MACH-SR1: DEVELOPMENT AND CHARACTERIZATION OF HYBRID ROCKET TECHNOLOGIES

THROUGH UNDERGRADUATE R&D

L. Kanner, N. Ortmann, T. Mastalir, E. Raine, K. McWilliams, C. Madsen, R. Rieber, O. Krauss, B. Lewis, and D. Owen

Aerospace Engineering Sciences Department, University of Colorado

Boulder, Colorado

ABSTRACT

The MaCH-SR1 (Multi-disciplinary University of Colorado Hybrid Student Rocket) program is a multi-year student project to develop a hybrid launch vehicle to carry 10 lb_m payloads to 125 km altitude. To achieve this objective, the vehicle must produce 13000 lb_f (57826.9 N) of thrust. To date, the largest thrust produced was 5000 lb_f (2241.1 N). Through five years, teams have focused on nearly all aspects of the system design including mechanical, material, thermal, chemical and data acquisition design and analysis. The design utilizes a hydroxyl-terminated polybutadiene (HTPB) and industrial grade nitrous oxide fuel combination. Lab scale engines (100-300 lb_f (444.8 – 1334.5 N) thrust) were used to prove design concepts, gain fuel regression rate data, and verify data acquisition systems. The larger engines of 1000 and 5000 lb_f (448.2 and 2241.1 N) thrust were fired to obtain data for thrust, chamber and feed pressures, and oxidizer mass flow rate. Early designs incorporated heavy steel casings, carried excess fuel, and used commercial off-the-shelf (COTS) components. As the project progressed, developments in fuel recipe and geometry, lightweight materials, improved fuel-casting methods, oxidizer feed system, data acquisition, and detailed models and simulations were made for nearly every aspect of the system. Future efforts of the MaCH-SR1 project include fully integrating the oxidizer feed system with the chamber, demonstrating high thrust capability, developing a stabilization and control system, and aerodynamic optimization.

INTRODUCTION

MaCH-SR1 is an undergraduate student project at the University of Colorado, aimed at developing a hybrid launch vehicle to carry 10 lb_m (4.536 kg) payloads to 78 mi (125 km) altitude. The six-phased program takes the project from demonstrating the feasibility of hybrid technology to advanced system development and flight-testing high-power hybrid rocket engines as shown in Figure 1. From 2000-2006, successive teams of undergraduate students have taken the project through four of the six phases.

Phase	Description	Dates
Phase 0	Feasibility of Hybrid Rocket Technology	Sept 2000 - Jul 2001
Phase I	Demonstration of Hybrid Rocket Technology	Jul 2001 – May 2003
Phase II	Demonstration of High-Power Hybrid Rocket Technology	May 2003 – May 2004
Phase III	Advanced System Development	Jun 2004 – May 2006
Phase IV	*Hybrid Rocket Flight Testing*	
Phase V	*High-Power Hybrid Rocket Flight Testing*	

Figure 1: MaCH-SR1 Phased Development

Piloted in 2000, the initial “Phase 0” team developed the design concepts and formulated the program vision, laying the groundwork for designing, building, and testing a hybrid rocket engine. Phase 1 began in 2001. New fuel casting methods were developed, new tooling was designed and built, essential instrumentation of the engine was applied, and the associated data acquisition system was developed. Phase 1 culminated in two static tests that provided the regression rate data necessary in the development of a computational model. Phase 2 began in 2003 and demonstrated vertical integration of a high-power (5000 lb_f (2241.1 N) thrust) engine system.

Extensive research was done on fuel port geometry and new solid fuel manufacturing techniques were developed. The engine instrumentation was improved, enabling the previously developed regression model to be correlated with the two static tests. Phase 3, Advanced System Development, began in 2004 and continued through 2006. The primary focus of developing technologies to bridge the gap between test engines and flight-capable engines included examining alternative materials for lightweight engine components, exploring options for ignition systems, developing standardized fuel processing, investigating new sensors for data acquisition, and development of a high-fidelity combustion model. Future developments will be aimed at developing a flight-worthy system and completing the last two phases of the project: flight-testing and high-power flight-testing.

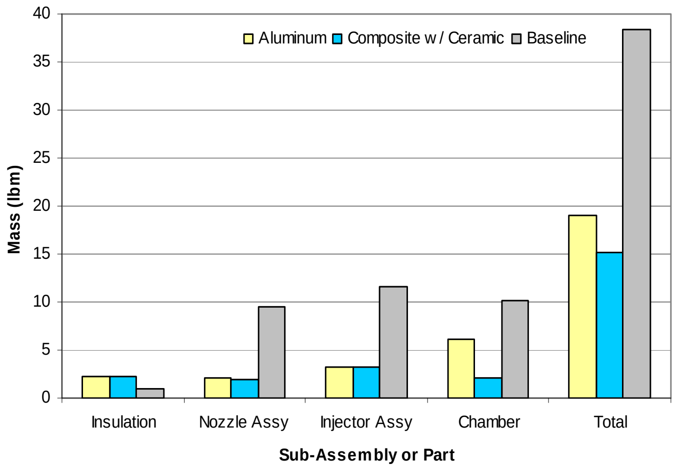

Figure 2: Weight Reduction Summary

MATERIAL DEVELOPMENT

The current engine configuration boasts a total weight reduction from both material and structural re-design of 62% over the original steel configuration, as shown in Figure 2. Advanced materials including aerospace metallic alloys, carbon-fiber composites, and various cutting-edge insulations have been tested and either been proven or dismissed for application on this hybrid rocket system. While successes are vital to forward progress, failures, such as a rupture of the oxidizer tank, lead to enhanced understanding and increased focus on safety during the design process.

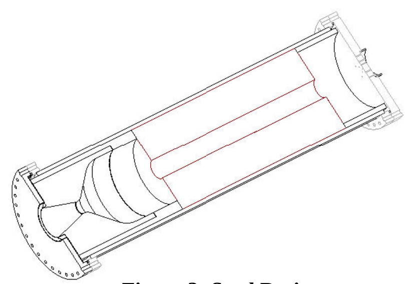

Figure 3: Steel Design

The initial MaCH-SR1 engine was a test-bed as shown in Figure 3. The majority of the system was constructed from off-the-shelf steel components, leading to unnecessarily high Factors of Safety (FOS). To decrease the dry mass, structural re-design and advanced materials were applied to nearly all of the hardware components.

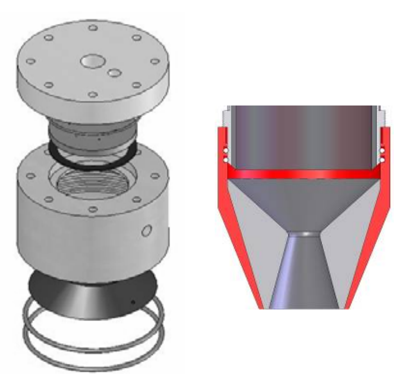

Figure 4: Aluminum Injector (left) and Nozzle (right) Fittings

Casing designs utilizing the strength of the fiberglass phenolic insulation were pursued in an attempt to decrease the system dry mass. In particular, the engine case phenolic was enclosed within a 3K x 3K carbon fiber composite wrap, providing an FOS closer to 2 than 10 as seen with the steel designs. FOS was slightly higher in the design of the injector and nozzle assembly fittings, both of which were designed from solid aluminum 6061-T6. The fitting design was altered significantly from the steel flange configuration for weight and aerodynamic concerns. The updated designs are shown in Figure 4.

Figure 5: Oxidizer Tank (cut-away)

Because aluminum has a much lower melting point than steel, insulation was added to the unprotected pre-combustor and post-combustor sections. Glass phenolic was the selected for the post-combustor because of its proven success in the early phases of MaCH-SR1, but the pre-combustor area required the insulation to contour to the surface of the injector fitting. For this purpose, a wet lay-up fiberglass called Cerafiber® was used. To prevent leak paths and keep the Cerafiber® from burning too quickly, Dow Corning ablative coating was also applied.

Composite materials were also used in the fabrication of an oxidizer tank. Ideally, the oxidizer tank would have consisted solely of a tank liner and a composite shell with valve fittings, but lack of sufficient facilities such as a filament winder made fabrication of such a tank impossible. Therefore, the tank was composed of four major parts as shown in Figure 5 the internal liner, the composite wrapped casing, the metal end-caps, and the carbon-fiber longitudinal bands. Ultimately, this design was limited by the bond strength between the composite casing and the end caps and failed structurally.

Figure 6: Eroded Castable Ceramic Nozzle

The nozzle material, requires more than sufficient material strength and density: it also must withstand significant thermal and pressure shock, as well as the corrosive high-temperature flame and exhaust. Throughout the program, the nozzle was consistently fabricated from a solid ATJ Iso-molded graphite Although the graphite design was extremely effective, factors lead to the evaluation of alternative designs: namely manufacturing, thermal conductivity, cost, and throat ablation. A copper design was considered, but cost, weight, and thermal insulation concerns and the inability to manufacture the part in-house limited the design. Instead, a castable silica oxide (SiO_2) ceramic nozzle was developed, serving both structural and thermal insulation purposes and mitigating fabrication concerns. Unfortunately, the ceramic nozzle's throat erosion proved to be severe in both time variation and magnitude. By the end of the hot-fire tests there was a considerable loss of thrust due to extreme erosion of the throat (leaving only a shape similar to that of the fuel port as shown in Figure 6). This failure provided valuable insight into the system and allowed for characterization of the ceramic material erosion via a MATLAB analysis using thrust, pressure, and flow rate data. The result of this analysis is summarized in the plot shown in Figure 7 and shows that the ceramic simply burned away more quickly than anticipated. While results varied for each nozzle design, it was determined that the traditional graphite nozzle provided the most consistent and cost-effective results in this application.

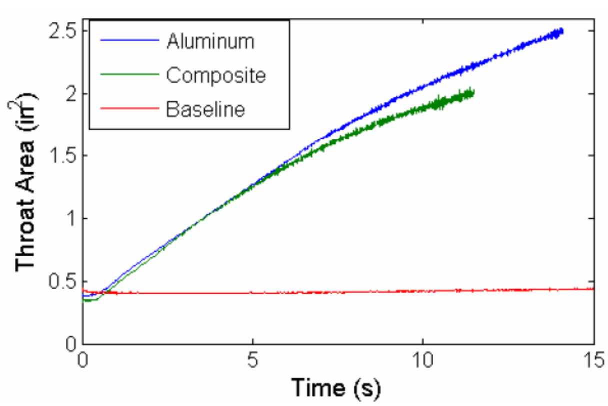

Figure 7: Nozzle Analysis Results

FUEL DEVELOPMENT

Bridging the gap between feasibility of hybrid rocket technology and high-powered engine testing requires a well-characterized and reliable propulsion system. Three main issues arise when designing the solid fuel for a hybrid rocket; all three are intimately related to each other and ultimately problematic for scaling to larger thrust systems. First, the combustion process in hybrid fuel grains is completely unlike that occurring in solid or liquid fueled rockets, in that a complex flow field develops within the combustion chamber that is completely dependent on geometry. The complex combustion and flow process leads to difficulties in scaling the predicted fuel consumption from a lab test to a full-scale device. Generally, larger diameter ports do not regress as fast as small ones, but the relationship is non-linear. Multiple-port designs also regress more slowly, as the oxidizer is dispersed among several perforations. The issues created by mixing the fuel and oxidizer also lead to another design issue: selecting a proper oxidizer-to-fuel (O/F) ratio. The mixture of fuel to oxidizer, combined with the chamber pressure, determines the chamber temperature, specific impulse (I_{SP}), and oxidizer mass for the motor.

The major propellant issues addressed during initial phases of the MaCH-SR1 project were performance characterization (the regression and consistency of the fuel), and efficient fuel grain design. Most of the historical regression data was obtained from lab and full-scale firings up to 1000 lb_f (448.2 N) that utilized simple round ports and large fuel margins. Lab scale demonstrations proved the performance of the fuel and paved the way for further optimization of the propellant grain. By focusing on the primary variable that determines thrust, the mass-flux level, the cross sectional area could be tailored to obtain neutral thrust profiles, particularly for the larger 5000 lb_f (2241.1 N) motor.

SOLID FUEL FORMULATION

Initial efforts focused development of the solid fuel recipe. Systematic trial and error established a repeatable recipe with desirable mechanical properties, such as high toughness and low radiative heat transfer. The final fuel recipe selection consisted of 83.56% Hydroxyl Terminated PolyButadiene (HTPB), 10.33% Isophorone Diisocyanate (IPDI), 0.5% castor oil, and 0.09% carbon black by weight. The formulation was cured at ~120^{o}F for 4 – 10 days, depending on the volume. The recipe proved to be reliable and successful during the demonstration and advancement project phases.

SOLID FUEL GRAIN DESIGN

The case-bonded fuel grain contains the solid portion of the fuel that reacts with the oxidizer to generate thrust in the system. Traditionally, the fuel grain and its interaction with the oxidizer has proven to be one of the highest-risk elements in the system. Similar to solid rocket motors, the grain contains an axial port whose shape and diameter largely determine thrust characteristics. However, unlike solid rocket motors, larger hybrid rocket motors typically use a grain with multiple ports.

The port requires careful attention to design; the fuel must be sized such that it does not regress completely before the end of the burn. As the required thrust increases, the length and port diameter of a single round port increases, leading to impractically long combustion chambers. However, as the diameter of the rocket increases, the port cross-section increases disproportionably to the fuel surface area and results in oxidizer passing down the port and exiting the nozzle without fully combusting. Multi-ports expose greater fuel surface area to the oxidizer, allowing for greater thrust in a shorter combustion chamber. Scaled multi-port designs were tested with unfavorable results. While the design decreased the length by 59%, the rapidly changing oxidizer to fuel ratio lead to variable thrust.

An efficient single port design required an increased internal surface area for a reduced port length and necessitated the use of fins in the fuel grain. Evaluations yielded a design using wide but short triangular fins for the 5000 lb_f (2241.1 N) thrust motor as shown in Figure 8. The design reduced the required grain length by 30% compared to a round port with similar mass-flux characteristics and chamber diameter. The grain design was successfully fired to >4500 lb_f (20017.0 N) thrust with combustion stability throughout the burn., and post-test observations show the port maintained its geometry and burned evenly as shown in Figure 9.

Figure 8: Port Geometry - 5000 lb_f engine pre-fire

FUEL PROCESSING DEVELOPMENT

Figure 9: 5000 lb_f fuel grain post-fire

During fuel recipe testing, the solid fuel was cast in 1 ft cardboard sonotube segments which were then bonded together and slid into the steel chamber. The segments were easy to cast, cure and handle, but were prone to leak paths. This was eliminated by casting the fuel directly into the chambers; necessitating larger cure ovens and a longer mandrel. Direct casting limited the visual inspection of fuel surfaces and attempts were made during processing to minimize voids and bubbles. Vacuum was added to the curing process, causing the fuel to bubble and extracting air while the fuel was heated. Fuel debonding and fuel grain inconstancy issues lead to the development of a vacuum casting system on the 300 lb_f (1334.5 N) engine. An apparatus was built to pour the fuel through a vacuum chamber into the engine casing, effectively degassing the fuel during the last step of liquid fuel processing.

Figure 10: 5000 lb_f Engine's Foam Mandrel Section

The mandrel is a piece of support equipment that is used to create the fuel ports during casting. The objective for mandrel design was easy removal after casting and minimal fuel interactions. Complicated port geometries also add complexity to mandrel manufacture and removal processes. The mixed star-round mandrel used for the 5000 lb_f (2241.1 N) engine grain design was constructed of stacked foam pieces (shown in Figure 10), machined to the desired profile. After cast and cure, the mandrel was partially removed by hand and remaining foam was dissolved with acetone. The foam facilitated both construction and removal of the mandrel, but left inconsistencies on the port surface. While this was not a concern for the larger engines, it was for the 300 lb_f (1334.5 N) lab-scale engine, and a Teflon mandrel with an oil surface treatment was developed.

OXIDIZER SELECTION

The use of liquid oxygen (LOX) for the oxidizer was established early in the program for the increased Isp, combustion temperature, and low mixture ratio required for optimal performance. Initial testing of LOX/HTPB motors ended prematurely though, due to the complexity of the cryogenic system and the reactive properties of LOX. Although nitrous oxide requires higher flow rates than LOX and a greater attention to two-phase flow, N_2O can be stored at room temperature, it is self-pressurizing, and feed lines and tanks do not need to be insulated against boil-off. These characteristics drove the program to the development and use of an N_2O oxidizer system.

MODEL DEVELOPMENT

Creating a hybrid rocket model was integral to the design, analysis, and verification of the MaCH-SR1 rocket. Initial models were simple yet effective spreadsheet calculations that determined the initial peak performance of the rocket utilizing ideal chemical rocket equations. Inputs into the model were the thrust, burn time, combustion chamber pressure, O/F ratio, and atmospheric pressure. GDL ProPEP, a freeware program from Martin Marietta Corporation (now Lockheed Martin), was used to calculate the combustion product's thermochemical parameters: mass, specific heat ratios, and combustion temperature. These were also input into the MaCH-SR1 spreadsheet model. From these inputs the nozzle throat area, nozzle expiation ratio, and mass flow rate were calculated. Mass flow rate, O/F ratio, and burn time was used for sizing oxidizer and fuel volumes.

The fuel regression rate for a single, cylindrical port was calculated using the basic regression formula:

$$\text{Fuel Regression Rate: } \dot{r} = a\,G_o^n \tag{1}$$

Where *a* and *n* are empirical determined coefficients from early lab-scale testing and G_o is the down-port oxidizer velocity. Assuming a constant regression rate, the required fuel thickness was found. The model also found the necessary orifice area in the injector plate for the calculated oxidizer mass flow rate via the chamber pressure, initial oxidizer pressure (which was assumed to remain constant) and oxidizer density. This simple spreadsheet model worked well for predicting initial start-up performance of the rocket. However, the model did not predict the time-varying effect of the fuel regression on performance, nor did it incorporate variations in the fuel regression rate.

A new, more sophisticated model was developed that accounted for the time-varying parameters. The new model was implemented in MATLAB, and based upon a differential equation with three state variables: oxidizer and fuel mass in the combustion chamber, and equivalent fuel radius. The forcing input to the equation was the injector oxidizer pressure. The MATLAB library of ODE solvers were used to solve this system of equations. In order to have the correct thermochemical parameters for any mixture ratio, look up tables were integrated into the code from GDL ProPEP data. For each time step, the code calculated the state variable, and from these the remaining characteristics of the rocket were calculated through the standard rocketry equations. The model provided improved insight into how the engine performance varied throughout the burn. This allowed for optimization of the rocket to provide a maximum average I_{sp} over the burn. In addition, by accounting for variation in fuel regression rate, it allowed for more precise sizing of the fuel grain.

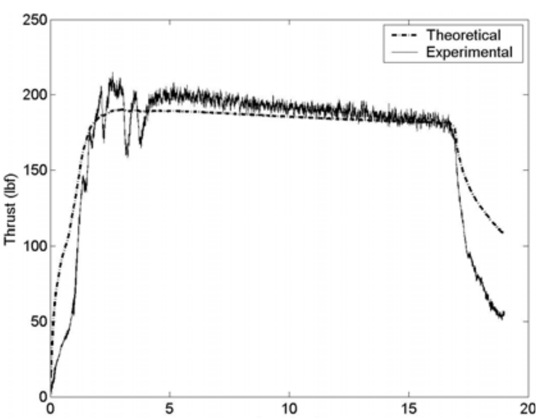

Figure 11: Theoretical and Experimental Thrust

A further improvement to the model was made by allowing experimental oxidizer injector pressure data to be input into the model. This allowed the model to be directly compared and validated with the experimental results. As shown in Figure 11, this new model correlated well with the measured experimental results.

The acoustics of the engine were also modeled as a simple resonator. The acoustic calculations were based on speed of sound inside the chamber and the length from the injector plate to the nozzle throat. The model's prediction of the first longitudinal acoustic mode at 1950 Hz correlated with the experimental data shown in Figure 12 with an error of only 1.5% from theoretical.

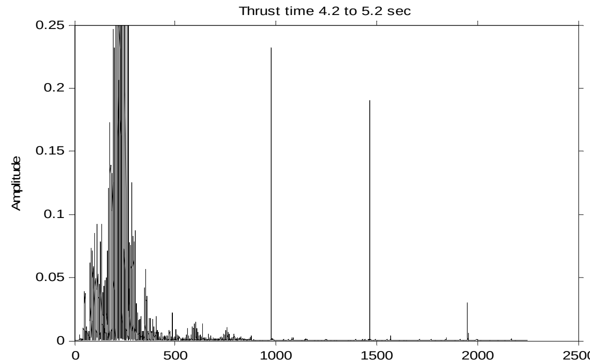

Figure 12: Frequency Distribution of Thrust

SUMMARY AND CONCLUSIONS

The MaCH-SR1 project has successfully demonstrated significant achievements and design advancements via a multi-year undergraduate research project. The ongoing development and improvements have developed the initial rocket design from a rough steel case motor to a high-tech, instrumented, and modeled test bed. By providing the opportunity for undergraduate students to conceive, design, integrate, test, and verify numerous aspects of a hybrid rocket design the MaCH-SR1 program has tested new technologies, produced repeatable results, and created mathematical modeling capabilities not available elsewhere. The MaCH-SR1 program has performed independent validation of industry standards, practices, and new technology research in a controlled, academic setting, thus proving the industry methods are applicable outside of their tightly controlled environment. Each success and each failure encountered by this program has added to the still-maturing knowledge base of hybrid rocket technologies; aiding in the development of this cutting-edge field.

FUTURE WORK

Future work on the MaCH-SR1 program is contingent upon continued student interest, and as such is a on a one year hiatus. Future teams have many technical challenges ahead of them including: instrumentation, an on-board ignition system, continued structural development, oxidizer containment, regulation, and delivery system design, aerodynamic enclosures, control system development, a communication system, and many other items.

In addition to the hardware, the software model can also be advanced. Two areas of the MATLAB model in particular have already been identified: fuel regression and injector mass flow calculations. The coefficients in the regression equation need to be more accurately determined to produce better results, and the effect of the heating of the injector plate on the discharge needs to be accounted for. Additional models will be required for areas other than engine performance as well.

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