The latest in hypersonic technology is the supersonic combustion ramjet, or scramjet. This air-breathing jet engine is capable of achieving speeds in excess of Mach 10 without the use of moving components or the aid of an onboard oxidizer. Researchers at the University of Virginia are currently testing a phase of scramjet operation known as mode-transition in their hypersonic wind tunnel. Wind tunnel testing, however, can only provide limited understanding, and in order to develop a truly reliable scramjet, the Hy-V project will conduct the first mode-transition flight test of a dual-mode hypersonic scramjet. A sounding rocket will be used to propel the scramjet to Mach 5 where the mixing of fuel and air for combustion is optimum and an altitude of 28 kilometers where the flow conditions simulated by the wind tunnel are matched. Heating of the payload during this flight is of great concern, since it is a goal of the project to recover the payload unharmed. The heat transferred to the exterior of the rocket as well as the combustion chamber’s interior have been calculated, opening the way for future finite element temperature analyses. For thermal management, ablative surfaces and heat sink options have also been researched.

I. INTRODUCTION
Supersonic combustion ramjets—or scramjets—are poised to become the next step in the aerospace industry’s pursuit of faster, more efficient flight. NASA’s recent Hyper-X program has demonstrated the feasibility of this technology, propelling an airframe integrated scramjet to nearly ten times the speed of sound. The success of this flight, however, does not guarantee the success of the technology. Accurate models of scramjet combustion are still inadequate, especially during the critical portion of scramjet operation known as mode-transition.

Without such models, current wind tunnel tests of hypersonic scramjets cannot move forward. Therefore, it is the eventual goal of this research to conduct the first atmospheric flight test of a dual-mode hypersonic scramjet during mode-transition. The findings of this experiment will be compared to the wind tunnel tests, providing researchers with the data necessary to develop accurate scramjet mode-transition modeling tools. To obtain the Mach 5 flight speed at which mode-transition occurs, the engine will be mounted at the leading edge of a Terrier Improved-Orion sounding rocket launched from the NASA Wallops Flight Facility. Near apogee, the clamshell nosecone of this rocket will separate, exposing the scramjet to the flow and beginning the experiment. In order to gain as much information from this flight test as possible, it is a goal of the researchers to recover, intact, the scramjet and the payload test section. In order to do so, heat transfer to the exterior skin of the sounding rocket and the interior of the scramjet’s combustion chamber will have to be closely regulated throughout the duration of the flight test.

The experiment has been dubbed Hy-V [pronounced High-Five]; ‘Hy’ referring to the hypersonic flight regime that defines scramjet flight, and the ‘V’ as the Roman numeral 5 referring to the flight Mach speed of the experiment, as well as the five members of the Virginia Space Grant Consortium that are collaborating on this project. As a letter, ‘V’ also represents the Commonwealth of Virginia.

The following presents a brief overview of the proposed flight test, including design of the scramjet and payload test section and the proposed flight trajectory. Next, the steps taken in calculating the heat transfer rates are shown, concluding with future plans for research and subsequent design decisions.

II. PROJECT OVERVIEW
The ultimate goal of the Hy-V project is to allow for the comparison and calibration with the University of Virginia’s Mach 2, continuous flow wind tunnel tests that are currently studying scramjet mode-transition. This will provide an empirical comparison between the atmospheric and ground tests and aid researchers who are seeking to improve the reliability of computational models of hypersonic vehicle.

GEOMETRY
To accomplish this reproduction, a to-scale replica of the scramjet combustor with a wind-tunnel tested isolator (Fig. 1) will be incorporated into the payload of the Terrier Improved-Orion sounding rocket. To integrate the two, two inlet geometries were proposed.
The first design (Fig. 2-top) has an annular leading edge with the scramjet flow path passing through its center. The blunted nose of this design causes a bow shock to form across the inlet to the combustor, resulting in subsonic flow behind the normal shock. Optimal mixing and ignition of the flow and the fuel inside a scramjet combustor, however, occurs at Mach 2. Therefore, this design must utilize a converging-diverging nozzle to reestablish supersonic flow. Since a nozzle is used to regulate the speed of the air within the combustor, this design is relatively immune to the effects that flight pitch and yaw have on the flow conditions. However, having the scramjet positioned through the center of the payload section has the disadvantage of having to vent the flow from the center of the payload section to the outside. To do so would involve elaborate duct work running from the outlet of the combustion chamber to the exterior of the sounding rocket. Such a system would require a significant amount of space, however, and with the flow path passing through the center, dividing the payload section, space is already severely limited.

The second proposed design (Fig. 3-bottom), incorporates a wedge with two scramjets running along opposing sides of the payload section. This design lacks the flow-regulating nozzle of the previous design. Instead, the angle of this design causes the formation of oblique shock waves on the tip of the wedge and the entrance to the scramjet duct. As flow crosses the shock waves pressure increases, and, as a result, the velocity of the flow decreases. The wedge is designed to have a twenty degree half angle. This angle is critical, in that it creates the necessary shock geometry to slow the Mach 5 flight speed to the desired Mach 2 flow speed within the combustion chamber. With conditions within the chamber being thus dependant on the angle of the oblique shock, this geometry can result in the flow being adversely affected by variations in the pitch of the rocket during the experiment. Therefore, attempting to maintain level or constant angle of attack flight during the experimental phase of the trajectory becomes very important. In the final selection process, however, several advantages to this design overcame this disadvantage. With the two chambers running along the outside of the payload section, the flow can be directly vented to the atmosphere, allowing one to reserve the middle of the payload section for the necessary sensing and data acquisition equipment. The added intricacy of the venting duct work is also unnecessary in this design. Most importantly, the wedge inlet allows for two scramjets to be simultaneously run during the flight experiment. With two scramjets, it would be possible to test two different fuels or two different ignition methods. Virginia Polytechnic Institute has proposed testing their plasma torch ignition system in one of the scramjets. This system has recently undergone hypersonic wind tunnel testing and is a promising candidate for reliably igniting the supersonic flow and fuel mixture. It has also been proposed to run both engines under the same conditions with the same fuel and ignition system. This would build some redundancy into the flight test and therefore reduce the uncertainty of the experimental results. Regardless of which of these proposed ideas is implemented, it is clear that having two scramjets running simultaneous tests is greatly to the advantage of the researchers. Therefore, all subsequent flight test decisions are based on the use of the wedge geometry for the payload’s inlet.

TRAJECTORY

The trajectory of the Terrier Improved-Orion (Fig. 3) is essential in recreating the wind tunnel’s flow conditions during the atmospheric flight test. At altitude, the sounding rocket will reach 28 kilometers. Here, atmospheric conditions are such that after passing through the oblique shocks created by the wedge inlet, pressures will parallel those within the wind tunnel. Since it is desirable that these conditions be matched throughout the 15 to 30 seconds of the scramjets’ combustion, the rocket’s trajectory was manipulated so that the altitude at apogee varies as little as possible. To do so, the rocket will be launched from an angle of 45
degrees from the horizontal. This angle also creates a trajectory in which the Mach number remains relatively constant during a portion of the flight, as seen in the top right of Fig. 4. This constant velocity is critical for a successful flight test. In order to achieve combustion, the rocket will need to travel at or above Mach 5 for the period of data acquisition. After passing through the shock waves created along the wedge, the flow velocity within the combustion chamber will be Mach 2. At this speed, mixing of the fuel and air for combustion is an optimum.

III. THERMAL MANAGEMENT

With the sounding rocket being launched at 45 degrees and spending the entire duration of its flight in the lower atmosphere where the density of the air is still significant, the heat transferred to the outer skin of the payload and to the interior of the combustion chamber will be greater than that of the typical sounding rocket flight. Since it is a goal of the project to recover the payload section intact for materials analysis, the heat transferred to the payload will need to be considered when selecting an ablative surface coating for the test section and when positioning the interior’s electronic components.

METHOD

The heat transfer in a hypersonic, turbulent flow has only recently become an area of interest for research scientists. Because of the inherent randomness of turbulent flow, mathematical solutions to such problems do not exist, so these recent empirical tests and their subsequent theories must serve as a best estimate for this atmospheric flight test. For the Hy-V experiment, one of the most popular empirical solutions for compressible skin-friction with turbulent boundary layers, the van Driest II, was used to calculate the heat transfer rates. This model assumes a calorically perfect gas and uses scaling factors to account for the compressibility of a high speed flow. These scaling factors are given as

\[ F_c = \frac{T_{aw}/T_e - 1}{\left(\sin^{-1} \kappa + \sin^{-1} \nu \right)^2}, \]  

\[ F_{Re_x} = \frac{\mu_e}{\mu_w F_c}, \]

where

\[ \kappa = \frac{T_{aw}/T_e + T_w/T_e - 2}{\sqrt{(T_{aw}/T_e + T_w/T_e)^2 - 4T_w/T_e}} \]

\[ \nu = \frac{T_{aw}/T_e - T_w/T_e}{\sqrt{(T_{aw}/T_e + T_w/T_e)^2 - 4T_w/T_e}} \]

where the subscripts \( w \) and \( e \) denote the wall and freestream conditions respectively. The incompressible skin friction coefficient and Reynolds number are then given by

\[ C_{f,i} = F_c C_f, \]

\[ Re_{x,i} = F_{Re_x} Re_x. \]

The Stanton number was then calculated for laminar or turbulent flow using

\[ C_h = \frac{0.4}{\sqrt{Re_{x,i}}} \text{ for } Re_{x,i} \leq 5 \times 10^5, \]
\[ C_h = \frac{C_{f,i}}{2} \text{ for } \text{Re}_{x,i} > 5 \times 10^5. \] (8)

From these the heat transfer rate was calculated by

\[ \dot{q} = C_h \rho_e \mu_e \left( h_{aw} - h_w \right), \] (9)

where the adiabatic wall enthalpy, the enthalpy of the freestream, and the enthalpy at the wall are

\[ h_{aw} = h_e + \frac{1}{2} r u_e^2, \] (10)
\[ h_e = c_p T_e, \] (11)
\[ h_w = c_p T_w, \] (12)

respectively. Here, \( r \) is the recovery factor for turbulent flow equal to 0.89 according to White, \( c_p \) is the constant-pressure of specific heat, and the temperature at the wall is assumed to be 300K.

**CALCULATION RESULTS**

For the proposed trajectory the heat transfer rates along the outer wall of the wedge inlet were calculated to exceed 100 MW as seen in Fig 6. This graph shows the heat transfer rates versus time along the outer surface of the wedge for the portion of the flight test after which the nosecone has separated, exposing the scramjet’s inlet. The various lines correlate to locations along the length of the wedge with the heat transfer rates being greatest at the tip of the wedge. At this location, the extreme drop in the heating rate seen between 65 and 130 seconds is due to the dependence of the Stanton number on the nature of the flow. This time corresponds to the only time for which the flow is laminar (\( \text{Re}_x < 5 \times 10^5 \)), and thus the heating rates are comparably negligible. The fluctuations of the heat transfer rates with respect to time are proportional to the fluctuations in dynamic pressure (Fig 4) that the rocket will experience throughout the flight. At the end of the wedge (Fig 3), the payload transitions into the two inlets, where the presence of an oblique shock wave will change the properties of the flow within the combustion chamber. Using these properties, the heat transfer rates within the combustion chamber are shown in Fig 7.

**PROPOSED SOLUTIONS**

Since the primary goal of this flight test is to obtain pressure and temperature measurements of the flow to compare to and calibrate the wind tunnel test data, the greatest heating concern is the heat transfer to the combustion chambers’ walls. Along the scramjets walls, the pressure taps and thermocouples will be placed to take measurements of the flow properties while the fuel is being injected and combustion is taking place. In the laboratory tests, water-cooling is
used to reduce the heating of the combustion chamber. With this method, water travels through a series of pipes in the combustion chamber’s casing and absorbs heat away from the chamber by transferring energy in the form of heat to the water and then removing it from the system. Such a system, however, cannot be practically implemented within the rocket’s payload. Both space and weight are limited by the constraints of the sounding rocket, and a water cooling system would require too much of each. A more space conscious alternative would be the use of a copper heat sink surrounding the interior walls of the combustion chamber. Copper, which has a high thermal conductivity of 400 Wm⁻¹K⁻¹, would have the advantage of being able to transfer large quantities of energy away from the wall of the combustion chamber and the sensing equipment and into the interior of the payload.

Excess heat in the interior of the payload is not desirable either, however. The data acquisition equipment and other temperature sensitive components will either have to be rated to withstand the high temperatures or be insulated against them. Decisions concerning this will be able to be made within the next year as more detailed thermal analysis takes place and a finite element temperature analysis is performed to determine the temperature distribution within the payload’s interior.

With the sounding rocket flying in the lower atmosphere, the heating of the exterior aluminum skin will also have to be regulated. The exterior of the rocket’s payload will be covered with an ablative material that will burn away slowly as it is heated while the remaining material will serve as an insulator on the aluminum skin of the rocket. In similar hypersonic experiments, including the University of Queensland’s HyShot, phenolic cork was used for this purpose. This material, as well as other resins, is currently being researched and will be tested before a final selection is made.

**IV. FUTURE PLANS**

The Hy-V flight test is set to occur in the summer of 2009 from the NASA Wallops Flight Facility in Wallops Island, Virginia. In order to have a successful launch and gather the desired data, the next two years will be rigorously devoted to the design and testing of the payload section. The research that is currently taking place will be expounded upon and, by the fall of 2007, the preliminary design will be finalized and first stage construction of the payload will begin.

Further modifications are to be made to improve and expand upon the heat transfer calculations. To create a more accurate model, the value for the temperature at the wall will be made variable, reflecting the temperature change as heat is transferred to the wall’s surface. A program is being written that has as its sole input the trajectory provided by NASA Wallops and from there outputs the heat transfer rates. With this program, if the trajectory is modified, the heat transferred to the payload section can be quickly recalculated and the research furthered without much delay.

Once the trajectory is finalized, more extensive heat transfer calculations will be able to take place. For the verification of the heat transfer rates calculations, it has been proposed to perform wind tunnel tests measuring the temperature change of the water in the cooling system, as well as the temperature change of the scramjet casing during combustion. From these values, estimates can be obtained for the heat transfer rates within the combustion chamber due to both the friction of the flow and the scramjet combustion. This will give an order of magnitude comparison for the calculations and allow finite element analyses to proceed with an increased degree of confidence.

These finite element analyses are the next step in the thermal management study of the sounding rocket payload and will provide the temperature distribution within the payload section and the scramjet combustor. Using this data, multiple ablative surface coatings will be modeled and tested in order to determine what methods for heat reduction are most feasible for this flight test. By 2008, when construction of the payload is underway, a finalized surface coating will be chosen based on these calculations. An interior temperature distribution will also allow for the layout of the scramjet’s electronic components to be determined. Temperature sensitive equipment will be placed towards the center and the rear of the payload test section, away from the intense heat of the scramjet combustors. It may also be found that additional insulation is necessary for some of the components.

This research, as well as the work of many other researchers, will help to guarantee a successful flight test in 2009. The results of this experiment will help researchers better understand scramjet mode transition and advance the field of hypersonic flight.
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