the rocket due to the separation charge as well as internal volume, pressure and temperature. Given an initial separation condition (e.g. distance of the two stages from each other), the model also returns the amount of black powder burned, time from initiation to separation, as well as changes in altitude and Mach number. This approach yields itself to a stochastic sensitivity and uncertainty analysis, where select input parameters are varied according to a pre-defined probability distribution. Large enough samples of different parameter combinations reveal trends in staging behavior and parameters which are critical for successful separation. Moreover, expectations and standard variations can be computed for some of the results, e.g. altitude change, largest pressure or time to separation. Such data informs two important questions of the rocket staging design: one, which combination of parameters is most likely to result in successful separation and does my rocket fall into that parameter space? Two, which of my input parameters do I need to know with greater accuracy to make better predictions? Finally, we present results and lessons learned from attempts to compare and validate the staging model with an experimental rocket surrogate setup. Both the model and the experiments informed the design of our rocket, to be flown at the Spaceport America Cup (SAC), in addition to common knowledge and experience.

MODEL DESCRIPTION

The main part of the model consists of five ODEs: (1) Law of motion for the second stage; (2) Law of motion for the first stage; (3) Change in the internal volume of the rocket due to the stage sliding apart; (4) First law of thermodynamics to determine temperature change due to energy released by the separation charge; (5) Changes in gas mass inside of the rocket body due to leaks and relief holes. This set of equations is augmented by the ideal gas law to compute pressure, an expression for variations in the constant volume specific heat with gas temperature, a burning model for black powder and equations for choked and unchoked air flow in and out of the rocket body.

The model is one-dimensional and only considers motion of the two stages along the rocket central axis as they slide along the interstage away from each other. It assumes a thermally perfect gas, constant aerodynamic drag coefficients and a constant frictional force between the stages until separation. The user can specify the usage of shear pins which will prevent any motion of the stages relative to each other until the internal pressure has built enough to overcome both the frictional force as well as the holding strength of the shear pins. The separation charge burning model currently considers burning of black powder in a charge well of arbitrary geometry. The burning and heat release process are time resolved and utilizes published results from so-called black powder strand burning experiment,[1 2] i.e. the mass burning rate is calculated by the density of the black powder, the surface area and an empirically measured regression rate, which is pressure dependent.

To initiate the model, the user needs to supply drag coefficients, ambient conditions (e.g. separation height), state of motion of the rocket at the initiation of the separation as well as masses and some geometric details. The main outputs of the model are changes in velocity of the two stages, temperature, pressure and mass of air inside of the separation chamber, mass of black powder burned, change in altitude and Mach number and time from ignition to separation. From that, the forces acting on the rocket bulkheads and accelerations can be calculated, too. Masses and dimensions are supplied by direct measurement on the actual rocket. Aerodynamic coefficients and flight conditions at the initiation of the staging event are obtained from simulations using RASAero II.[3]

SENSITIVITY AND UNCERTAINTY ANALYSIS

For the stochastic, scatter sensitivity analysis and uncertainty quantification (i.e. a Monte-Carlo approach)[4 5] we allow the burning rate prefactor, the separation height, the booster and sustainer mass, the drag coefficients and the frictional force between the stages to vary. The variations are prescribed by normal probability distributions, with mean values corresponding to nominal or desired rocket operation and standard deviations based on literature and confidence in our preliminary estimates and measurements. In some cases, the distributions must be restricted to avoid unphysical values. Random number generators based on these distributions provide parameter combinations for the staging model to evaluate. The resulting data can be used to compute expectation values, confidence intervals and reveals specific sensitivities in the outcomes of the model.

EXPERIMENTAL SETUP

The experimental setup consists of a sustainer surrogate and a stationary interstage coupler, to avoid damage to the actual flight hardware. We perform the experiments in both vertical and horizontal configurations with the same separation charge mass of 4F Black powder in both cases. The mass of the sustainer surrogate has to be modified to account for performing the dynamic measurement in a stationary-booster reference frame and the frictional forces are measured using a spring scale. Proper consideration of forces acting in the experimental and flight cases allows for a normalized comparison between model and experimental results. This enables further validation and refinement of the model to make sure that the likelihood of successful separation during the actual competition flight is maximized.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Starting with the sensitivity analysis, the model shows that for the range of input parameters provided a mass of 5-g of black powder results in successful separation in more than 60% of the cases and the most important variable for successful staging is the black powder burning rate. While this is not completely unexpected it is significant as the burning rate of black powder strongly depends on grain properties, initial compression and surrounding gas pressure. Changes in burning rate with compression can be observed even just for hand compacted black powder. The analysis also yielded an upper limit of 600-N for the frictional force between stages to guarantee successful separation in most cases. Overall, 90% of the 3000 random samples taken from the prescribed parameter space result in successful separation with a mean time to stage separation of 210-ms, increasing confidence in our rocket designed for the SAC. The uncertainty quantification shows rather large standard deviations for some of the figures of merit. Specifically, the reliability of black powder burning parameters is questionable due to a lack of relevant, published data.

The two experiments performed with 5g of 4 F black powder both yielded successful separation of the sustainer surrogate from the stationary stage coupler. Comparing accelerations from the two experiments to the model showed that the acceleration is significantly underpredicted, most likely due to the simplified burning model and uncertainties in the burning rates provided by literature. Moreover, a normalization of the experimental accelerations does not collapse the results, showing that the staging process changes with the hardware setup in a non-trivial manner.

An analytic model is proposed that aids in the design of staged amateur rockets. It can be used to predict staging behavior, quickly explore the design space and provide information regarding parameter sensitivity and reliability. The analysis yields bounds for the required black powder mass and minimum burning rate, the maximum allowable frictional force between the stages as well as a time-resolved separation process including variables such as temperature and pressure inside the rocket. The results were used to design staging related parts of our SAC rocket and successful outcomes were confirmed in two surrogate experiments. Predicting and verifying our rocket design in this fashion increases operational safety and confidence in achieving the competitions goal successfully. Using the experimental data we also attempted a validation of the proposed model. This showed that the model underpredicts the timescales and accelerations during the black powder initiated staging process. Thus, at the present time the model can only provide conservative estimates but no exact data, making further refinement desirable. In the future, a better quantification of black powder burning parameters should be attempted to make the model more accurate. Moreover, the burning model should account for the point of ignition of the charge well, which results in a more confined space for burning and potentially stronger increases in pressure/acceleration than currently predicted.

REFERENCES

[1] Brown, M. E. and Rugunanan, R. A., "A Temperature-Profile Study of the Combustion of Black Powder and its Constituent Binary Mixtures", Propellants, Explosives, Pyrotechnics , Vol. 14, 1989, pp. 69-75.

[2] Sasse, R. A., "A Comprehensive Review of Black Powder", Technical Report BRL-TR-2630, US ARMY Ballistic Research Laboratory Aberdeen Proving Ground, Maryland, January 1985.

[3] Rogers, C. E. and Cooper, D., "RASAero II: Rocket Aerodynamic Analysis and Flight Simulation Software", Rogers Aeroscience , Copyright 2016.

[4] Saltelli, A., Ratto, M., Andres, T., Campolongo, F., Cariboni, J., Gatelli, D., Saisana, M and Tarantola, S., "Global Sensitivity Analysis: The Primer", John Wiley & Sons, 2008.

[5] Madsen, H. O., Krenk, S. and Lind, N. C., "Methods of structural safety", Courier Corporation, 2006.

Design, Analysis and Testing of an Additively-Manufactured Bipropellant Liquid Rocket Injector

Team 29 Project Technical Presentation to the 2019 SA Cup

Nick Gloria and Max Newport
Stanford University, Stanford, California, 94305

INTRODUCTION

Additive manufacturing has changed aerospace propulsion design over the past few decades, simplifying production processes, reducing costs, and enabling more efficient designs. In the rocket industry, metal 3D-printing has led to a stream of innovations and startup companies.

DESIGN REQUIREMENTS

The Over the course of two academic years, our team at Stanford University has partnered with 3D-Systems, a pioneer in the metal 3D-printing industry, and ProtoLabs, a rapid-prototyping service, to experiment with a 3Dprinted metal injector. The injector component is ideal for reaping the benefits of 3D-printing. This part often requires complicated patterns of orifices and manifolds. The injector also sees strong thermal gradients from close proximity to the flame zone, requiring cooling. Our project has utilized the Direct Metal Printing technology with an Inconel and later a stainless steel alloy to create an injector that (1) combines multiple parts into one, simplifying production, (2) has a sophisticated but efficient injector spray pattern, (3) has a complex and efficient internal manifold volume that would be impossible to reductively machine, (4) withstands the high thermal loading from intense combustion, and (5) reduces the part mass by means of internal lattices.

While 3D-printing eliminates many constraints that reductive manufacturing imposes, it also introduces a new set of constraints, which are reflected in the design of our injector. The most significant constraint is the requirement of a maximum overhang angle, which limits the creation of internal volumes. Additionally, a minimum wall thickness constraint is the main mass-driver for the injector design at this small scale. The constraints imposed by 3D-printing can be accommodated with creative solutions.

MANUFACTURING

In the first prototype, a like impinging spray pattern was used, with orifices at the limits of 3D-printing capability at 15 thousandths of an inch in diameter. Additionally, an array of film cooling orifices released fuel near the chamber walls. This design led to highly-stable combustion with low RMS combustion chamber pressure oscillations and almost negligible ablation of the combustion chamber liner.

Subsequently, a second prototype was designed that reduced the part mass by 50%. The design process was reinvented - the internal fluid volumes were designed first and a containing shell was constructed around them. A lattice structure was incorporated to take the pressure loads of the injector face with a low mass penalty. The spray pattern was changed to a quintuplet unlike impinging scheme in order to increase the fuel hole diameter and thus manufacturing consistency.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Critical to the short development cycle of the engine was the use of a high-fidelity model of nitrous oxide flow modelling, based on recent research completed at laboratories at Stanford University. The Homogeneous Equilibrium Model was used along with non-ideal propellant tank thermodynamic modelling to estimate the oxidizer mass flow rate within 8% from later test results. The results of this engine development project have demonstrated that 3D-printing indeed provides numerous advantages in rocket injector performance.

Numerical Modeling of a Student-Made Parachute

Team 40 Project Technical Presentation to the 2019 SA Cup

Hélène Papillon Laroche and Olivier Jobin
Polytechnique Montreal, 2500 chemin de Polytechnique, Montreal, Qc, H3T 1J6, Canada

INTRODUCTION

For the second time in its history, Oronos will launch a rocket with student-made parachutes at the Spaceport America Cup 2019. The team created and validated, through numerical simulation and mathematical modelling, a methodology to assist the design of parachutes for the years to come. A numerical methodology using CFD was developed to design new shapes and sizes of parachutes. This allows the team to go beyond the limitations of size and shape of mainstream parachute manufacturers. More often than not, rocketry teams need to choose and unoptimized parachute for their rockets because of the limited variety of commercial shapes and sizes . With the developed procedure, it is possible to move past this limitation.

RESEARCH OVERVIEW

Using an open-source CFD software (SU2) developed by several university research teams, 3D simulations were done to demonstrate and to simulate the behavior of the flow around parachutes. SU2 uses a finite volume method to solve the Navier-Stokes equations. First, to validate the proposed approach, a commercial parachute from Fruity Chutes was selected from their website. Using Engineering Sketch-Pad (ESP), an open-source solid-modeling system developed by the MIT, a 3D model of the selected parachute was produced. The main reason why the team choose ESP as the 3D modelling software is its ability to create "water-tight" CAD and insure an expected behavior of the flow around the geometry (the flow does not penetrate the parachute's surface). It was used to produce an unstructured mesh with Pointwise, a commercial software used in the aerospace industry. Unstructured meshes were generated because of their ability to discretize more easily complex geometries rather than structured meshes. Then, the mesh was used in SU2 to simulate the flow field around the parachute. Preliminary results showed an error of 1,33% between the drag coefficient calculated by the team and the one given by Fruity Chutes for an elliptical parachute. With these results, the team concluded that the CFD definition of the problem was adequate.

With this computational strategy, the team created a mesh of the actual dimensions of the main parachute fitted for Atlas: a toroidal parachute of 120-in of diameter. The computational domain extends from the parachute's surface to a farfield, modeled as a vertical cylinder. Its diameter and height are respectively 8 and 12.5 times the parachute diameter. The mesh discretizing this domain is composed of 2,46 million elements to simulate the flow around the geometry. During the mesh generation, a special attention has been directed to the boundary layer at the parachute surface and the wake upstream the geometry. The computations are done with a Reynolds-Averaged Navier–Stokes (steady flow) and Unsteady Reynolds-Averaged Navier–Stokes (unsteady flow) equations with a Spalart–Allmaras turbulence model. Because of the transient behavior of the phenomena, steady and unsteady simulations were performed. Furthermore, the Navier-Stokes equations were chosen to simulate the flow in viscous regime. This allows the measurement of the drag induced by skin friction, not only the form drag measured by other methods such as Euler equations. To create the tests cases, a "velocity inlet" boundary condition is used on the bottom of the cylinder corresponding to a descent rate of 30-ft/s, the maximum allowed by the competition's rule. A "pressure outlet" boundary condition is applied to the side and the top of the cylinder. The computations were done on Compute Canada's newest supercomputer, Béluga, the most powerful research computer in Canada.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Results of these simulations were in the range of expected values of commercial parachutes such as the ones sold by Fruity Chutes. The physics was successfully captured by the simulations. Results show symmetrical recirculation zones above the parachute. Additionally, the pressure underneath the canopy is higher than the pressure above the parachute. This indicates that the results are in accordance with the physics of the flow. With the aforementioned results, it is possible to conclude that the approach is well suited for the requirements of the team and could be applied in aerodynamic analyzes for the rockets in future works.

Centralized Flight Computer Design and Testing

Team 42 Project Technical Presentation to the 2019 SA Cup

B. Dempsey and Y. Sahota
Cornell University ECE, Ithaca, NY, 14853

INTRODUCTION

The idea behind the central flight computer came from noticing that in previous years Cornell Rocketry Team (CRT) had many electronics onboard the rocket, and many were collecting the same data throughout flight. However these devices were not communicating with each other, so it was necessary to have this redundancy. These redundancies wasted valuable CRT time in designing PCBs that shared the same function and made it difficult to put the rocket together right before flight because we had more components. By performing all or most of the measurements on a single flight computer board and sending data out to other boards using radio communications, we would be able to build a much more elegant rocket.

CONCEPTION

We wanted to create a central system that could not only send telemetry information, but also actively control flight surfaces and exert control over the entire flight process from launch to landing. We already had existing systems to work off of for the telemetry portion of this new system, but the other desired features were entirely new. Beyond discussion of this new system, we began identifying components and picked out parts that would allow us to fulfill our requirements. This was a precursor to the actual design process for the first revision of the board.

DESIGN

After coming up with the idea for the flight computer, we set out to design the PCB using Altium Designer, which was kindly provided to us by Altium. Using the components we found during the conception step, we created footprints and schematic symbols for each component, creating a library as we went. Next, we started wiring all of the components together, making sure that both the connections on the board and the connections between our board and the Pi were correct. After having a schematic we were confident in, we started the layout where we arranged components close to other components it is actually wired to, and we also tried to spread out components that produce lots of heat.

For the second revision of the board, we used information we learned from the original board to help create a board with better functionality. To this extent, we tried switching from linear voltage regulators to buck converters because linear voltage regulators are horribly inefficient (and waste power as heat). However, because our flight computer is a high current device and needs three different voltages, we needed to add five buck converters. When this many buck converters are attached to the same board, unless the decoupling between power and ground is superb, there will be too much noise on the lines to get a consistent voltage. We ended up having up to plus or minus 25% noise on the power lines, so 3.3-V could swing up to over 4-V or down to 2.5-V, which causes the devices using this power to have all kinds of issues ranging from total failure, to simply getting bizarre values from time to time. In most cases, we didn’t see total failure of devices, so it was less obvious what was wrong with the board and it took longer to diagnose these problems. While we had many issues with the power supply for the board, the sensors themselves all worked.

TESTING

The initial revision of the flight computer was tested at our subscale flight with the airbrake subsystem. This revision had a functioning barometer, which was used to estimate the altitude to control the airbrake fins. Data was also recorded from available sensors during the flight to analyze the behavior of the airbrake and its effect on the target altitude. After the first revision successfully deployed the airbrake at the target altitude, we began expanding our suite of sensors for the second revision.

The next revision of the flight computer involved testing of the onboard accelerometer, gyroscope, an RTK GPS module, and the blast charge ignitor. The accelerometer was tested using the High Impulse Test Rig (HITR), which was created by our Independent Testing and Validation (INTEV) subteam last year. The other sensors were tested using standard testing procedures. The ignitor was tested using an e-match and simulating apogee on the flight computer. This test was successful and repeatable, suggesting that the flight computer was ready for testing onboard the next launch.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

While the flight computer was not finished this year, we were able to get through two versions of the flight computer, the original and one revision. With only one revision, a PCB is unlikely to carry out every function properly when it has so many components. Additionally, as this was the first year we were using a Raspberry Pi, we learned many valuable lessons about Linux in general, and especially about sudo access. With many younger members on our team, there were multiple occasions where we had to reflash SD cards with a fresh install of Raspbian because one of our members had deleted something they shouldn't have. This taught us rather quickly that not everyone should have full sudo access, but we found that to access many of the hardware capabilities of the Pi a user had to have more permissions than that of a normal user. We also spent significant time working to get all of our team just enough permissions so that they could work on their systems, but they did not have enough power to break Linux. In the next season, we plan on finishing the flight computer so that it is capable of all the functions we have previously mentioned. It will use a barometer, gyroscope, and accelerometer in combination with a hybrid Kalman filter to determine characteristics of flight. This data will then be sent to the payload, used to set off blast charges, and live streamed from the ground station via radio communication. Furthermore, the flight computer will be running accurate simulations of apogee so that we can determine how much the air brake should deflect at any given point.

Rocket CAN: A Modular Extensible Avionics System

Team 45 Project Technical Presentation to the 2019 SA Cup

Alex Mihaila
University of Waterloo, Waterloo, Ontario, N2L 3G1, Canada

INTRODUCTION

Waterloo Rocketry has been designing and building hybrid sounding rockets for nine years. As the mechanical systems onboard rocket have become more complex, the corresponding electrical systems have evolved in order to support them. In the last competition cycle, an engine electronics system was developed, consisting of a student-designed printed circuit board (PCB) containing remotely actuated valve drivers, radio communication, a full analog and digital sensor suite, and data logging. As further extensions to the rocket were discussed, it became apparent that extending the existing electronics system would be intractable due to physical constraints, power requirements, and extensive subsystem coupling. This year, we aim to solve these problems by decoupling the existing monolithic avionics into several single-function modules connected by a Controller Area Network (CAN Bus). In its current configuration, this system (named RocketCAN) contains a radio board, multiple valve driver boards, a sensor suite, a logging system, and a GPS receiver. The system integrates with the team's existing Ground Support Equipment (GSE), fails safe, and allows straightforward integration of future systems. The system hardware and software are entirely student designed.

RESEARCH OVERVIEW

The first iteration of engine control electronics was developed for the 2018 competition cycle, in which two electronically actuated ball valves were added to the rocket. The first valve, known as the "injector valve," replaced the team's previous pyro-valve between the oxidizer tank and combustion chamber. This was done in order to achieve a rapid thrust ramp at launch. The second valve, known as the "vent valve," replaced a permanent vent hole in the oxidizer tank. Both valves must be actuated remotely. Additionally, the system should collect oxidizer tank pressure during fill operations and relay them back to the launch operators. Finally, the system must fail as safely as possible.

In order to accomplish this, a printed circuit board (PCB) containing a radio module and a motor controller was designed. A flight data acquisition system was also incorporated into the board. Since the injector and vent valves were driven by the same type of ball valve, the same board could be used to control both. As such, two copies of the same PCB were manufactured and placed inside the vent and injector sections. The PCB was designed such that the valve could be configured to open (for the vent valve) or remain stationary (for the injector valve) upon loss of communication. This system was used to successfully launch Unexploded Ordnance (UXO) at IREC in 2018.

While successful, the original engine control electronics had several drawbacks. First of all, since all the radio communication, valve driving, sensing, and logging capabilities were contained on a single PCB, the boards were large, with an area of 10 square inches each. These were difficult to accommodate in the crowded vent and injector sections. Second, since the boards were identical, functionality such as radio communication and data acquisition were duplicated between the boards. This added unnecessary cost and increased power consumption. Third, because the submodules were coupled together on a single PCB, failure of one submodule meant that the entire PCB needed to be swapped out. Since a single board was rather expensive, this was undesirable. Finally, and most importantly, the system was difficult to extend. Any new functionality would require either redesigning the engine electronics entirely (making it even more unwieldy), or designing a separate PCB that potentially duplicated existing functionality (which is both cost- and power-inefficient).

RocketCAN was developed to address these problems. The functionality of the original engine electronics are decoupled into several single-function modules connected by a Controller Area Network (CAN bus). The current RocketCAN configuration consists of a radio board, an injector valve driver, a vent valve driver, a flight sensor suite, a system logger, and a GPS receiver. The system implements the same valve fail-safes as before, while adding general system health monitoring. The modules freely share information over the CAN bus, allowing all modules to access resources such as the radio transceiver and sensor module. This minimizes the amount of component and functionality duplication, as well as power consumption (especially by the radio module). Power consumption is reduced further by allowing the radio board to control power to the rest of the system, forcing it into a low-power state while the rocket is idle. The resulting PCBs are also significantly smaller and easier to mount inside the rocket. Finally, RocketCAN's configuration is easily extensible. A new subsystem simply needs to include a CAN communication module, after which it can communicate with all other boards on the bus. Potential extensions such as a flight computer, a state estimation module, or an air braking system would be able to use existing sensor data without implementing their own independent sensor suite.

The CAN protocol was chosen for its high data rate, priority-based arbitration, and abundance of inexpensive hardware. The bus can safely be used by many boards at once without the risk of a low-priority message (eg. Sensor data) preempting a high-priority message (eg. valve command). Microcontrollers with CAN modules, stand-alone CAN modules, and CAN transceivers are all widely available, making it inexpensive and simple to add CAN functionality to a subsystem.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

RocketCAN today matches and exceeds all of the capabilities of the previous engine instrumentation project and allows avionics to be improved rapidly and with comparative ease in the future. System improvements in this competition cycle include increased battery life, GPS position recording and broadcasting, and visibility into system health. Future system improvements will likely include additional sensors, redundant radio communication, state estimation hardware, real-time flight telemetry, and payload integration. RocketCAN is intended to serve as a development baseline in the future.

Design, Analysis, and Testing of Parachute Reefing Systems

Team 49 Project Technical Presentation to the 2019 SA Cup

Alexandra Rivard
Université Laval, Quebec, Quebec,Canada, G1V0A6

INTRODUCTION

In rocketry, it is well-known that weight is one of the most important factor to consider. Laval University Aerospace Group (GAUL acronym in french) has developed and built many sounding rockets in the past using commercial rocket motor and is currently developing a hybrid engine. With that evolution in mind, the team is trying to reduce future vehicle weight as much as possible. Even though GAUL is participating in the COTS category this year, it took the opportunity to drastically change the rocket size and design in prevision of next year challenge. One of the design revision was to reduce the number of parachute used to one instead of two while still producing a dual event recovery.

SOLUTION

The solution GAUL found for this problematic was the use of a reef to control the opening of the single parachute making it possible to use it as a drogue and as a main. While in flight, shortly after the rocket reaches it's apogee, the first recovery event is triggered releasing the parachute but the reef, which is a nylon string typically used as a suspension line placed in a loop, restricting its full opening. When the vehicule reaches a target altitude after the apogee, a mechanism breaks the reef allowing to full deployment of the canopy which slows down the rocket descent.

DESIGN

There are many ways to cut a reef. The selected design is a striker that hits the reef with great kinetic energy and breaks it. The energy comes from a black powder explosion. The striker itself is guided by an aluminium tube called a casing and is stopped by a steel cap. An E-Match is used to ignite the black powder. The 3D-printed component used to hold the E-Match comes from our homemade CO2 deployment system to make it versatile. The reef, as said earlier, is a nylon string loop, but the object that gets cut by the striker is a plastic tie-wrap since this material is easier to cut, is cheaper and reduces waste since it allows to reuse the same nylon cord. There are two reef-cutters on the parachute for redundancy. Both reef-cutters are on the same support and will cut the same tie-wrap to release the reef. The support is 3D printed to fit the exterior shape of the reef-cutters. It supports two reef-cutters, a connection box and two permanent electric wires that run down the suspension lines to join the avionic part in the rocket.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Ground tests results were conclusive enough to give confidence that the system could be trusted. Its strength is its simplicity, it's redundancy and also the fact that the mechanism is triggered by two stratologger, a well know and tested commercial component which has been used for years by the GAUL. An interesting follow-on work would be a system that would control the parachute steering allowing it to land as close as possible to the launch pad. While not helping on the weight issue, the system would save recovery time and ensure the rocket never get lost.

Student Organization for Aerospace Research Sounding Rocket Composite Overwrapped Pressure Vessel (COPV)

Team 50 Project Technical Presentation to the 2019 SA Cup

J. Martens, M. Corbiell, W. Nelson, I. Savytskyy
University of Calgary, Calgary, AB, T2L 1B3, Canada

INTRODUCTION

The University of Calgary Student Organization for Aerospace Research (SOAR) had previously designed, tested, and used an all aluminum (6061-T6) vessel to store 42-L of Nitrous Oxide for its flight vehicle Atlantis I in IREC 2017. The goal of the presented project was to significantly reduce the weight of the tank, down from 50-lbs. The project originally designed a 1/16-in walled liner with overwrap that was 17-lbs, but due to sourcing constraints, a 1/8-in liner was purchased. The presented design still has a 25-lb or 50% weight reduction.

This abstract will describe the problem formulation, analysis, and design of the vessel. The project entails the vessel geometry, the physical attachments, fluid access, the model generation and results of initial analysis, and finite element analysis (FEA).

FINAL CONCEPT

The Final Concept is a tank that exists as a structural member of the airframe, since the pressure loads are significantly larger than the flight and handling loads, the other loads are withstood by the pressure vessel design and redundant structures (external airframe body tubes) are neglected to save weight. The end caps are not hemispheres, but isotensoid shapes.[4] The optimal shape for isotropic materials is a hemisphere, but the optimal shape for an-isotropic materials is a shape called an isotensoid that appears elliptical in nature. It is not well defined analytically, but rather defined numerically.[4] Fluid is accessed through dual polar ports that have -12 SAE female threads. Radial-axial (RADAX) joints are epoxied onto short body tube sections, which themselves are epoxied to either end of the vessel for attachment to the rest of the airframe. We concluded this is the best attachment method from ANSYS FEA because it evenly distributes the load over the attachment area.

MODEL FORMULATION

Before the design was modelled in ANSYS FEA, the fiber direction and volume distribution had to be predicted to more accurately predict the vessel's behavior. The fiber buildup on the end caps are modeled for geodesic windings, meaning the carbon fiber bands travel in a straight path on a curved surface. The friction force between the first laminate layer and the aluminum liner plus the friction between each adjacent laminate layers does not need to be accounted for as the geodesic does not deviate from a straight path. For a carbon fiber band entering the end cap section from the cylindrical section, the winding angle is fully defined by the ratio of the port radius to the cylindrical radius. The winding angle with respect to the intersection meridian line is shown in literature.[2] When the band is tangent to the port, the winding angle becomes 90°. Multiple methods have been introduced in literature to predict composite thickness across end caps. The simplest method provides a reasonable prediction until the scheme approaches very close to the port where the predicted thickness approaches infinity. Other methods divide the thickness predictions into two regions, the first region being within one bandwidth of the port and the second region being between one bandwidth and the cylinder radius. The equation used to calculate the composite thickness within one bandwidth is also presented in literature.[3]

The seed design that was brought into FEA was determined through netting analysis.[5] In netting analysis, as with the calculations of the isotensoid end cap profile, the stress across the cylindrical portion of the vessel should be loaded in the fiber direction in the case of pure helical windings. It ignores the matrix and assumes the stress is taken up fully by the fibers, a simplification of laminate theory. In the case with both hoop and helical windings, the optimal case is when all fibers experience the same stress. This means that the hoop and helical fibers should have equal stress, thus defining the optimal angle based on the thickness ratio of the hoop to helical layers.[5]

FIRST BODY SECTION

Stress in the cylindrical section was compared to the results found in literature using two different methods. The first method was using netting analysis and the second method was verifying through ANSYS composite FEA. ANSYS was chosen as the simulation software because it was readily available to the team under the University of Calgary's academic licenses. A research paper[3] ran a similar simulation on ABAQUS and a comparison of results with the team's ANSYS simulations would additionally verify the results as accurate. In the below paragraphs, units will be displayed in metric to follow what was used in the compared literature.

The case ran in the literature study used a 1200-mm long pressure vessel with a 300-mm diameter[3]. The results from the team's ANSYS analysis is within 1.38% of the literature value. The ANSYS analysis also did not take into account the aluminum liner whereas the literature simulation did. The effects of the liner were determined negligible due to its extreme thinness of 0.3-mm. The netting analysis result showed to be 6.94% higher than the literature case. This was expected due to the fact that netting analysis completely neglects the strength contribution of the resin in the carbon fiber composite and therefore usually overpredicts the stress. The ANSYS simulation is an improved modeling technique as it takes into account the out-plane shear stresses.

For our design, the composite overwrap and the liner were imported separately and meshed. ANSYS Composite PrepPost (ACP) was used to model the composite layers. The ACP software provided the freedom necessary to implement modifications on composite structures. The composite overwrap was imported from Solidworks as a shell layer and each additional composite layer was extruded from the base layer.

Two areas where the model accuracy can be improved from this basic test case are modeling the fiber volume buildup and the change in fiber orientation across the end caps. A MATLAB script was written to calculate the values of the fiber direction and fiber build. The fiber orientation and build up across the end caps were modeled in ACP via a lookup table, using data outputted by the MATLAB script. The lookup table defined the fiber orientation and layer thicknesses for every node in the model.

The physical properties are slightly more involved than standard FEA analysis. As mentioned earlier, the carbon fiber material provided by the sponsor is TRH50 18K. The epoxy resin is Hexion EPON 826 and the curing agent is Hexion Epikure 9551. The combined unidirectional composite material is first defined to have a certain fiber volume ratio, meaning that a certain percentage of the composite volume is composed of fiber. A fiber ratio of 65% was chosen as the composite material as data was readily available for this volume ratio and it matched the required performance characteristics. In addition to the fiber volume ratio, it has been recommended by Luxfer (the sponsor) to take 70-80% of the rated ultimate strength of the composite material as supported by experimental data.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Ground A composite overwrapped pressure vessel has been successfully created with a theoretical burst pressure of 3000-psi. The internal pressure forces are the strength determining requirement of the vessel, structural loads can be absorbed. The COPV has been manufactured and tested in time for this year's Spaceport America Cup, with reported tests of industry standard proof load and leak test. As a result of the weight savings achieved, SOAR should be able to compete in the 30,000-ft category this year. It is recommended when time and resources permit that the vessel should be burst tested to further validate the design.

REFERENCES

[1] Sulaiman, S., Borazjani, S., and Tang, S. H., "Finite element analysis of filament-wound composite pressure vessel under internal pressure," *IOP Conference Series: Materials Science and Engineering*, vol. 50, 2013.

[2] Moskvichev, E., "Numerical modeling of stress-strain behavior of composite overwrapped pressure vessel," *Procedia Structural Integrity*, vol. 2, June 2016, pp. 2512–2518.

[3] Wang, R., Jiao, W., Liu, W., and Yang, F., "Dome Thickness Prediction of Composite Pressure Vessels by a Cubic Spline Function and Finite Element Analysis," *Polymers & Polymer Composites*, vol. 19, Jan. 2011, pp. 227-234.

[4] Peters, S. T., Composite filament winding, Materials Park: ASM International, 2011.

[5] Zu, L., *Design and optimization of filament wound composite pressure vessels*, Oisterwijk: BOXPress, 2012.

Improving APCP Solids Distribution with Multi-modal Formulations

Team 51 Project Technical Presentation to the 2019 SA Cup

Jeremy Nelson, Matthew Crawley
Southern Utah University, Cedar City, Utah, 84720

INTRODUCTION

The RocketBirds team has been working with SRAD ammonium perchlorate composite propellant (APCP) for the past four years. A major flaw was a significant presence of voids in the propellant. Voids reduce the density of the propellant which negatively impacts the performance of a motor. Additionally, voids significantly increase the amount of surface area available to be burned in a motor in a way that is unpredictable. This makes it difficult to accurately project a motor's performance and could potentially lead to an anomaly during firing. The team decided to focus on reducing propellant voids to minimize these unknowns. A vacuum pump and pressure chamber were implemented into the mixing process, but it was determined the current propellant formula used was too viscous for voids to be readily evacuated during vacuuming process. Although there are many ways to improve the viscosity of a formula, one of the most effective methods is to improve the rheology by optimizing the size and distribution of the AP particles.

THEORY AND RESULTS

The Typical APCP formula consists of solid AP particles and a liquid binder. In a unimodal formula, a significant amount of liquid binder is used to fill the gaps between the particles. The viscosity of such a formula is relatively high. In a bimodal formula, the smaller particles fill most of the gaps in between the larger AP particles instead of those gaps being filled by liquid binder. This results in more liquid binder available elsewhere. Therefore, viscosity is reduced, and voids are more effectively evacuated in the vacuum. By switching from an unimodal formula to a bimodal formula, propellant density increased from 93% to 98% of the theoretical maximum. We furthered this concept by switching from a bimodal formula to a trimodal formula. The two sizes of AP currently used were kept and a small fraction of a third larger size was added. The exact distribution used was chosen from a research paper in which various AP size distributions were tested for viscosity.[2] The result was another significant reduction in viscosity compared to the previous bimodal formula. The density increased to 99.5% of the maximum theoretical. This improvement was realized despite the time spent in the vacuum chamber being halved. We believe that additional reductions in viscosity can be achieved by further optimization. According to another research paper, the viscosity is lowest when the size ratio of fine to coarse particles is 1:10 or less.[2] At this point, the particles fit together so tightly they almost interact independently of each other. Additionally, the most optimal distribution for these particles is a fine to coarse ratio of 3:7. We were unable to incorporate these last concepts into our project this year, but we plan to in the future.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

The Significant improvements have been to the quality of our propellant since our first year. Our density closely matches the theoretical maximum, and we are no longer concerned about voids adversely affecting our motors. A positive side effect of the reduced viscosity is our propellant can now be poured into the casting tubes for easy grain casting instead of being packed. Although the AP distribution was improved this year, there are further optimizations that can be implemented. These improvements could allow for an increase in the solids content of the propellant, which would increase performance.

REFERENCES

[1] Farris, R.J., "Prediction of the Viscosity of Multimodal Suspensions from Unimodal Viscosity Data," *Transactions of the Society of Rheology*, 1968.

[2] Rodić, Vesna, and Bajlovski, Miomir, "Influence of Trimodal Fraction Mixture of Ammonium-Perchlorate on Characteristics of Composite Rocket Propellants," *Scientific-Technical Review*, Vol.LVI,No.2,2006

Development of a Dielectric Barrier Discharge Plasma Actuator for Delaying Flow Separation Over a Rocket

Team 52 Project Technical Presentation to the 2019 SA Cup

James V. Ferrese
University of Washington Society for Advanced Rocket Propulsion, Seattle, WA, 98926

INTRODUCTION

The University of Washington's Society for Advanced Rocket Propulsion's (SARP) new payload is a dielectric barrier discharge (DBD) plasma actuator. This payload offers significant engineering challenges due to its lack of precedent within SARP but equally as many learning opportunities that push the boundaries of aerospace engineering technologies being researched today.[1] Plasma actuators are aerodynamic flow control devices which offer significant advantages over traditional flow control methods because of their relatively low mass, lack of moving parts, fast response time and power conversion efficiency. For SARP's first year of development on this project, the immediate goal is to validate the technology and prove that the plasma actuator has some effect on the flow characteristics over the nose cone during its coast phase. The project began with significant research into past plasma actuator studies and then extended to the fabrication, test, optimization and integration of a working system comprised of several subsystems including power supply electronics, electrode development, plasma activation and control, flow diagnostics and structural support. The end goal of this plasma actuator system is to delay flow transition by generating a circumferential plasma around the rocket's nose cone, reducing drag on the rocket and increasing its apogee capability.

PLASMA ACTUATOR BACKGROUND

The most basic configuration of a DBD plasma actuator is composed of two parallel and asymmetric electrodes separated both vertically and horizontally by a dielectric material.[1-3] The upper electrode, lying flush with the upper surface of the dielectric and exposed to air, is fed an ac voltage usually in the frequency range of 5 to 100-kHz. The voltage signal ranges between two and 20-kV so as to generate a strong electric field between the exposed and buried electrodes. The buried electrode is grounded, lying beneath the dielectric, usually downstream of the upper electrode and entirely encased in insulating substrate. Upon applying the voltage to the upper electrode, a strong electric field is created between the two electrodes.[2] This electric field is greatly influenced by several parameters including the signal frequency, voltage and shape as well as the electrode geometry. The air molecules just above the buried electrode (nearest the upper surface of the dielectric) become ionized from the strong electric field. As electrons are ripped from the air atoms in the presence of the electric field, they leave behind positively charged ions in about equal amounts. In this state, the ionized air is considered a plasma and glows violet. In the presence of the strong electric field, the plasma is accelerated, imparting momentum on the surrounding air and generating a body-force.[1-3] For airflow over the dielectric, this body-force increases the velocity of the flow in the boundary layer. This increase in boundary layer flow velocity delays the point of flow separation over the body allowing the flow to return to laminar before losing its momentum and returning back to turbulent further downstream. Embedded into a rocket's nose cone, the plasma actuator has potential to reduce the skin-friction drag on the rocket. While the concept behind plasma actuators is relatively straightforward, integrating one into a full-scale rocket has proved technically challenging. Much of the previous plasma actuator research has been in a laboratory setting where space, weight and funding are not considered constraints. This makes it significantly simpler to generate the plasma as off-the-shelf systems exist for generating the high voltage and high frequency signal needed to ionize the air. Unfortunately, these systems are large, heavy and expensive – not ideal for the payload of a 14-foot sounding rocket. Furthermore, in a laboratory setting, the plasma can be isolated from other sensitive electronics (the plasma generates significant electromagnetic interference) and also manually controlled. However, on a rocket launching to 30,000-feet, the plasma will be near sensitive avionics hardware and must be activated autonomously. These constraints have led to engineering challenges in the development of a feasible plasma actuator system launched as a rocket's payload.

PLASMA ACTUATOR DEVELOPMENT AND TESTING

For the onboard power supply, a battery pack powering a zero-voltage switching (ZVS) driver and flyback transformer was chosen. The ZVS driver accepts a dc voltage and uses two MOSFET's to generate an ac signal at 25-kHz without amplifying or attenuating the voltage. The ac voltage is then fed into a flyback transformer to amplify the voltage to the kilovolts range. The total power consumption of the power supply system is around 50-W. This system was tested and optimized using iterative testing methods and LTspice® simulations.

The dielectrics of the electrodes are made of a high-temperature 3D-printed resin with the electrodes applied as sputter coated copper in order to minimize their thickness (hence increasing the electric field and minimizing the chance of flow tripping over the exposed electrode). Many iterations of electrode geometry were tested including their shape, thickness, width, gapping and dielectric properties in order to produce the most stable coronal plasma discharge with the highest induced flow velocity. In order to safely operate this system during launch, a series of redundant safety mechanisms will be used including two interlock systems, an arming switch and software safeguards to ensure no high voltage will be produced until after launch and before nose cone ejection.

The onboard controller autonomously activates the plasma upon engine cutoff. Using two redundant microcontrollers and several different sensors (inertial measurement unit, barometer, altimeter and timer), the controllers can determine the rocket's flight stage in real time. Upon entry into the coast phase, the controllers will activate the power supply, actuating it on and off for the duration of the coast phase. If off-nominal flight trajectory occurs, the fault tolerant software will immediately shut down the power supply and prevent it from producing plasma. On nominal flight conditions however, the controllers will also be continuously logging data from three sources. First, they will be recording data on the batteries' power usage so that it can be correlated in time with the two other measurements. These measurements include the heat flux and temperature sensors developed by the diagnostics team as well as current measurements through the grounded electrode (an indicator of plasma generation). The data will be locally saved on an SD card for further analysis upon payload recovery.

As previously mentioned, the diagnostics team has worked to determine the most accurate way to measure changes in the flow characteristics across the plasma actuator. This was accomplished using heat flux and temperature sensors adhered to the outer surface of the nose cone and placed both fore and aft of the electrodes. These sensors were chosen based on their relatively low cost, high sampling rate and ability to measure small changes in heat transfer. Due to the significant difference in heat transfer between laminar and turbulent flow, it is expected that the aft sensors will measure less heat flux than the fore sensors during the period the plasma is active, indicating a return to laminar flow.

All of this hardware is encased in a 3U CubeSat-sized aluminum housing. Not only does this housing provide structural support for all the previously mentioned hardware, it also protects the sensitive electronics from electromagnetic interference (EMI) produced by the plasma. Furthermore, design considerations were taken in order to make the electronics easily accessible and removable so they could be quickly inspected, maintained or replaced. In addition, due to the high-power electronics within the structure, every precaution was taken to efficiently house the electronics while also reducing any chance of internal arcing or EMI.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Each of the aforementioned subsystems has been tested individually and together as a complete system. The power supply and electrodes have been optimized to generate a coronal plasma with the largest induced flow velocity. EMI effects on avionics hardware have been tested to ensure the plasma does not interfere with the telemetry and recovery systems. The onboard controllers have been tested both on the ground and in-flight to make sure the flight phase detection algorithm and safety failsafes work as expected. The diagnostics team has calibrated their sensors and tested them in relevant environments to ensure all sensors work properly and have the response time capable of measuring quick and minute changes in flow heat transfer. The structures team has produced hand-calculations and has ran finite element analysis to ensure their structure will withstand flight loads. Finally, the entire system has been tested together to assess that all subsystems work in synergy. Next year, the team will focus on further optimizing the power supply so that variable ac frequencies can be produced as well as finding ways to draw more power out of the system and into the plasma in a more efficient manner. Furthermore, emphasis will be placed on quantitative testing including measuring induced flow with custom pitot tubes and visualizing the changes in flow characteristics with wind tunnel testing. In future years, this system could be extended to an active trajectory-control system capable of adjusting the attitude of the rocket during its flight in order to keep it on its trajectory. Certainly, with the groundwork achieved by this year's payload team, these ambitious goals will be achieved.

REFERENCES

[1]Corke, T. C., Post, M. L., and Orlov, D. M., "Single Dielectric Barrier Discharge Plasma Enhanced Aerodynamics: Physics, Modeling and Applications," University of Notre Dame, Center for Flow Physics and Control, Aerospace and Mechanical Engineering Department, 2008.

[2] Corke, T. C., Enloe, C. L., and Wilkinson, S. P., "Dielectric Barrier Discharge Plasma Actuators for Flow Control," Annual Review of Fluid Mechanics 2010.42:505-529, 2010.

[3] Houser, N. M., "Manufacturing of Dielectric Barrier Discharge Plasma Actuator for Degradation Resistance," Master's Thesis, Applied Science Graduate Department of Aerospace Studies, University of Toronto, Toronto, Ont., 2013.

Design and Test of an Omnidirectional Patch Antenna Array Integrated On a Sounding Rocket's Airframe

Team 54 Project Technical Presentation to the 2019 SA Cup

Alexandre A. Devienne
Swiss Federal Institute of Technology, EPFL, Lausanne, Switzerland

INTRODUCTION

This abstract presents the development and test process of an omnidirectional antenna array which is integrated around the body of a 15-cm (6-in) diameter rocket, and acts as part of its fuselage. The antenna array is constituted of three rectangular patches, each on a separate panel. The panels are conformed to the cylindrical shape of the rocket. and can easily be unmounted to provide access to the rocket interior. Two working sets of antennas have been designed to work in different frequency bands. They can easily be swapped to comply with radio frequency (RF) regulations at different launch locations (868-MHz for Europe, and 900-MHz for the US). Compared to conventional patch antennas, the present design has double the bandwidth for the same thickness of 1.6 mm (2.1% instead of 1%), allowing for coverage of the entire Industrial Scientific and Medical (ISM) 900 MHz band. A prototype was manufactured for validation of simulation results, and verification of the manufacturing method. After correction in the etching and conforming process of the antenna, a precision better than 0.02% (200-um on 110-mm) was achieved for the flights model's copper patch dimensions. Measurements in an anechoic chamber matched the simulations' results, confirming a low gain variation of less than 2-dB in the rocket roll plane, and two blind spots (gain less than -5-dB) of the array in 60-degree full width cones along the rocket fuselage.

INTEGRATION ON THE ROCKET

The conventional method to integrate an antenna inside a rocket, and get a 360° coverage, is to use a dipole antenna in a section of the rocket which is otherwise empty. This section needs to be transparent to electromagnetic radiation, preventing the use of carbon fiber (CF). The inability to use CF presents additional structural challenges. This method is effective, but not scalable to large-diameter rockets without losing valuable space inside the rocket. To solve this problem, we designed an antenna that could be placed on the rocket's exterior whilst guaranteeing performance, irrespective of the inner structure of the rocket. Microstrip patch antennas were chosen for their low-profile, ground plane which guarantees their performance irrespective of the rocket interior, and inherent low directivity. An array of these were mounted around the rocket circumference to obtain an omnidirectional radiation pattern.

Inspired by our fins module, we developed an open module which serves as a support for mounting our antennas, and at the same time provides access to the inside of the rocket. This open structure comprises three carbon beams and three aluminum rings (bottom, middle and top), which serve as structural interface between the module and the rest of the rocket. The carbon beams are connected to the rings using pressure clamps, tightened with bolts. The antennas are screwed onto the surface of the rings (between the bottom and middle), as well as glass fiber panels (between the middle and top) acting as an RF window for the GPS antennas inside the rocket. The open design of this module was a necessity due to the risk of the research nature of this antenna project, and its criticality for a successful mission. The module allows for integration of a COTS antenna inside our rocket, whilst still getting a reasonable communication link. This modular, flexible design also allows for optimization of the design of the antennas, without needing to jointly adapt the structure to the design, which simplified development and manufacturing. Finally, this module can also serve as an experimental platform for different antenna shapes, or other experiments that could be realized on the surface of a rocket fuselage.

INTEGRATION ON THE ROCKET

Microstrip patch antennas are a sandwich structure, consisting of a low-loss dielectric substrate between two thin copper layers. The choice of the substrate determines the mechanical and electrical properties of the final antenna. A tradeoff analysis between substrate conformability, impact on rocket aerodynamics and electrical performance led us to choose IsoClad® 917 (ϵr = 2.17, tan δ = 0.0013) with a thickness of 1.6-mm as our substrate. We chose to operate in the 900 MHz ISM band as a compromise between operating in a low frequency band (430-MHz), which requires large antennas, and in a high frequency band (2.4-GHz), which has higher transmission losses. This compromise, however, forced us to use the neighboring but different (868-MHz) Short Range Device (SRD) band in Europe due to local RF regulations, requiring construction of two sets of antennas.

We started the antenna design by simulating variations of patch antennas with coaxial feed, using ANSYS® HFSS, Release 15.0.[1] A primary limitation of patch antennas (compared to conventional antennas) are their inherently low bandwidth (1% of the working frequency)[2], which would normally cover less than half of the entire ISM band for the thickness chosen. To increase the bandwidth (while keeping the same thickness), a dual resonance design was chosen, with the dimensions of the rectangular patch optimized such that both TM010 and TM100 modes were in close vicinity, effectively doubling the bandwidth. With such a design, the antenna polarization is neither linear nor circular, but has an ellipticity that varies along the bandwidth. The direction of ellipticity was found to be a function of which quadran of the patch the feed is, and the right-handed design was arbitrarily chosen.

Second order modes design were also investigated, and had interesting radiation patterns, especially when conformed to a cylindrical rocket surface. However, this solution was not chosen, as it requires larger than desired dimensions for the patch, which would prevent reasonable integration within the rocket. A dual-resonance, nearlysquare conformal patch antenna, which used only first order modes, was selected and prototyped. The prototype measurements matched our simulation results, validating our simulation model of the rocket.

Nearly square patch antennas have a beamwidth around 60 to 80-degrees. To provide a full 360° coverage, an array of multiple patches (called elements) around the rocket circumference was designed. Different numbers of elements distributed around the rocket diameter were studied. Three elements were found to provide the best coverage, whilst limiting the cross-coupling between them. Two different phasings of the elements were simulated: in phase and uniform sequential phasing3. With the element fed in phase, the simulated radiation pattern is similar to the one from a half-wavelength dipole antenna, with blind spots along the rocket body. Despite the fact that a uniform sequential phasing of the array could provide a coverage without any blind spots, we decided to feed the array in phase, due to ease of implementation and increased stability across the entire bandwidth.

MANUFACTURING AND TESTING

Two sets of antennas were manufactured, one tuned for the 868-MHz band, the other for the 900-MHz band. During manufacturing, the expansion of the patch due to the bending process must be taken into account, as expansion can be up to twice the required precision for the patch's dimensions. Indeed, to guarantee a bandwidth able to cover the whole 900-MHz band, the etched copper patch dimensions must be correct to within 0.2-mm (less than 0.2% of the patch dimensions). The first set of antennas manufactured allowed for quantification of the expansion, such that the second set could be built with accurate dimensions. The substrate was also found to have a very low shape retention when bent.[4][5] Thermal treatment was necessary to improve its shape retention, such that the different antenna panels would remain conformed to a cylindrical shape. To validate the bandwidth, radiation pattern, and polarity from the simulations, multiple tests were performed. The bandwidth (S11) of the antenna was measured using a Vector Network Analyzer. For the first set of antennas, compared to the simulations, a small shift in the bandwidth was measured. This difference was explained by a difference in patch dimensions between the manufactured antenna and the targeted design. This was confirmed by additional simulations, using the dimensions from the manufactured antenna, validating the accuracy of the simulation models used. The radiation patterns of the antennas were measured in an anechoic chamber, both for a single antenna and the full array. The single antenna was mounted on an electrical model of the rocket, which confirmed the patch beamwidth to within 5-degrees of our simulations. The array performance could only be measured to validate the stage itself, without the surrounding rocket structure, due to the space constraints of the chamber. The radiation pattern measurements are consistent with the simulations, despite imperfections in the power splitter, which feeds the antennas with a 20-degrees phase difference instead of in-phase. These imperfections validated our choice to use a stable and simple phasing strategy for the array elements.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

This abstract presented the design, manufacturing and test of a conformal patch antenna array. The manufactured antennas performance (bandwidth and radiation pattern) match closely the simulations thanks to the development of a high precision manufacturing process. The designed array has an omnidirectional radiation pattern, allowing a communication link between the rocket and the ground station, irrespective of rocket orientation. We look forward to seeing this in action at the 2019 IREC on our rocket, while taking part in the 10'000 feet competition. The presented modular array design also opens research opportunities into electrical beam steering, to improve the communication link reliability, and suppress the blind spots along the rocket axis.

REFERENCES

[1]ANSYS HFSS: High Frequency Electromagnetic Field Simulation Software" Available: https://www.ansys.com/products/electronics/ansys-hfss.

[2] Zürcher, J.-F., and Gardiol, F. E., Broadband patch antennas, Artech House, Norwood, MA, 1995.

[3] Deng, C., Li, Y., Zhang, Z., and Feng, Z., "A Wideband Isotropic Radiated Planar Antenna Using Sequential Rotated L-Shaped Monopoles," Antennas and Propagation, IEEE Transactions on, vol. 62, Mar. 2014, pp. 1461–1464.

[4] Coonrod, J., "Reliably Bend and Form Microwave PCBs," *Microwave Journal*, vol. 56, Oct. 2013, pp. 92–95.

[5] Rogers Corporation, "Bending Etched Antenna Boards," 2003.

Mechanical Rocket Nozzle Ablation

Team 62 Project Technical Presentation to the 2019 SA Cup

Patrick Begalowski, Rachel Cruz, Thomas Hein, and Michal Pękala
University of Illinois at Chicago, Chicago, IL, 60607

INTRODUCTION

To increase the reliability and performance of solid rocket motors, the research and experimental methods of this project will seek to explain the mechanical ablation and thermal effects of solid propellant combustion processes on nozzles. Upon utilization of various nozzle geometries and materials, the initial research promotes further scientific and engineering design exploration in aeronautical and aerospace applications. The Advanced Photon Source at Argonne National Laboratory allows the incorporation of X-ray radiography for ablation rate measurements and observation of other mechanical phenomena. Experimentation involves a solid propellant strand burner as an apparatus to contain commercially available rocket motors at a small-scale.

IGNITION SYSTEM

We will implement a keyed and timed ignition system. This type of setup will provide safety and redundancy for rocket motor ignition. Another benefit to this setup will be a timed feature allows for analysis instrumentation to determine motor start. This, in turn, will allow for an easier analysis and precision measurements of key events including but not limited to igniter startup, igniter pressure build delay, propellant ignition, and motor burn events. This setup also meets the requirements set by the Argonne National Laboratory.

DATA AQUISITION

For Data Acquisition, the team is using a Measurement and Computing 1208-FS-Plus DAQ instrument. To analyze the analog data from our Keyed and Timed Ignition box, along with a pressure transducer located on the strand burner pressure chamber, we are also using Texas Instruments InstaCal Software. This software has been set up to where there is almost no noticeable delay between the power supply and the ignition box itself. This means during experimentation, we can expect a slight delay in the image processing data, which is explored further in the next slide by zooming into the graph produced in InstaCal. The software has also been zoomed in on the analog output from the power source and ignition box. The visible delay here is approximately 0.0125-seconds. Moving forward, the pressure transducer will also be included on the oscilloscope analysis, so the team can evaluate the delay between the voltage output of the ignition box and the actual ignition of the motor.

PYTHON CODE

A high-speed imaging camera will be used from the X-ray. This should capture roughly three thousand images per second. Each image is 1024 by 1024 pixels with each pixel measuring 0,001". Each test burn will be within one to three seconds producing anywhere from 3000-9000 images. When the entire experiment is complete, there should be roughly 54,000 – 162,000 total images. Python code will be developed to process these images and measure how the edge of the nozzle changed over time. During the burn, the nozzle may move around during the capture relative to the X-ray based camera. Stabilizing the images relative to the nozzle may require Python code to implement particle image velocimetry tracking. Once stable images are generated, the next step involves finding the edge of the nozzle used image-based machine learning.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

The Advanced Photon Source, in conjunction with a high-speed camera, allows for optical observation of interactions between propellant product species and nozzle materials. Using Python image processing, along with experimental data, the effects of surface shear forces on nozzle materials was measured and used to graph surface nozzle ablation.

Design of a Low Power Telemetry System

Team 66 Project Technical Presentation to the 2019 SA Cup

Eric Lawin and Thomas Clark
University of Minesota, Duluth, MN, 55812

INTRODUCTION

The electronics committee was tasked with the responsibility of designing and building a low power live telemetry system required for the rocket. Our work is subdivided into two main categories, Receiver and Transmitter. Designs were made to meet club requirements in addition to meeting competition requirements as outlined in the rule book.

RECEIVER

To receive data from the rocket a completely custom circuit board was designed using components found in industry. The Printed Circuit Board (PCB) consists of several main components, the Processor, the SD card, the LoRa module, and support circuitry. The processor is an ATMega2560 with 256-KB of Flash memory clocked at 16-MHz. The ATMega2560 microprocessor provides a powerful command structure enabling fast data acquisition with dependability and low power consumption. To output the data received a 20x4 character display is utilized allowing for real time data to be shown and to ensure that the transmitter is working properly. All data received from the transmitter is stored on a SD card in an Excel sheet to allow for easy data analysis. The inputs allow for the user to scroll through menus allowing different data to be viewed or even manipulated all from the simple five button interface. Two redundant receivers will be used to receive the signal from the rocket. One receiver will use a custom directional antenna named a “Yagi” and the other will use a simple omni directional antenna. The Yagi antenna was designed and confirmed to be the best option for long distance communications by Electrical Engineering faculty. A rocketeer will need to track the rocket via line of sight by pointing the Yagi antenna directly at the rocket. The Yagi allows for a longer distance between receiver and transmitter furthering the chances of success. However, if we are unable to track the rocket visually the omni directional antenna will pick up the signal regardless of orientation but with a reduced range. To ensure receiver reliability high-speed PCB construction techniques shown in the book, Integrity issues and Printed Circuit Board Design by Douglas Brooks, was utilized. These practices allow for high speed signals to be properly channeled with little to no cross talk.

TRANSMITTER

The transmitter PCB is a completely custom form factor that uses parts found in industry. Utilizing the same processor as in the receiver, the transmitter will have transmitting capabilities provided by the LoRa accelerometer data provided by the AIS3624DQTR microchip, barometric data provided by the MS560702BA03-50 barometric pressure sensor, which will all be stored on a microSD card. Using these modules, we found their schematics from the breakout boards and studied them to point where we could effectively implement their design to fit our needs. Using these we created an Eagle library from scratch to add these sensors and put them in a PCB. The accelerometer was designed using the component’s design on the data sheet. The accelerometer can withstand a maximum of 24-Gs and can receive data in all three axes. The AIS3624DQTR has a rating of ±24-G’s with the ability for both I2C and SPI protocols. The output frequency is ~500-Hz. The data pins are sent to the microSD portion of the PCB to allow for fast data logging. The barometric pressure sensor chosen is the MS5607-02BA03 and has a max pressure of 1,200-mbar (120,000-Pa). The device is also capable of both I2C and SPI protocols. The group concluded that this device has sufficient specifications to gather the necessary information. The data pins will run to the microSD portion of the PCB. To survive the high G forces related to rocket launches all components other than surface mount resistors and capacitors will be staked down to the PCB using NASA guidelines. In addition, the transmitter allows for up to four external analog inputs that can be user customized through software. This allows for future use of the PCB design with abilities to attach sensors as needed. To provide best battery life a custom power solution was devised. The research conducted resulted in a schematic design found and used it as a blueprint for the design of the power supply. The battery being used is a Lithium-Ion battery. This choice was made because of the power, capacitance, and its small weight. Another portion needed is a boost converter system to get the voltage up to 5-V. The device being used in the design is a MT3608 Boost Converter IC. The Boost Converter is a DC-DC Step-Up Converter where the input voltage limit is max 24-V and the output voltage being up to 28-V. Allowing for a smooth and uninterrupted supply to the receiver and transmitter PCBs. The Transmitter PCB has very small form factor with dimensions of two inches by four inches. This allows for the transmitter to be placed in the nose cone alongside other avionics.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Over the course of two years the bulldog rocketry committee has built and tested two major iterations of a low power telemetry system. Bulldog Rocketry's low power telemetry system meets demands set out by the team. The system tracks and records the rocket in flight and on the ground. The use of LoRa low powered radio modules has allowed for a small form factor and ultimate reliability while reducing cost and power consumption when compared to similar telemetry systems. This board will ensure future success of the rocket and bulldog rocketry. The electronics committee is working on new designs based on the LoRa system which will further increase range and improve the user interface.

Orthotropic Fin Flutter Analysis by Two Way Fluid-Structure Interaction

Team 74 Project Technical Presentation to the 2019 SA Cup

Mohammed M. Khan and Joel A. McQuaid
Ryerson University, Toronto, ON, M5B 2K3, Canada

INTRODUCTION

The purpose of this research is to investigate the flutter performance of the Ryerson Rocketry Club's Northern Lightning's supersonic fins through numerical modelling. The fin is comprised of a 12 ply HexTwo AS4C 3k Carbon Fiber weave with fiberglass scrim stabilizing threads, and alternating 90° and 30° ply angles. This analysis was performed using a two-way Fluid-Structure Interaction model within ANSYS. The fin laminate was modelled in ANSYS ACP as an orthotropic Finite Element Analysis model with material properties that were experimentally validated through tensile testing procedures as per the ASTM D3039 standard. The unsteady numerical model coupled a structural deflection solver to ANSYS CFX to calculate aerodynamic pressure loads, and fin laminate deflections. Based on the fin's cross-section, a flutter speed of Mach 4.5 ±0.25 was found, which is well outside the flight envelope of the Northern Lightning.

The SRAD supersonic fins designed by Ryerson Rockety Club (RRC) were successfully flight tested at both the 2017, and 2018 Competitons. During the two flight tests, the fin's performance was validated as it experienced speeds greater than Mach 1.53 without any structural or aerodynamic issues. To further improve on fin analysis, a numerical model was formulated to help predict the exact flutter points. To maintain accuracy in our solution, a two-way FSI model was formulated, and using ANSYS ACP, the 12-ply laminate was modelled to be fully incorporated into the structure as an orthotropic material.[1]

MODELLING METHODOLOGY

A Fluid-Structure Interaction (FSI) model is a method of coupling the solution process for a Finite Element Analysis (FEA) solver, and a Computational Fluid Dynamics (CFD) solver. With this coupling, it is possible to perform advanced aerodynamic testing of flexible structures. Using this CFD's solution file as the initial conditions, the FEA solver can take in the pressure loads to calculate the structural deflections and resulting mesh displacements. These mesh displacements are used to update the CFD computational grid to now model a deformed structure. The aerodynamic force loadings are recalculated by the CFD solver, and passed back to the FEA solver for the process to repeat. In order to achieve this within ANSYS Workbench, ANSYS CFX (Transient) module, and a Transient Structural module were utilized.

FEA AND CFD VALIDATION STUDIES

To save on computational time, two FEA models were meshed. Model 1 is the actual fin geometry, with tapered leading and trailing edges, whereas Model 2 is a fin with a rectangular cross section. Due to this, grid density in both models is significantly high in the thickness direction. A 12 ply laminate requires 12 elements across the fin's thickness direction. As a result, Model 1 has an even higher grid density because of the need for additional refinement at the ply drop-off locations (leading, and trailing edge tapers).

Based on these preliminary tests, the following two conclusions can be drawn. Model 2 was selected to be the model of choice to find the fin's flutter point initially. This is because the runtimes are 25% of what they are for the actual fin geometry (Model 1) as a result of the much coarser FEA grid. Once a flutter point is found, it can then be validated with Model 1 to ensure the speed is indeed correct. This saves a significant amount of time compared to iteratively running different flight speeds using Model 1. Secondly, regardless of which model is used, significant computer storage space is required to run either of the cases.

The validation procedure for the CFD model required some amount of simplification. It was performed by comparing the acquired drag coefficient values to those obtained from the FTM's Mach sweep. Since those models showed very good comparison to actual flight data when predicting rocket apogee, it has been assumed that the aerodynamic drag coefficients that were found were indeed accurate. The CFD model is employed only half of the Northern Lightning, and the computational grid was a 3D C-grid domain which allows for varying of the inflow velocity direction. This was done so that a specified side slip angle may be applied.

FEA AND CFD VALIDATION STUDIES

Using this model, a Mach number sweep was performed on Model 2. The steady state simulation at a 1 degree side slip was run followed by the initiation of the FSI model. As the Mach number was steadily increased, the oscillation profile showed a diverging amplitude. In testing it was found Model 2's fin laminate has a flutter speed of M4.5±0.25. Although the fin laminate was engineered to be exceptionally stiff which has been demonstrated here. In considering the fin joint between the laminate and the airframe, it is hypothesized that any fracture due to flutter will initiate here. This interface is the primary weakness of the RRC fin design. Thereby, the actual flutter speeds are expected to be in the range of M2 to M3 due to this interface failing first.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

The formulated two-way FSI model has proven to be effective in predicting orthotropic fin flutter speeds. This models FEA component was validated through a three-level grid refinement study using Richardson Extrapolation methodologies. Some fin material properties were able to be validated through ASTM D3039 composite laminate tensile testing procedures which showed good agreement with published values.[2] CFD model accuracy was also validated using actual flight data from the Arctic Thunder from the 2018 Spaceport America Cup. Assuming a rectangular fin cross-section, a flutter speed of M4.5±0.25 was determined. Actual tapered fin geometry could not be simulated as of yet due to computational limitations, however, the RRC team plans to expand this project to include this geometry in the future. This fin flutter speed is numerically accurate. Extensive experimental testing with a supersonic wind tunnel would be required to experimentally validate this result.

REFERENCES

[1] ANSYS Inc., "Northern Lightning ACP Model," Ryerson Rocketry Club, Toronto, 2019.

[2] "ASTM D3039 Standard Test Method for Tensile Properties of Polymer Matrix Composite Materials," ASTM International, West Conshohocken, 2019.

[3] L. P. Kollar and G. S. Springer, Mechanics of Composite Structures, Cambridge: Cambridge University Press, 2003.

[4] Cesaroni Technology, "CTI O3400 Motor Data," 26 10 2012. [Online]. Available: http://www.pro38.com/products/pro98/motor/MotorData.php?prodid=21062O3400-P. [Accessed 20 04 2019].

[5] OpenRocket, "Northern Lightning Flight Dynamics Results," Ryerson Rocketry Club, Toronto, 2019.

[6] J. D. Anderson Jr., Fundamentals of Aerodynamics, New York: McGraw-Hill, 2011.

[7] J. D. Anderson, Fundamentals of Aerodynamics, New York: McGraw-Hill, 2011.

[8] ANSYS Inc., "Northern Lightning CFD Model," Ryerson Rocketry Club, Toronto, 2019.

CFD Analysis with Turbulence Models of an Experimental Sounding Rocket

Team 77 Project Technical Presentation to the 2019 SA Cup

R. J. C. Saboya
Aeronautics Institute of Technology - ITA, São José dos Campos, São Paulo, Brazil, 12228-900

INTRODUCTION

The search for better measurements of aerodynamical parameters is crucial to apogee estimations and to verify stability, which relies on drag coefficient (CD) and static margin measurements. In order to calculate these values for the RD-09 rocket, the ANSYS Fluent® software was used to create computational fluid dynamics (CFD) simulations based on 2D axisymmetric (without cameras and fins) and 3D geometries of the RD-09, with respect to mach number, attack angle, and atmosphere conditions expected at Spaceport America launch, using Reynolds Averaged Navier Stokes (RANS) models to simulate boundary layer and viscosity effects. The primary objectives of this study are to validate semi-empirical data calculations and to introduce CFD methods to the team, as it was already confirmed in 2018 an underestimation of drag force by Missile Datcom results in comparison to wind tunnel tests at subsonic speeds using Prandlt-Glauert corrections.

GEOMETRY

The first step to create a CFD simulation is to conceive a geometry that will be the fluid space, for that to happen, the first models introduced for mesh generation were created using drawings and geometry projections of the RD-09 CAD model assembled on the 3D experience platform, the objective here was to create geometries that can be tweaked to create the fluid domain, which will be used to implement finite volume methods (mesh generation), however, regions at the boundaries of the rocket surface should be treated with extra care, with any minor details being simplified in order to reduce errors during the meshing phase.

MESHING

One of the most time-consuming parts of the study, the meshes for each 2D/3D geometries were evaluated in terms of aspect ratio, orthogonal quality, and skewness: the first one shows the size ratio between the larger and smaller dimensions of a cell, the second and third ones determines the shape quality of the cells. The presence of low quality elements need proper correction, otherwise the CFD solver will not produce correct results.

Another critical aspect of mesh generation involves sizing correctly each section of the fluid domain, which means smaller cells at specific places, specially at boundary layer regions, where turbulence model are dependent of high discretization. For every model tested, near wall mesh size were tested in terms of a dimensionless wall distance, called y plus (y+), with a target value of[1], in order to resolve properly the velocity gradients due to shear stresses.

For 3D models, the rocket surface also needed refinement, and the most required areas of attention were the fins and cameras. Three dimensional models demand more time to be created, and have higher chance of creating numerical divergences, therefore less models were conceived.

CFD ANALYSIS

The main objectives regarding the simulations were to obtain drag estimations for mach numbers from 0.053, which is the expected launch rail departure at Spaceport weather conditions, up to mach 0.8, the expected maximum velocity during flight, and to validate the results, a 2D model of the 2018 competition rocket, RD-08, were simulated with the same conditions created during wind tunnel tests. For RD-09, two different RANS methods were used, Spallart-Allmaras[1] and Transition-SST[2] models, in which, in simplified terms, the main difference between them relies on boundary layer development and viscous effects, and the results demonstrate increased accuracy of the drag coefficient using the latter, suggesting that the transition model better describes the flow around the rocket. Another innovative aspect was detecting transonic regions closer to the fins at high subsonic speeds, related to diameter increase at the SRAD motor casing, where velocities went up to mach 0.95 at the maximum velocity, however, these fenomena do not interfere the flow located at the fins leading edge due to dissipation effects. The software also preditcted values of pressure center locations for 3D models, showing proximity with results from semi-empirical softwares.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Since apogee estimations are highly dependent on the total rocket drag, the work presented provides a more exact solution for CD, and coupled with flight mechanics, the team is able to calculate the apogee with minimal error margins. Further simulations will be made also considering the nozzle plume and its effects on RD-09's aerodynamics, and for future works the team can validade the CFD results using wind tunnel tests at ITA aerodynamics laboratories, having the advantage of using subscaled models and without the need of previous flight test for next competitions.

REFERENCES

[1] RUMSEY, C., "The Spalart-Allmaras Turbulence Model" [online database], URL: https://turbmodels.larc.nasa.gov/spalart.html [cited 20 may 2019].

[2] RUMSEY, C., "The Langtry-Menter 4-equation Transitional SST Model" [online database], URL: https://turbmodels.larc.nasa.gov/langtrymenter_4eqn.html [cited 20 may 2019].

Apogee Precision With Airbrake Control and SRAD Rocket Simulator

Team 90 Project Technical Presentation to the 2019 SA Cup

H. Dragesund, T. Haddal, P. Holmqvist, H. Løvik, E. Rydningen, M. Sunde, B. Tapley, T. Winter
Norwegian University of Science and Technology (NTNU), 7031 Trondheim, Norway

INTRODUCTION

One of our main goals in the Spaceport America Cup is to achieve an apogee of 10’000 ft as precisely as possible. This goal will be achieved through a SRAD rocket simulator, a discrete PI controller and actively controlled air brakes. The simulations provide the reference data that the PI controller together with a Kalman filter uses to dynamically correct towards the optimal trajectory in real-time. To verify the control algorithm, we have developed a system that runs the flight computer with data from the SRAD rocket simulator. The following abstract describes how the CFD analysis, stochastic variables and Monte Carlo simulations are used to produce a normal distribution for the optimal deployment of the air brakes.

RESEARCH OVEVIEW

Air brakes are implemented to control the apogee of the trajectory of the rocket. The air brakes are positioned right above the motor, and the breaking surfaces extend out of the rocket body perpendicular to the flight direction. All three breaking surfaces will be extended simultaneously, equally, and symmetrically. The required area of the breaking surfaces was found by first calculating how much the rocket would overshoot without air brakes, and then calculating how much the drag coefficient would have to increase after burnout to reach exactly 10’000 ft. The air brake control system needs input on when to break, as well as a lookup table as a reference. To provide this essential information, we made an SRAD rocket simulator, named Penumbra. Given a weather forecast on the day of interest, Penumbra will provide the optimal air brake deployment time with the corresponding trajectory. The program is based on aerodynamic forces given by CFD data, rigid body equations, thrust data, and Monte Carlo simulations. Because there are large uncertainties in empirical formulas for drag in the compressible and transonic flow region, we decided to do a thorough 3D CFD analysis of the rocket. The CAD model of the rocket exterior was meshed carefully, and run in ANSYS Fluent to simulate drag, lift and moments for combinations of AoAs and velocitie of interest. These forces were interpolated and used in the rigid body equations. A version of the CAD model with the air brakes deployed was also analyzed in the same manner. Although this is a considerably more time-consuming and difficult task than using empirical formulas, we are confident it will provide a significantly more accurate trajectory. The CFD modelswere tested for the lowvelocity flowregime, by conducting a wind tunnel test of the rocket atNTNU. To simulate the trajectory and the time evolving configuration (COM, COP, mass, etc.) of the rocket, we need to obtain the equations of motion given all the forces acting on the rocket. The rocket is considered as a rigid body with 6 DOF, and we may therefore apply the rigid body equations (Euler’s equations of motion) to derive the appropriate equations of motion given all the forces acting on the rocket. Penumbra attempts to predict the trajectory of a rocket given appropriate input such as its geometry (class 1) or CFD/CAD data of that rocket (class 2), and an initial condition such as an initial position, an initial orientation in space or wind data, if available. The output of the program is a complete overview of the kinematics of the rocket during flight such as a plot of its trajectory along with the corresponding time evolution of translational and angular data. Time evolution of rigid body characteristics such as the mass, the center of mass or the center of pressure is also available. One particular feature in Penumbra that we will continue to develop is the stochastic approach on trajectory calculations. The thrust data, aerodynamic forces, and wind data are all made stochastic. With this approach, we can provide simulations that can emulate the actual trajectory more realistically by simulating many different outcomes of a given rocket launch, given wind profiles at the launch site. The simulator uses the shooting method to determine whether or not it should deploy the air brakes. It does this each time step of the simulation, to find the optimal time to deploy. As many of the variables are kept stochastic, Monte Carlo simulations are performed, which in the end produces a normal distribution about the optimal air brake deployment time. The mean of this is then used in the control system as the optimal trajectory. Our control system utilizes the height measurement from a barometer and an acceleration measurement from an IMU. The controller is a discrete PI controller that tries to follow a reference velocity that we get from the optimal trajectory. Because velocities are not easily measured directly, we have implemented a Kalman filter to estimate this. A Kalman filter is an optimal observer which fuses different measurements and a mathematical model of the system in a theoretically optimal way. The mathematical model is simply based upon that the height is the velocity integrated over time, and velocity is the acceleration integrated over time. The measured height and acceleration are sent to the Kalman filter as a measurement and actuation signal respectively. Test launches of small-scale rockets were conducted to obtain barometer and IMU data which we used to tune the parameters of the Kalman filter. The Kalman filter outputs estimates for height and velocity. The height estimate is used in a lookup table which outputs the optimal reference velocity. This reference is then subtracted from the estimated velocity from the observer, and the result is used as error in the PI controller. The lookup table represents the optimal trajectory calculated by our simulations in a compact and efficient manner. To index the lookup table we use the altitude of the rocket. The output is the optimal velocity for the given height. The reason for using a lookup table is the ability to do most calculations before the flight, minimizing the flight computer resources used by the control system. This is especially important because of the high velocities. The average cycle time when the control algorithm is running on the flight computer is 5 ms which means that we can calculate the air brakes control signal every meter even at maximum velocity. To verify expected functionality from the control algorithm we have developed a system that lets us run the fligh computer that we will use in the rocket with data from Penumbra. While the simulation is running on a separate computer, data is sent via serial communication to the flight computer where it perceives the data as sensor data. One package of data is sent after each numerical iteration in the simulation. A timestamp representing how long Penumbra has simulated is also sent with the data package. This timestamp is used to sync the two computers so the operationa time at the flight computer is in sync with the data it receives from Penumbra. After each iteration the flight compute sends its rocket state data alongside with the air brakes control signal back to the PC where we can verify that the flight computer perceived the simulations and behaved in a rational manner. The data can be displayed both in real time and after a full simulation.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

This developed system and approach is beneficial in many ways. First of all it lets us tune our controller based on simulations. We also get some verification that our logic isn't flawed by observing that the control algorithm manages to follow the optimal trajectory using various simulations. The benefit of executing on the hardware is also that we get to run the control system with the right clock frequency and memory specifications reducing the possibility of unexpected behaviors.

Optomising Composite Rocket Fin Laminate Design Using Non-Standard Ply Angles

Team 98 Project Technical Presentation to the 2019 SA Cup

Rupert Glover
University of Bath, Bath, United Kingdom,BA2 7AY

INTRODUCTION

Rocket stability is generally defined by the locations of the centre of gravity and centre of pressure. The competition rules stated the centre of gravity was required to be at least two times the diameter of the rocket above the centre of pressure, known as the stability margin. At the very aft of the rocket, the fins are a significant contributor to the position of the centre of gravity. Hence, an overdesign in the thickness of the fins may add unnecessary weight to the bottom of the rocket, bringing the centre of gravity downwards, and therefore reducing the stability. While this can be counteracted by placing weight near the nose, an overall net increase in the weight of the rocket requires a larger motor and is generally an inefficient design compared with reducing the weight of the fins. Factoring in the potential safety risks of not properly modelling the failure modes of a composite, the previously proposed design leaves significant room for improvement. The aim of this report is therefore to better understand the dynamic and static requirements of a sounding rocket stability fin, and to produce an accurate model which can predict the response of a composite layup to these requirements. From this, it is hoped a more efficient design can be generated which will optimise the weight of the fins. The investigation of this report was not intended to produce physical components for use on the 2019 launch vehicle due to budget and time constraints, however it is hoped the methodology could be used by subsequent teams in their designs.

LITERATURE REVIEW

Manufactured composites are a material which has seen extensive use within the aerospace industry due to their ability to be fine-tuned towards stiffness requirements. However, while a plethora of information is available from academic resources on the topic of static and dynamics constraints regarding composites, often it has been found that these constraints have been considered separately. Notably, it has been seen that angle plies will increase the flutter speed due to their torsional response [1, 2, 3, 4, 5], however as lift and drag tend to act orthogonal to the free-stream it may be argued that 0° and 90° may be more appropriate for static loading [6, 7, 8, 9, 10, 11]. The balancing between these two cases presents an interesting area for optimisation.

METHODOLOGY

The analysis followed three stages; building of a computation model to predict dynamic and static behavior, empirical analysis to validate the models, and a final round of optimisation using non-standard ply angles (angles other than 0°, ±45°, and 90°).

The static model used classical laminate theory to calculate the predicted bending of the laminate under lift and drag, and this could be used to find the expected stresses in each ply. These stresses could then be evaluated against three failure criteria; Max Stress, a simple non-interactive method which indicated failure mode; and Tsai-Hill and Hoffman, two interactive methods which consider multi-axis loading.

The dynamic model was based on an aroelastic method proposed by Wright and Cooper, [12] which uses strip theory and simplifies the motion of wing's pitch and lift to a panel which is fixed at its root chord by two uncoupled, orthogonal, rotational springs. A model developed by the author took the effective stiffness of a given layup to evaluate the instability speeds the wing. However, it was predicted that this would likely not fully reflect the behaviour of a laminate and hence a more sophisticated model was provided by a doctor of composites at the University of Bath which calculated the instability based on the strain energy of a laminate. The static and dynamic models were used to produce a suitable laminate for empirical validation.

Static loading was validated with a simple deadweight static load test, with the wing's root edge clamped to the edge of a work top and increasing loads suspended from the tip edge. The response of the wing was compared against expected deflection and elasticity, and also compared against failure loads. Only the lift component of the load was tested as drag was considered negligible compared to lift. The dynamic testing was performed using a shaker table and laser vibrometer, conducting a sweep through driving frequencies to identify the natural frequencies and mode shapes. These natural frequencies were then compared against the calculated frequencies at 0m/s wind speed in the computational model. The aeroelastic response could not be tested over the full range of wing speeds at which the launch vehicle travelled (0 to 275m/s) as there was no wind tunnel available to replicate these speeds.

The final round of development used the previously produced computational models to investigate the possibility of non-standard ply angles. A parametric study was conducted, iterating through various angle combinations toidentify general trends. These observations were then used to simply the problem space, such as requirements of certain angles in specific plies, to reduce the number of variables to an acceptable amount for a brute force optimization and generate an optimum laminate for weight.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

A model was successfully produced to predict the static loading, with a strong correlation between the model's results and two empirically tested fins. Two separate dynamic model were used, one model based on the effective stiffnesses and one supplied model based on strain energy. The former was unable to model the bend-twist coupling which was critical in composite design, and while the latter was more promising in its modelling of bend-twist the calculated natural frequencies could not be validated against empirically observed values.

The results concluded how a fin could be optimised. Specifically, 0° plies are almost always a requirement on the outer layer due to large bending moments, while angle plies are preferred in central plies to provide torsional stiffness and resist dynamic failure. Positive angles produced better dynamic behaviour, however this was limited by a transition from flutter to divergence, with the latter being enabled by positive angles. The number of plies was reduced from 8 to 6 by using non-standard ply angles. The optimum layups were [0/α/-α]s, where α could be between 8 and 13°, or [α/-α/-β/β/-α/α] where α could take a value between 4° and 7°, while β was preferred as either 90° or 0°. Divergence was the primary dynamic limiting factor for these specific layups due to the presence of positive angle plies, however in general flutter has occurred before divergence in almost all other layups.

The investigation has satisfied the majority of the original goals, and it is hoped that the results will meaningfully contribute to wider academia. While not all of the computational models could be validated in the investigation, the results present trends which correlate with most of existing literature and provides a methodology as a foundation for future years to develop and improve apon. Some improvements for future investigation include empirical analysis of the full range of airspeeds. Additionally, an early aim of this project was to optimise the fastening layout of the fin, as the high number of bolts added significant weight to the design. This optimisation was not completed due to time constraints of the project. It is likely that it would not be a complex investigation as the main loading case was a bending moment whereas the main concern with fastening of composites is a direct tension or compression load.[9 13] However, it still presents an extra dimension for optimisation. In particular as plies orientated orthogonal to the load, or at 90° in the scenario, are best for mechanical fastening and these angles have largely been neglected in the analysis.

REFERENCES

[1] T. A. Weisshaar and B. L. Foist, "Vibration tailoring of advanced composite lifting surfaces," Journal of Aircraft, vol. 22, no. 2, pp. 141-147, 1985.

[2] T. Niblett, "Divergence and Flutter of Swept-Forward Wings with Crossflexibilities," Royal Aircaft Establishment, 1980.

[3] G. A. Georghiades, S. J. Guo and J. R. Banerjee, "Flutter characteristics of lamianted composite wings," Journal of Aircraft, vol. 33, no. 6, pp. 1204-1206, 1996.

[4] S. J. Guo, J. R. Bannerjee and C. W. Cheung, "The effect of laminate lay-up on the flutter speed of composite wings," Journal of Aerospace Engineering, vol. 217, no. 3, pp. 115-122, 2003.

[5] O. Stodieck, J. E. Cooper, P. M. Weaver and P. Kealy, "Improved aeroelastic tailoring using tow-steered composites," Composite Structures, vol. 106, pp. 703-715, 2013.

[6] S. W. Tsai and H. T. Hahn, Introduction to Composite Materials, Technomic Publishing Company, 1980.

[7] M. W. Hyer and S. R. White, Stress Analysis of Fibre-Reinforced Composite Materials, Boston WCB/McGraw-Hill, 1998.

[8] R. Talreja, "Assessment of the fundamentals of failure theories for composite materials," Composite Science Technology, vol. 105, pp. 190-201, 2014.

[9] M. C.-Y. Niu, Composite Airframe Structures: Practical Design Information and Data, Hong Kong: Conmilit Press, 2010.

[10] A. C. Orifici, I. Herszberg and R. S. Thomson, "Review of Methodologies for Composite Materials Modelling Incorporating Faliure," Composite Structures, vol. 86, no. 1-3, pp. 194-210, 2008.
[11] K. Rohwer, "Predicting fiber composite damage and failure," Journal of Composite Materials, vol. 49, no. 21, pp. 2673-2683, 2015.
[12] J. R. Wright and J. E. Cooper, Introduction to Aircraft Aeroelasticity and Loads, John Wiley, 2007.
[13] J. R. Vinson, "Mechanical Fastening of Polymer Composites," Polymer Engineering Science, vol. 29, no. 19, pp. 1332-1339, 1989.

Design and Evaluation of a Supersonic Pitot Tube for Velocity Verification of Rocket Systems

Team 107 Project Technical Presentation to the 2019 SA Cup

Space Cowboys at Mississippi State University
Department of Aerospace Engineering, Mississippi State, MS, 39762

INTRODUCTION

Over the last few decades, the falling cost of composite materials have allowed for the wide scale adoption of composite structures within the aerospace industry. Composite structures have allowed major structural elements to become thinner and lighter, enabling aerospace systems to achieve higher performance statistics than traditional materials would allow. One area that has seen vast performance increase is in small, sounding rockets, especially at the collegiate level. This increase in performance has led to more collegiate rockets operating within the supersonic regime. Being able to accurately measure velocity has become increasingly more challenging. Traditional methods, such as accelerometeretric or barometric measuring systems can lead to compounding errors, greatly reducing accuracy. These velocity errors are acceptable for subsonic flight, as it is possible to correct for those errors post flight. Unfortunately, at supersonic flight this is not possible, and flight velocity during this time is usually found through flight simulations. A supersonic pitot tube has the potential to greatly improve the accuracy of determining supersonic flight velocity and as well as providing data that will influence future sounding rocket designs.

PITOT TUBE PLACEMENT AND PACKAGING

As with any flight system, a Pitot tube must be packaged in a way which reduces weight as well as takes up minimal space. These criteria must be met whilst allowing the Pitot tube to have access to its surrounding environment. The two main environmental elements the tube must measure are stagnation pressure and static pressure. In order for the unit to have access to the stagnation pressure, it must be open to oncoming flow. Therefore, there are two potential locations for a Pitot tube on a rocket: protruding from the side, or mounted in the nose cone.

Many aerospace vehicles incorporate a side-protruding Pitot tube. This setup allows the system to have access to virtually un-spoiled airflow, whilst also being open to the static pressure of the environment. In a rocket however, this method presents many problems. The main issue, particularly for smaller scale collegiate rockets, is aerodynamic drag. Due to the fact that collegiate rockets have limitations in their allowance for flight correction whilst airborne, they are required to be as streamlined as possible so as to not vear off course. Therefore a side-mounted pitot tube to an extent is out of the question.

Nose cone based pitot tubes on the other hand are much more welcoming for a system such as a rocket. The main benefit of incorporating a system such as this into a nose cone is its design streamline. Rather than protruding out of the side of the rocket into the flow, this variation allows the Pitot tube to be located in the center of the flow, whilst not causing a side drag force. One difficulty to this method is the means by which the system measures static pressure. Since the tube unit is not located off the side of the rocket, it does not have direct access to passing flow and therefore static pressure. This issue can be overcome by simply creating a small hole on the side of the rocket through which a sensor can measure static pressure. All measurements of stagnation pressure and static pressure can be recorded by means of pressure sensors and stored in a main computer located centrally in the rocket body.

A shortcoming caused by placing the Pitot tube in the tip of the rocket is its induced drag. As expected, by placing an open device in the middle of a fluid stream, there is a drag force created. This force will not only hinder the rocket's acceleration capabilities, but it can also create structural strain on the rocket nose. This effect is particularly an issue in the subsonic flight realm of a rocket. As a rocket travels supersonically, the oncoming flow creates a shockwave at the nose which essentially acts as a fluid nose cone. This effect allows the Pitot tube to slip easily through the air as the rocket travels. However, as the rocket approaches the speeds within the subsonic realm, this shockwave effect is not present. Rather, the pitot tube causes aerodynamic drag at an increasing rate with the rocket's speed. This generally may not be a large issue for collegiate rockets as well as others. Depending on the specific design, a rocket will progress through the subsonic velocity range fairly quickly before spending the duration of the flight supersonic.

An additional issue found in packaging a nose-mounted Pitot tube, is in transferring the recorded data to the computer. For the flight system to read the sensor outputs, the stagnation pressure readings must first go from the tip of the tube down the cone to the flight computer. While this issue is not one of great disturbance, the problem comes in measurement error. While the flow is being transferred from the tip, drag within the tubing may induce vortices or pressure differentials which in turn could affect pressure readings. This problem however can be overcome by means of a flight computer located higher within the tube.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Much of the nose cone's shape parameters were predefined constraints, except for the nose tip. The nose tip had to maintain the conical outer profile, and had to be between 2 and 2.25-inch long in order for the simulation to maintain a + 2% [of apogee] agreement with the rocket's simulated performance. Within those constraints the base tip was modified to include a continuous and smooth passage for the air. The passage goes from the leading edge of the nose tip to the interior of the nose cone. This allows for the stagnation pressure to be taken within the controlled environment of rocket's interior. The nose tip designs were both evaluated at a velocity of Mach 1.5 at standard sea level conditions: a temperature of 293-K, and an ambient pressure 101325-Pa. Using the base nose cone as the standard design, the performance changes between these- two nose cones are as follows: the open tip design shows a -0.3221% difference in model temperature and a 6.546% decrease in Y-direction normal drag force. The change in model temperature is negligible and can safely be ignored. The 6.546% difference in drag force is significant enough to cause changes to rocket performance; however, the drag force is very dependent on individual test coupons, so real world testing is required to properly evaluate how the pitot tube will affect drag force.

A pitot tube is a system which shows promise as method of calculating velocity of collegiate rockets that operate within the supersonic regime. At supersonic flight, the pitot tube system will provide an independent, velocity data set that can be correlated with other more traditional flight speed measuring devices. This additional data set will allow for increased accuracy of rocket airspeed while within supersonic flight. In order for this pitot tube system to be useful for many rocket systems, it must be able to be used and integrated easily and quickly into existing nose cone designs and predefined rocket performance calculations. Special design considerations must be made when designing the open tip nose to not affect performance. Analysis shows that it is possible to design a pitot tube tip whose performance parameters are close to a given nose cone and that fits the other design limitations of a pitot tube. Moving forward it is important to begin producing test coupons for the pitot tube system and doing real world evaluations. These real world evaluations will allow for a better understanding of how manufacturing variance affects the performance of the pitot tube as well as granting insight to the limitations of this design.

REFERENCES

[1] "Form 2 Tech Specs," Formlabs Available: https://formlabs.com/3d-printers/form-2/tech-specs/.

[2] Hall, N., "Crossed Shock Waves," NASA Available: https:///www.grc.nasa.gov/www/k-12/airplane-crosshock.html

[3] Hall, N., "Normal Shock Waves," NASA Available: https://www.grc.nasa.gov/www/k-12/airplane/normal.html

[4] Hall, N., "Oblique Shock Waves," NASA Available: https://www.grc.nasa.gov/www/k-12/airplane/oblique.html

[5] Hall, N., "Flow Past a Cone," NASA Available: https://www.grc.nasa.gov/www/k-12/airplane/coneflow.html

[6] Solidworks Flow Simulation, Solidworks, Ver. 2018. Dassault Systemes, 2018

The Aries V Dual Liquid Propellant Rocket Engine

Team 112 Project Technical Presentation to the 2019 SA Cup

Donald Rosier III
Colorado State University, Fort Collins, Colorado, 80521

INTRODUCTION

Sounding rockets play an important role in atmospheric research and experimentation in sub-orbital space. A major component in determining the performance and abilities of these rockets are their motors. Innovations in rocket motor design can propel sounding rockets to new heights in their abilities to collect meaningful data. The Aries V rocket utilizes a dual liquid propellant engine to launch a carbon composite airframe carrying a payload of scientific merit to an apogee of 10,000-feet. The engine design was selected due to the liquid propellant's higher energy density over solid or hybrid types and to bring recognition to Colorado State Universities abilities in innovative liquid propellant motor design.

DESIGN OVERVIEW

The Aries V dual liquid propellant engine utilizes nitrous oxide oxidizer and ethanol fuel during the liquid phase of combustion to produce thrust. The liquid propellant design was selected because of the overall challenges to develop a robust platform. Nitrous oxide was selected as the oxidizer due to the pressure and temperature at which it vaporizes. This behavior means the entire system can be pressurized by the oxidizer itself and eliminates the need for turbopumps. This drives the oxidizer and, with the help of a piston, the fuel out of the propellant tanks, through a custom plumbing system, and into the combustion chamber. These tanks are mounted coaxially and are separate from the combustion chamber to dampen any vibrations experienced during flight and mitigate the risk of pogoing during flight. A propellant delivery system was developed this year to control the flow of the propellants into the combustion chamber, which should mitigate the pressure spike seen at the beginning of the burn in the combustion chamber.

To ignite the motor, a small block of solid rocket propellant, a preheater, is ignited to prime the combustion chamber to a high temperature prior to injection of liquid propellant. This is necessary because nitrous oxide and ethanol are non-hypergolic by nature and need an ignition source. A pyrobolt driven actuator is activated to initiate flow of liquid propellants through the propellant delivery system where it enters the top most portion of the combustion chamber, the injector. The injector serves to mix the oxidizer and fuel just before entering the combustion chamber. The Aries V motor utilizes a pintle style injector where fuel is injected radially from six orifices in a pintle where it impinges on the flow of oxidizer to provide efficient and thorough mixing as well as ensure motor efficiency. Once the propellants are mixed and enter the combustion chamber, the heat and flame from the preheater ignite the liquid propellant.

The liquid rocket engine, designated as ALRE-3, will produce an average 500lbs of thrust during the 8 second burn, and the peak thrust, which is 550lbs, would be seen at the beginning of the burn. This will be beneficial because the rocket will be traveling at a safe velocity off the rail, meaning that the rocket would achieve the desired 10,000ft AGL apogee.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

The innovations made on this year's rocket were greatly guided by the Aries III & IV program's rocket design and performance at the IREC. The experiences at competition last year has allowed Colorado State's IREC team to identify and eliminate design weaknesses and develop an innovative and optimized engine design. Although a flight test of the liquid motor has not been able to be conducted, static ground motor fires have provided verification to the design's effectiveness and reliability. Success of this propulsion system at this year's IREC will open new doors in Colorado State University's abilities in atmospheric experimentation and in the schools overall standing in academically driven rocket design.

Design, Analysis, and Construction of a Rocket Fin Using Custom Multi-objective Optimizer

Team 119 Project Technical Presentation to the 2019 SA Cup

Emanuel Raad, SaiPrashanth Ramesh, Dino Odobasic, Ben Clarke, and Austin Cooper
University of Windsor, Windsor, Ontario, N9B 3P4, Canada

INTRODUCTION

Designing a rocket is a careful balance between stability and performance. The fins serve the purpose to stabilize the rocket at the cost of increased drag. OpenRocket1 is an existing tool that can be used to size the planform of the fin. Given a value for the the span, root chord, tip chord and sweep angle, the software can output the apogee and minimum stability of the rocket. It is of interest to find the fin geometry set that will result in the most stable rocket without sacrificing aerodynamic performance. Although OpenRocket comes with an optimization tool, it was not deemed sufficient for the task, since it does not support multiple constraint functions or multiple objectives. A custom optimization script was written in Python to extend the functionality of the software. Another area where OpenRocket falls short is the characterization of the coefficient of drag of a fin with an airfoil. To find the airfoil best suited for the mission, a computer simulation was conducted to compare the performance of the diamond airfoil to the biconvex airfoil. With the fin planform and airfoil determined, it is now possible to construct the fin. Several methods of construction were investigated, with a focus on inexpensive manufacturing. The manufacuting techinique should be repeatable, since any difference in one of the fins can make the rocket unstable.

PLANFORM SIZING: A MULTI-OBJECTIVE OPTIMIZATION PROBLEM

The optimization problem was defined using two desirability functions[2], one for the apogee requirement (9144m) and one for the stability requirement (1.5 body calibers). The value of the desirability function approaches unity as the parameter approaches the desired quantity, and the value decreases as the parameter undershoots or overshoots the desired quantity. Next, any number of constraints can be defined. This is done to eliminate unfeasible geometries from the design space. For example, the tip chord is constraint to be smaller than the root chord.

Now that the problem is defined, it needs to be solved. A genetic algorithm was chosen for the problem. The input to the algorithm is a fin geometry set. That geometry gets passed to OpenRocket, which acts as a black box that then outputs the apogee and minimum stability. From these results, the next generation of fin geometries is created. Due to the multi-objective nature of the problem, there is no one absolute solution. Instead, the solution is a pareto optimal front, where certain combinations of the fin planform geometries favor high apogee-low stability designs, and other combinations favor low apogee-high stability designs. It is up to the designer to pick one geometry amongst the pareto optimal for the mission, based on the material and the maximum flight velocity.

DETERMINING AIRFOIL SHAPE THROUGH COMPUTER SIMULATIONS

Two candidates were explored for the fin airfoil: the biconvex airfoil, and the diamond airfoil. The optimum design of the fin airfoil was determined using a Computational Fluid Dynamics software, Ansys FLUENT. The $K-\omega$ SST turbulence model coupled with the density based solver were employed to obtain coefficients of drag. The simulations were run at two representative Mach numbers, Mach 1.4 and Mach 0.4. The airfoils' mesh was optimized by a number of ways. This entailed using a greater concentration of divisions on the walls on the rounded leading and trailing edges, the addition of inflation layers and the overall reduction in the size of mesh elements. A standard of y+ <5 was maintained in all airfoil simulations. This ensured the successful capture of the viscous sublayer on the wall boundary layers. Capturing the viscous sublayer produced sensible coefficient of drag results. Convergence of the residuals, coefficient of drag, and volume integrals were monitored to ensure that the results were accurate. The results from the CFD experiment suggested that the coefficient of drag of the diamond airfoil in the supersonic regime is nearly 33% lower than that of the biconvex airfoil. This resulted in the final decision of the diamond airfoil being chosen for the mission.

CONSTRUCTION

A two part female mold is constructed using a CNC machine. Carbon fiber is laid onto each half of the mold, and then the air is evacuated with a vacuum. After the two halves are made, additional sheets of carbon fiber and more epoxy is applied then the mold halves are bolted together. The pressure bonds the two halves with the new sheets to form a whole fin. When the part leaves the mold, the edges need to be trimmed and sanded. This process is repeatable and ensures that the diamond airfoil is captured each time, since the pressure of the molding process pushes the fibers into the contour of the mold. Alternative inexpensive methods of construction were also considered, but they were less successful due to the small thickness of the fin. The construction of a male mold through additive layer manufacturing was attempted, but the printer did not capture the diamond profile correctly. Next, a male mold made from a foam cutter was constructed, but the foam was too fragile to use for laying composites.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

This paper outlined the design, analysis and construction of a fin for use on a high powered rocket. The initial sizing was determined through a custom-written optimization script that uses OpenRocket as a blackbox for the simulation data. Given an input fin geometry, the blackbox returns flight data such as the apogee and the minimum stability. Then, based on constraints and objectives, the optimization script will find a set of fin geometries that satisfy the multiple objectives. From there, a computation fluid dynamics experiment was conducted to find the airfoil that produced the least amount of drag. The diamond airfoil and the biconvex airfoil were compared, and it was show that the diamond airfoil has superior performance for this specific mission. The fins were finally constructed out of carbon fiber using a female mold and vacuum bagging. In future work, the optimization software can be extended to take into account the effect of the geometry on the structural integrity of the fin. For example, fin geometries that exasterbate flutter would be heavily penalized in the algorithm. The same script can also be extended to optimize other aspects of the rocket geometry, such as the length of the nose cone and the boat tail. With the fins constructured, they can be subject to non-destructive tests to validate the analysis conducted. For instance, a torsion test should be done to determine the shear modulus. Data from the test flight and the IREC flight can be used to calculate the coefficient of drag of the rocket, and compare with the simulated values. he innovations made on this year's rocket were greatly guided by the Aries III & IV program's rocket design and performance at the

REFERENCES

[1] Niskanen, S., "OpenRocket," Software Package. Ver 15.3, 2015.
[2] NIST, Engineering Statistics Handbook, 2012, Chap. 5.

Design of a Hig Power Sounding Rocket to Reach a 150,000 Foot Apogee

Team 200 Project Technical Presentation to the 2019 SA Cup

Chris Gilroy, Hawii Boryio, Wes Johnston, Roman Rositani, Brian Johnstone
Oregon State University, School of MIME, Corvallis, OR, 97504

INTRODUCTION

The Oregon State University (OSU) 150k High-Altitude Rocket Team (HART) is a student-led, two-stage competition rocket team that is a part of the OSU AIAA branch, founded in 2012. The 2018-2019 OSU HART consists of 13 mechanical engineers, 3 electrical engineers, 2 computer scientists, and 4 chemical engineers. Each of these members is of senior standing in their respective majors. This team is most proud of using a 4-inch diameter on both stages (4-4). The first OSU HART, which worked on their rocket during the 2016-2017 school year, used a 3-inch sustainer diameter and a 4-inch booster diameter (3-4). This rocket flew to an apogee altitude of 80,000-ft in Blackrock, Nevada. The second OSU HART worked on their rocket during the 2017-2018 school year and used a 4-inch diameter on both stages, hoping to reach a higher apogee altitude than the previous team. However, that team had a catastrophic failure during testing and were unable to launch as planned at the 2018 Spaceport America Cup. The third and current OSU HART, which has been working on the project since September 2018, plans to improve the designs of the previous OSU HART and launch at the 2019 Spaceport America Cup. If successful, this team will be the first OSU rocket team to successfully launch a 4-4 rocket.

PROPULSION

Propulsion consists of two student researched and designed rocket motors. They are ammonium perchlorate composite propellant motors with approximately 17,000-Ns of impulse. The challenges with making the rocket motor a 4-4 design is that it could have trouble separating. To combat this we have attempted to change our grain geometry to use two Bates and four pseudo-finocyl grains. This will shorten the burnout time of the booster, which will allow for less time for the sustainer to drift off course after separation. It is necessary for the second-stage of the rocket to produce a large initial thrust. In the span of about .25-seconds, 700-pounds of thrust will increase the velocity of the rocket and surpass the speed of the booster stage.

AERODYNAMICS AND RECOVERY

The 4-4 design affected the aerodynamic performance of the rocket by decreasing the overall drag from having a uniform diameter going down the rocket's length compared to a design with different diameters between the stages. This design affected the recovery system by allowing the packing density to be the same in each stage, as the parachutes were of similar size and had the same volume of space in the body tubes to use. This allowed for uniformity in parachute preparing, making the designs and packing easier for the team.

STRUCTURES

A crucial design aspect regarding a rocket with equal diameter booster and sustainer stages came with the interstage coupler. The settled upon design had two sections. The lower portion would slide into the interior of the booster airframe, while the upper portion allowed the sustainer motor casing to slide in. This design allowed for a sleek, nearly flat coupler which could firmly join both stages while maximizing aerodynamic efficiency. It was manufactured out of anodized aluminum to minimize the risk of galling during the separation events. Reinforcements of the airframes was also done to maximize the likelihood of mission success. During separation and recovery events, the airframe is under immense strain. The structures team worked to mitigate this by reinforcing the motor casing attachments and the upper ends of both airframes with additional layers of carbon fiber and kevlar.

AVIONICS

The Avionics sub-team is responsible for ensuring the staging and recovery events of a two-stage, solid fuel rocket. This included the creation of black powder charges for staging and recovery events, the design and manufacturing of electronics housing bays, and the selection and implementation of flight computers to ensure successful separation, parachute deployment, and motor ignition. The sub-team decided to use 4 gram primary staging charges and 4.5-gram secondary charges. The avionics bays are one-piece, CNC machined Delrin. The primary commercial flight computers are the Altus Metrum TeleMega v2.0; the redundant flight computers are the EasyMega v2.0. The bays also house student developed altimeters that will transmit telemetry to the ground; this telemetry will be filtered and displayed by a student developed Kalman filter and ground station user interface.

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Overall, the 4-4 design has improved the rocket in two aspects. The first is expected apogee, as the 4-4 design rocket has an expected apogee of 114,000-ft, while the previous 3-4 design only reached 80,000-ft. This is an increase of almost 50% (34,000 ft.). The second is design uniformity. Since the diameters were the same, the diameters of all components could be the same for the booster and sustainer stages. While the length and other dimensions may be different, this one dimension allows for easier manufacturing and assembly because many of the custom manufacturing and assembly parts depend on the diameter to be used properly. With this design, the team is expected to surpass the previous teams and be the first Oregon State University rocket team to reach 100,000-ft.

A Proof-of-Concept for Hypersonic Air-breathing Propulsion Flight Test Using Sounding Rockets

Team 202 Project Technical Presentation to the 2019 SA Cup

Reed Antonich, Jarrett Elrod, Ethan Hunt, David Lomonte, William McCormick, Amanda Noori, Jesse Nyffenegger, and Siliviu Solomon

Florida Institute of Technology, Melbourne, Florida, 32901

INTRODUCTION

High Altitude/Mach Rocket (HAMR) is a capstone design project at the Florida Institute of Technology with the goal to fly a sounding rocket to near Mach 3 and 50,000 ft. The purpose of this flight is to collect flight data and then to compare this data to predictions from Computational Fluid Dynamics (CFD) in order to identify where the CFD models can be improved. All seniors in an engineering discipline at Florida Tech are required to complete a capstone project where they design, build, and test projects as a group. Since the spring of 2018 the team has been preparing for the full power test flight at the 2019 Spaceport America Cup.

PROJECT OVERVIEW

Hypersonic and supersonic flight are increasingly important fields of flight with billions of dollars being invested in research for both military and commercial applications. However, testing propulsion systems at these speeds is costly, hard to engineer, and dangerous. Most hypersonic and supersonic air breathing engines are extensively modeled in CFD and wind tunnels. However, there exists a gap between these methods, which use idealized flows, and flight tests, because of the real atmosphere and the vibratory environments. This is a growing problem as more and more military and even civilian applications are operating in these flight regimes.

We present a solution to this problem as our project. Our design consists of two subsystems: a payload consisting of a nosecone with modeled ramjet geometry, and a launch vehicle to propel the nosecone to Mach 2.8. By collecting temperature and pressure data from the nosecone we can measure the discrepancies between the idealized CFD model and the real-world performance. The payload nosecone will be manufactured with a ramjet geometry designed operate at a specific Mach number. The launch vehicle will be a rocket, with a fiberglass body to allow RF communication, and is powered by a Class 2 solid rocket motor. The rocket will be recovered by parachutes and located with GPS and radio transmitters. Because of the difficulty involved with the logistics and scheduling of high-altitude flights, it has been decided to enter the project into the 2019 Spaceport America Cup.

The inspiration for HAMR was a series of tests in the 90s and early 2000s called HYSHOT and HIFIRE. Large sounding rockets were used to boost payloads to hypersonic speeds in order to test new experimental hypersonic propulsion and flight systems. The importance of hypersonic flight has only grown since then, with China, Russia, and the United States all developing hypersonic weapons systems. With this growing importance the team set out to demonstrate that even universities with limited capabilities could provide meaningful impact on the field.

SYSTEM ARCHITECTURE

HAMR has two major assemblies: the launch vehicle and the payload. The launch vehicle holds the rocket motor, keeps the vehicle stable, and deploys parachutes for recovery. The Airframe subsystem contains the requirements for vehicle strength and integrity, stability, and the motor. The Recovery subsystem controls the recovery of the launch vehicle and payload, including parachute deployment and tracking. The payload assembly is responsible for collecting flight data such as the velocity of the rocket and the pressure and temperature of the air within the nose cone. The payload subsystems include physical payload design and payload electronics.

The payload geometry is designed to incorporate several features commonly found on ramjets, including an external compression zone that uses oblique shockwaves to celebrate the flow before entering the intake, wedges that promote turbulence where fuel would be injected, and a converging-diverging exit nozzle. The payload geometry is assembled from 6 CNC machined components to form the full nosecone. All machining was done by students in the Florida Tech machine shop. Immediately aft of the nosecone is the data acquisition system (DAQ) which uses 20 temperature and pressure sensors to collect data during the flight. The data is then stored on SD cards protected within a “vault” in the case of a sub-optimal landing.

The design of the launch vehicle system was informed greatly by the design report of “Don’t Debate This” which was the first successful minimum diameter Cesaroni N5800 flight, as ours also uses that motor. Similar to that rocket, our rocket uses aluminum fins, with the difference that ours are welded to the fin can, instead of brazed. Over 18 different combinations of fin airfoil, geometry, and alloy were analyzed with ANSYS Finite Element Analysis to find the optimal design. To ensure that the body tube was strong enough to withstand the acceleration of nearly 40 Gs a custom order of fiberglass tubes was ordered from Franklin Fiber and Lamitex. The recovery system uses a conventional “dual deployment” scheme, with a drogue deployed at apogee and a main deployed at 2000 ft. The altimeters are dissimilar models from PerfectFlite, to minimize the chances of a single failure mode causing both to fail. Tracking is provided by an Eggfinder TX GOS tracker, and a backup Communication Specialists RF transmitter. The rail buttons are designed to be a supersonic airfoil, to minimize drag.

TESTING

Before the full powered flight in June, several tests were performed. These included recovery ejection tests, tracking and locating tests, and structural strength tests. The culmination of the testing was a flight on an Aerotech K1250 motor to serve as a system shake down. The successful flight was done in coordination with the Spaceport Rocketry Association (NAR number 342).

RESULTS, CONCLUSIONS, AND FOLLOW-ON WORK

Moving forward the immediate goal of the project is a successful flight at Spaceport America. Once data has been collected a proper analysis can be performed to compare experimental and predicted values. Throughout the project this has been the goal, even with issues with manufacturing and assembly of the payload and launch vehicle. Once this analysis has been performed future teams can use the information and results to inform the design of supersonic or hypersonic air breathing propulsion systems. These systems could be tested at future Spaceport America Cup competitions