Airbreathing Propulsion

Mixing of Supersonic Jets in a RBCC Strutjet Propulsion System

The Strutjet approach to Rocket Based Combined Cycle (RBCC) propulsion depends upon fuel-rich flows from the rocket nozzles and turbine exhaust products mixing with the ingested air for successful operation in the ramjet and scramjet modes. It is desirable to delay this mixing process in the air-augmented mode of operation present during take-off and low speed flight.

A scale model of the Strutjet device was built and tested to investigate the mixing of the streams as a function of distance from the Strut exit plane in simulated sea level take-off conditions. The Planar Laser Induced Fluorescence (PLIF) diagnostic method was employed to observe the mixing of the turbine exhaust gas with the gases from both the primary rockets and the ingested air. The ratio of the pressure in the turbine exhaust to that in the rocket nozzle wall at the point of their intersection was the independent variable in these experiments. Tests were accomplished at values of 1.0 (the original design point), 1.5 and 2.0 for this parameter and images of the degree of mixing taken at 8 locations downstream of the nozzle exit plane. The results illustrate the development of the mixing zone from the exit plane of the strut to a distance of about 18 equivalent rocket nozzle exit diameters downstream (18”). These images show the mixing to be confined until a short distance downstream of the nozzle, for a single nozzle geometry set. The lateral expansion is more pronounced at pressure ratios of 1.0 and 1.5 indicating that mixing with the ingested air flow would be likely to begin at an L/D of approximately 1 downstream of the nozzle exit plane. Of the pressure ratios tested in this research, a value of 2.0 delays the mixing until 2” downstream and is the best value at the operating conditions considered.

Sam Muller, Doug Parkinson, and Matt Turner completed research on this project under the direction of Dr. Clark Hawk.
Investigation of Ejector Performance in RBCC System

Ongoing research into highly reusable, Single Stage to Orbit (SSTO) vehicles seeks to attain dramatic reductions in launch costs through improvements in vehicle performance and operations. A key issue in realizing the SSTO goal is the development of air breathing propulsion concepts such as Rocket-Based Combined Cycle (RBCC). The RBCC concept is designed to obtain near optimum vehicle performance at all points in the launch trajectory by operating in four different modes: ducted rocket or air-augmented rocket mode, ramjet mode, scramjet mode, and a pure rocket mode.

Under a NASA Marshall Space Flight Center contract, the UAH Propulsion Research Center (PRC) designed and built a cold-flow Strutjet facility to simulate the air-augmented mode. A 1/6 scale model of a single strut has been built and is undergoing testing to investigate the mixing of the rocket and turbine exhausts with the ingested air. A complementary experimental program is also underway to examine the induced flow-field generated by rocket nozzles confined in a rectangular duct. Characterizing the induced flow behavior is critical to understanding and optimizing the performance of future Strutjet-based RBCC propulsion systems. The objectives for the UAH induced flow experiments are: (1) characterize the induced airflow as a function of the primary rocket flow; (2) determine the initial expansion rate of the rocket and turbine exhaust plumes; and (3) assess the expansion losses in the mixer/ejector section downstream of the strut.

Various flow visualization tests and pressure surveys have been performed. Pressure measurements indicate that the secondary flow induced in the rectangular duct by the primary rockets chokes when the rocket chamber pressure is between 200 and 300 psia. Flow visualization studies indicate the propagation of a starting shock in the duct as the rocket pressure is ramped up to 300 psia. The secondary flow choking is believed to be due to the minimum area in the duct/strut gap and not due to mass addition. Several total pressure surveys have also been made in the strut gap and the plume-mixing region. These surveys indicate a complex flow structure including barrel shocks and possibly vortices. Pressure traverses indicated the complexity of the flow in the supersonic rocket exhaust.

Future work will consist of repeating the pressure surveys with higher resolution instrumentation. This includes smaller diameter probes and higher precision pressure transducers. A CFD model of the induced flow is also being developed. Finally, tests of a single nozzle strut and other nozzle configurations are planned.

Mignon Thames, with the assistance of Doug Parkinson and Serena Gautney, completed research on this project under the direction of Dr. Brian Landrum.
The Apparent Failure of Scaling Methods in Ramjet Connected-Pipe Testing

The results of an evaluation of scaling methods for ramjet connected-pipe testing using previously reported data are presented. The evaluation of scaling methods includes the “pD” scaling method and another method (termed “mA” scaling here), which has not been previously documented, but is in current use by the ramjet testing community. In previous experimental studies, the results have not been systematically analyzed to determine the success or failure of the scaling techniques. This research presents for the first time such an evaluation, and it is concluded that neither of the two methods is successful in scaling ramjet combustors.

![Figure 1. Schematic of Connected-Pipe Testing](image)

John Blevins completed this research under the direction of Dr. Hugh Coleman.

Advanced Propulsion Concepts

Rayleigh Doppler Velocimetry Measurements of a Microwave Thruster

The performance of a microwave plasma thruster augmented with a magnetohydrodynamic accelerator is being evaluated. Both electrical conductivity and flow velocity are needed to determine the efficiency at the inlet to a magnetohydrodynamic accelerator. This work focuses on the measurement of velocity and temperature in the plumes of a microwave plasma thruster. Tests were conducted at pressures from atmospheric down to 1 torr. Rayleigh scattered light from an injection-seeded, frequency doubled Nd:YAG laser is analyzed using a planar mirror Fabry-Perot interferometer operating in the static imaging mode. Images were taken with an
ICCD camera and digitally stored for later evaluation using a least squares curve fit. Good agreement was seen between thermocouple data and the Rayleigh scattering technique at atmospheric conditions. Preliminary data taken at 1 torr show the Rayleigh signal dominated by scattered laser light coming from windows and metal surfaces. Methods of reducing this scatter along with the theory of Rayleigh scattering are under investigation.

Matthew Culley and Jonathan Jones completed this research under the direction of Dr. Clark Hawk.

**Design and Optimization of Advanced Magneto-Electric Propulsion Concepts**

Magneto-electric propulsion systems refer to thrusting systems powered by electric power. Magnetohydrodynamic, arcjet, and magnetoplasmadynamic thrusters are just a few of these advanced concepts. While very similar in operational principles with the conventional chemical rocket motors, the electric propulsion systems have the advantages of higher specific impulse although at low levels of thrust. These characteristics make them suitable primarily for in-orbit auxiliary propulsion systems and orbit transfer vehicles. Increased power levels of an electric propulsion system could also provide the primary propulsion capability of future interplanetary missions.

Several projects are ongoing and are oriented in two directions: improved design and optimization of plasma propulsion systems and plasma/plume diagnostics. The design and optimization of such a propulsion system start with attempts to improve the physical models used to describe the complex interactions between the fluid flow and the electromagnetic field in the plasma of such a system. Emphasis is put on continuous and pulsed microwave generated plasmas, as such systems have longer operational lifetime than the classic arcjet thrusters. Computer simulations and experimental data are merged in studies of a variety of propellants and plasma conditions.
The result of such studies is the design of new, more efficient thrusting systems.

Stelu Deaconu performed this research under the direction of Dr. Hugh Coleman and Dr. Clark Hawk

**Thermal Characterization of a Direct Gain Solar Thermal Engine**

A thermal/fluids analysis of a direct gain solar thermal upper stage engine is presented and the results are discussed. The engine was designed and constructed at the NASA Marshall Space Flight Center for ground testing in a facility that can provide about 10 kilowatts of concentrated solar energy to the engine. The engine transfers energy to a coolant (hydrogen) that is heated and accelerated through a nozzle to produce thrust. For nominal design values and a hydrogen flowrate of 2 lb/hr., the results of the analysis show that the hydrogen temperature in the chamber (nozzle entrance) reaches about 3800°F after 30 minutes of heating and about 3850°F at steady-state (slightly below the desired design temperature of about 4100°F). Sensitivity analyses showed these results to be relatively insensitive to the values used for the absorber surface infrared emissivity and the convection coefficient within the cooling ducts but very sensitive to the hydrogen flowrate. Decreasing the hydrogen flowrate to 1 lb/hr. increases the hydrogen steady-state chamber temperature to about 4700°F, but also of course causes a decrease in thrust.

![Figure 1. Schematic of Solar Thermal Engine](image)

Reginald Alexander completed this research under the direction of Dr. Hugh Coleman.
Investigation of Plasma Injection in Water

Efficient methods for manufacturing low cost, high quality metal coatings are needed by the U.S. military, NASA, and industry. Thermal spray technologies are often used to produce these coatings on metals, ceramics, polymers, and composites. Of the various thermal spray processes currently available, the highest quality coating process is Vacuum Plasma Spray (VPS). This process has been demonstrated to produce coatings with material properties similar to the wrought versions of the parent material. Russian researchers have proposed an alternative method to produce the plasma spray coatings without the need of vacuum chambers. This "Ultimate" or Underwater Plasma Spray (UPS) process, applies coatings in a surrounding water environment. The development of good plasma spray coating qualities depends on optimization between various parameters including: spraying distance, traverse speed, plasma jet energy (power), powder feed rate, and substrate material properties. Running the process underwater significantly modifies the gas dynamics and heat transfer.

We are currently investigating the injection of plasma into water. The ultimate objective is to develop a model to predict the penetration length and the temperature distribution as a function of the gas flow rate and the initial chamber temperature. The results of this study are applicable to the material coating processes, waste remediation, and possibly advanced marine propulsion. A computational code to predict the penetration length of argon plasma in water has been developed. Appropriate values for expansion angle, speed ratio, and water droplet diameter were determined from comparison with UAH experimental data. The code was then used to investigate the parameters affecting the penetration length and degree of ionization. Results show that there is a significant number of entrained water droplets in the plume. This suggests that the channel flow model that ignores water contamination does not adequately represent the plume development. The magnitude of entrained water is the primary factor that determines the rate of gas cooling and thus the plasma penetration length.

![CCD camera images of argon plasma injection in air and water.](argon/ into 5cm depth water)

Graduate Research Assistant Soichi Matsuura performed this research under the direction of Dr. D. Brian Landrum
Hybrid Rocket Propulsion

Hybrid Rocket Regression Rate Tailoring Program

Hybrid rocket motors combine the characteristics of both solid motors and liquid rocket engines. The propellant consists of two components, one in the solid phase (usually the fuel) and one in the liquid or gaseous phase (usually the oxidizer). Figure 1 shows a schematic of the hybrid combustion process for a gaseous oxidizer flowing over a solid fuel grain.

One of the problems associated with hybrid rockets is a low regression rate of the solid fuel grain. This problem dictates multiple ports in the fuel grain, resulting in a low density-impulse and a poor volume loading. The purpose of this program was to investigate new solid fuel formulations with higher regression rates. These regression rates were achieved by adding ammonium perchlorate (AP), a solid phase oxidizer, and iron oxide ($\text{Fe}_2\text{O}_3$), a catalyst, to the baseline fuel of hydroxyl-terminated polybutadiene (HTPB). Table 1 lists the seven different fuel formulations.

<table>
<thead>
<tr>
<th>Fuel Formulation Number</th>
<th>% HTPB</th>
<th>% AP</th>
<th>% Iron Oxide</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 (baseline)</td>
<td>100</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>2</td>
<td>75</td>
<td>25</td>
<td>0</td>
</tr>
<tr>
<td>3</td>
<td>70</td>
<td>30</td>
<td>0</td>
</tr>
<tr>
<td>4</td>
<td>70</td>
<td>25</td>
<td>5</td>
</tr>
<tr>
<td>5</td>
<td>72.5</td>
<td>27.5</td>
<td>0</td>
</tr>
<tr>
<td>6</td>
<td>72.5</td>
<td>25</td>
<td>2.5</td>
</tr>
<tr>
<td>7</td>
<td>70</td>
<td>27.5</td>
<td>2.5</td>
</tr>
</tbody>
</table>
The ballistic properties of these new fuels were investigated by a variety of test methods in the UAH High Pressure Laboratory. The purpose of the initial testing was to evaluate the flammability of the propellants. This was accomplished by igniting a small sample using a propane torch inside one of the test cells. For increased safety in an actual motor, the propellant sample should not sustain combustion without an oxidizer flux. The fuel formulations in this case all had an oxidizer in the binder, namely ammonium perchlorate. As a result, each sample was capable of sustaining combustion until all of the AP was consumed. This is not the ideal result, but it is not an absolute contraindication either. The combustion was neither violent nor very rapid.

Next, the formulations were tested in the closed bomb. The closed bomb has a nitrogen feed/vent system to control internal pressure. In addition, a pressure transducer and an ultrasonic transducer attached to the bomb allow the data acquisition system to record pressure data versus regression rate. With this data, a correlation between pressure and regression rate (if any) can be calculated. In addition, the bomb is inside an environmental chamber, which allows the initial temperature to be varied, so that temperature sensitivity of the propellant can be determined.

After closed bomb testing, extinguishment tests were conducted to determine if rapid depressurization would extinguish the propellant. This test is performed using an open bomb that is sealed at the top by Mylar sheets. These sheets have been rated for 100 psi through in house testing, which allows the creation of seals for various pressure levels. The seal is ruptured remotely using a nichrome wire. Also, the rate of pressure decay was varied to determine if the depressurization rate had any effect. This was accomplished through the use of orifices of various sizes.

After characterizing the ballistics of the fuel formulation, several sample fuel grains were cast for testing in the UAH Labscale Hybrid Motor. The fuel grains measured 12 in. in length and had a 1.5-in. diameter port. They were case bonded to a phenolic sleeve to facilitate cartridge loading of the motor. Figure 2 shows a schematic of the laboratory scale hybrid motor.

The feed system was designed to supply up to 2 pounds mass of oxygen each second.

The regression rate was to be measured through the use of an ultrasonic transducer attached to the phenolic sleeve of the fuel grain. The data acquisition system was to be used to collect data on chamber pressures, temperatures, regression rate and oxidizer flow rates.
This project is on hold until NASA-supplied hardware can be delivered. Once the feed system has been assembled and tested, each fuel formulation will be burned in the motor. After the data from these experiments has been analyzed, the next iteration of fuel formulations will be determined and follow on testing will be proposed.

Ted Rochford performed this research under Principal Investigator Dr. Robert Frederick, Jr. and Dr. Marlow Moser.

**Laser Propulsion**

*Introduction of a New Concept of Ablative Laser Propulsion (ALP)*

Recently demonstrated conceptual viability of laser propulsion opens a new field for experimental research. As expected, this research encompasses a broad range of problems concerned with lightcraft aerodynamics, geometry of focusing elements, laser irradiation characteristics defining beam divergence, and propagation and plasma excitation (wavelength, repetition rates, pulse duration and structure, etc). In spite of all this, the keystone question that will set the directions for development of the field is “what sort of propellants must be used for laser propulsion?” “What kind of standard can be applied to the selection of the right propellant?” In order to address these questions, the strict distinction must be made between laser propulsion based on solid and fluid propellants, which in terms of LP are characterized by several fundamental differences, such as directionality of the thrust, laser-induced breakdown threshold irradiances, ablation pressures, *etc.* The comparative analysis of these two major groups of propellants demands more attention toward solid propellant based laser propulsion.

Eric Broyles conducted this work under the direction of Dr. Andrew Pakhomov.

**Liquid Rocket Propulsion**

*NO2/ Propane Rocket Tests*

A proof of principle test of a nitrous oxide/propane rocket concept was tested on the UAH campus. The project was funded by the Alabama Space Grant Consortium and was a joint effort between the PRC and MicroCraft Corporation. The concept is attractive for small orbital maneuvering engines because the propellants are commercially available and relatively easy to handle. Also because of the high vapor pressure of liquid nitrous oxide the system can be designed to be self pressurizing, simplifying the overall system. The tests were accomplished at a sub-scale rocket test facility that has been constructed on the edge of the UAH campus.
This facility can handle rocket firings up to 150 lbf. The test stand is surrounded by an earthen berm and operated remotely for the safety of test personnel. The initial tests were at chamber pressures of 150 to 300 psi and had a cstar efficiency of nearly 100% with a double counter swirling coaxial injector. A follow-on effort is expected to further develop this concept.

Dr. Marlow Moser, assisted by Nathan Smith, conducted this research.

**Pulse Tube Refrigerator**

First described in 1963, a pulse tube cooler is a closed system that uses an oscillating pressure at one end (typically produced by a compressor) to generate a gas flow in the rest of the system. Under the right conditions, this gas flow (usually helium gas) will carry heat away from a low temperature point (the cold heat exchanger). A single-stage orifice pulse tube cooler has reached temperatures as low as 30K. A 3-stage cooler developed in Japan reached 3.6 Kelvin in 1994. A cheap, robust cryocooler could be used in various space exploration, civilian, and military applications.

The research plan for this project is to implement a CFD analysis of the pulse tube refrigerator. The analysis will simulate the momentum and enthalpy streaming in the whole refrigerator and determine the critical parameters and components for cooling performance. Models will be developed for the 1-D and 2-D thermodynamic and hydrodynamic characteristics of each component. The model simulations will be compared to experimental data obtained in a prototype cooler.
Figure 4. Orifice Pulse Tube

Mr. Robert Middleton, Professor Brian Landrum, and Graduate Student Zhongmin Li carried out this research