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A STUDY OF METHODS TO INVESTIGATE NOZZLE BOUNDARY LAYER TRANSITION

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I. Research Objectives and Potential Impact on Propulsion

Supersonic flow is a topic of strong interest which arises in applications ranging from the nozzle of a booster rocket to the National Aerospace Plane. Many characteristics of supersonic flow have been studied in detail and are well understood. The transition of a supersonic laminar boundary layer to turbulence, however, has been very difficult to study. Experimental measurements of transition are particularly difficult in rocket nozzle applications because the flow temperatures exceed 2300° C. In this case, only temperature measurements along the outside of the nozzle wall can be made. This yields the wall heat flux within the nozzle but does not indicate flow velocities or temperatures. It also does not provide a tool to predict the performance of nozzles.

To further investigate the nozzle flow, numerical computations are employed. The computations produce complete flow velocity and temperature fields within the nozzle. As a check these results can be compared with experimental data at the wall. Once an accurate numerical scheme has been validated, it can be used as a design tool to predict the performance of other nozzle designs without the cost of experimental testing. Typically the numerical analysis assumes either a laminar boundary layer or a fully turbulent boundary layer which is steady and two-dimensional. Boundary layer transition is not considered. Computing both the completely laminar boundary layer and the completely turbulent boundary layer conditions gives the minimum and maximum wall heat flux possible for a specified geometry. When the experimental heat flux measurements lie between these two values, the nature of the boundary layer is unknown. The boundary layer may have transitioned from laminar to turbulent, three-dimensional structures may be present in the boundary layer, or the inlet flow conditions may not be correctly specified in the computation.

In the NASA Lewis 1030:1 Area Ratio Nozzle, a series of experiments were conducted over a range of chamber pressures (Smith, 1988). The nozzle being tested was a low thrust nozzle design for space applications such as for orbital transfer vehicles. The throat diameter of the nozzle was 1 inch and the chamber pressure was varied from 360 to 1004 psi to give a thrust range of 500 to 1200 lbs. The heat flux measurements were compared with numerical predictions. At low chamber pressures, the experimental heat flux data corresponded closely to the laminar boundary layer computations. When higher chamber pressures were tested, the heat flux was found to be between the laminar and fully turbulent boundary layer predictions. The characteristics of the boundary layer were not correctly described by the laminar or turbulent calculation and therefore the heat flux was not predicted. Under these conditions, the nature of the boundary layer can not be inferred from the data.

A boundary layer stability analysis can reveal the onset of transition or the growth of a three-dimensional structure in a laminar boundary layer. In the present study, a stability analysis will be used to investigate the supersonic boundary layer in a rocket nozzle. The study will focus on the NASA Lewis high-area-ratio nozzle conditions which experimentally produced wall heat flux values between those predicted by laminar and turbulent computations. Through a stability analysis, the location where transition begins and the structure of the most unstable disturbance can be predicted. Tollmien-Schlichting (planar waves) and Taylor-Görtler (longitudinal vortices) instabilities will be considered as possible transition mechanisms. This study will define the boundary layer region which is laminar and which can be predicted accurately by a two-dimensional, steady, laminar computation. Establishing the structure of the most amplified instability will then lead to an accurate model of the transition region. On the concave wall of the supersonic nozzle, it is expected that transition will be triggered by the Taylor-Görtler instability. The stability analysis will determine the wavelength of the disturbance which grows most rapidly. A three dimensional boundary layer computation will then predict the enhancement of the heat flux when a vortex structure of the dominant wavelength is present. These predictive models will be compared with the results from high-area-ratio nozzle experiments reported by Smith (1988). Once validated, the methods developed can be used in predicting the performance of new nozzle designs.

II. Current Status and Results

The research program will investigate the boundary layer structure found in high-area-ratio rocket nozzles. The experimental results of Smith, (1988) will be used to validate the numerical findings. All chamber pressures tested will be repeated in this numerical investigation. As higher chamber pressure results become available, numerical computations will also be conducted at those conditions. This research program will yield a predictive tool useful in analyzing other rocket nozzle designs. The research can be divided into three tasks, a boundary layer computation, a stability analysis and a transition model development.

The first task is to compute the laminar boundary layer flow throughout the entire length of the nozzle. This provides the mean flow which will be used in the stability analysis. At low chamber pressure conditions, the laminar computation should produce heat flux values similar to the experimental results. At high chamber pressures, the laminar heat flux predicted from the computations will be below the experimental values. This indicates that the experimental boundary layer begins to transition to turbulent flow. At the high chamber pressure conditions, a stability analysis will be used to predict the location where the laminar boundary layer begins to undergo transition. The first stage of the research has been started. A well-tested computer program is used to solve the compressible Navier-Stokes equations in the entire nozzle. The program uses a flux-splitting scheme and has been shown to give accurate results for a wide variety of nozzle geometries. Accurate results are also expected for the high-area-ratio nozzle of interest.

Accurate flow inputs and a smooth computational grid are required for accurate numerical results. A chemical equilibrium and composition program* was used to determine the chemical composition and properties of the inflow gas using the experimental conditions cited by Smith (1988). A frozen flow (constant chemical composition) assumption was then made when computing the flow within the supersonic nozzle. Using tabulated properties for hydrogen-oxygen systems (Svehla, 1964), constants for the Sutherlands law viscosity expression were found so that the viscosity was accurately predicted throughout the entire temperature range of interest. A computational grid was produced which includes mesh clustering near the nozzle wall in order to resolve the boundary layer (see figure 1). Mesh clustering was also added near the throat where strong velocity gradients are expected. To minimize the skewness of the velocity vectors with respect to the grid cells, each spanwise grid line is a circular arc which is normal to the nozzle wall.

Presently we have obtained nozzle results when the chamber pressure is 360 psi. Within the diverging nozzle, the wall temperature was set to the average wall temperature measured experimentally. The velocity contours within the supersonic portion of the nozzle are shown in figure 2 and the Mach contours near the throat are shown in figure 3. In figure 4, the heat flux from the present computations is compared with the experimental results. The computational results are approximately 10% above the experimental results. For the low chamber pressure conditions, the computational results should accurately predict the experimental measurements.

To get a better comparison between the computational and experimental results, several modifications are currently being tested. The accuracy of the parameters in the viscosity model is being reexamined and improvements in the model across the operating temperature range will be implemented. The program is being modified so that a temperature distribution can be specified along the wall. Currently, the wall temperature is set to a constant in the entire supersonic region. The grid resolution near the nozzle wall and in the inviscid flow region is also being tested to assure that the numerical solution is grid independent. The modifications will lead to a more accurate representation of the problem and should also yield more accurate heat flux results. When the experimental heat flux results are correctly predicted, it will be inferred that the boundary layer flow within the nozzle has been correctly described by the computation.

After producing accurate results at the low chamber pressure, high chamber conditions will be computed. The laminar boundary layer results at the high chamber pressures will be used as the mean velocity profiles for the stability analysis. The location where the boundary layer begins to transition and the structure of the disturbance which triggers transition will be determined by the stability analysis.

III. Proposed Work for Coming Year

In the second phase of this research, the stability of the laminar boundary layer will be tested numerically. The stability analysis will indicate the structure and wavelength of the disturbance which will be amplified most rapidly and thus which will

* Referred to as CEC76 and developed by S. Gordon and B. J. McBride at the NASA Lewis Research Center.

cause the laminar boundary layer to undergo transition to turbulence. At every streamwise location, the amplification rate (α) for all wavenumbers is determined for both Tollmien-Schlichting and Taylor-Görtler instabilities. The amplification ratio (a) can be used to determine which disturbance has grown most rapidly throughout the boundary layer development. It is defined as the ratio of the amplitude of a disturbance to its amplitude at neutral stability.

$$a = \exp\left(-\int_{x_n}^{x} \alpha dx^{*}\right)$$

The disturbance which has the largest amplification ratio will initiate transition of the laminar boundary layer. Typically transition occurs when the amplitude ratio reaches e^9 or e^{10}. The criterion has been found to give an accurate prediction of the transition point when either Tollmien-Schlichting or Taylor-Görtler instabilities cause transition of compressible or incompressible boundary layers. This method for predicting the transition location is known as the e^N method.

Chen, *et al.* (1985) investigated the transition of the boundary layer in the diverging nozzle of a supersonic wind tunnel. Tollmien-Schlichting and Taylor-Görtler instabilities were considered along the curved wall. From a boundary layer stability analysis, they found that Taylor-Görtler instabilities grew more rapidly in the supersonic nozzle and caused the transition of the laminar boundary layer on the wind tunnel walls. Transition in the experimental facility occurred at the location where the amplification ratio from stability analysis had a value of e^9 to e^{11}. They suggest an amplification ratio of $e^{9.2}$ as a design criterion for transition.

To predict the location where transition begins, the e^N method will be used in the current investigation. The e^N method was successfully used to predict transition in the supersonic wind tunnel (Chen, *et al.*) and the method is expected to give accurate results in the current rocket nozzle study since the two applications have similar geometries.

Chen, *et al.* determined that the Taylor-Görtler vortices were responsible for the boundary layer transition in the wind tunnel nozzle. It is expected that Taylor-Görtler vortices will also trigger transition in the current investigation. The third task in the proposed research will be to predict the enhancement of the heat transfer due to the longitudinal vortices. To do this, a vortex array will be added to the boundary layer inflow of a three-dimensional compressible boundary layer computation. The wall heat flux will be determined and comparisons will be made with experimental results.

References

Chen, F.J., Malik, M.R., Beckwith, I.E., (1985), "Instabilities and Transition in the Wall Boundary Layers of Low-Disturbance Supersonic Nozzles", AIAA Paper 85-1573.

Smith, T.A., (1988) "Boundary Layer Development as a Function of Chamber Pressure in the NASA Lewis 1030:1 Area Ratio Rocket Nozzle", NASA TM 100917. Also AIAA Paper 88-3301.

Svehla, R.A., (1964), "Thermodynamic and Transport Properties for the Hydrogen-Oxygen System", NASA SP 3011.

Figures

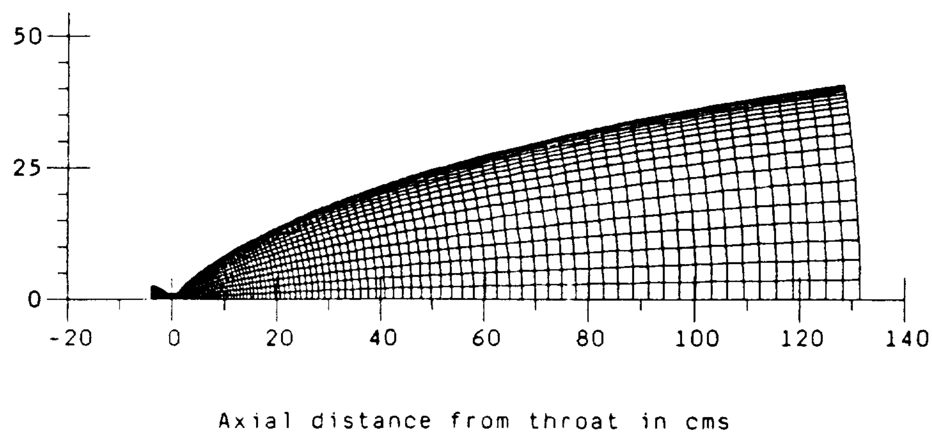

Figure 1. Computational grid.

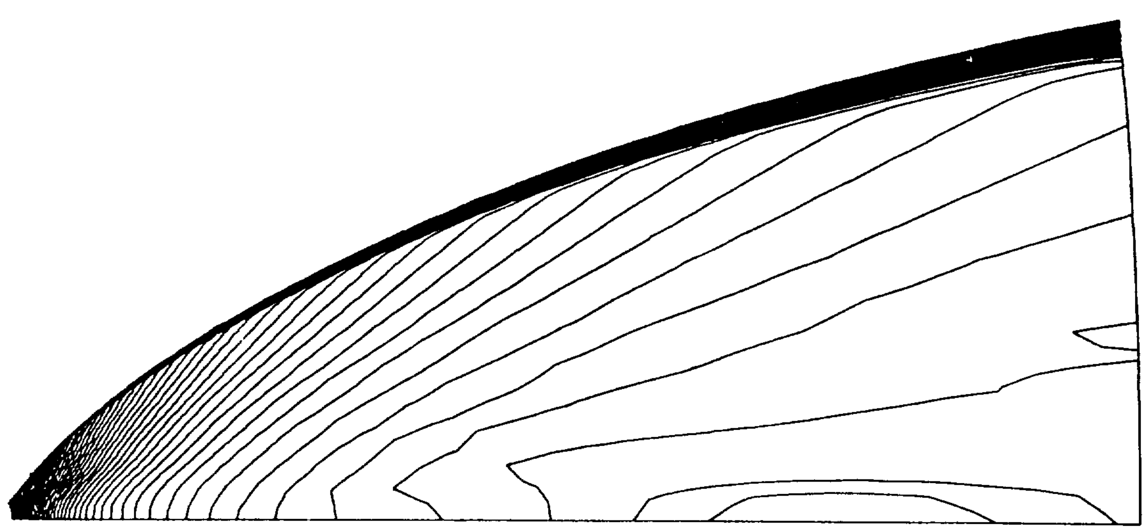

Figure 2. Velocity contours from computation.

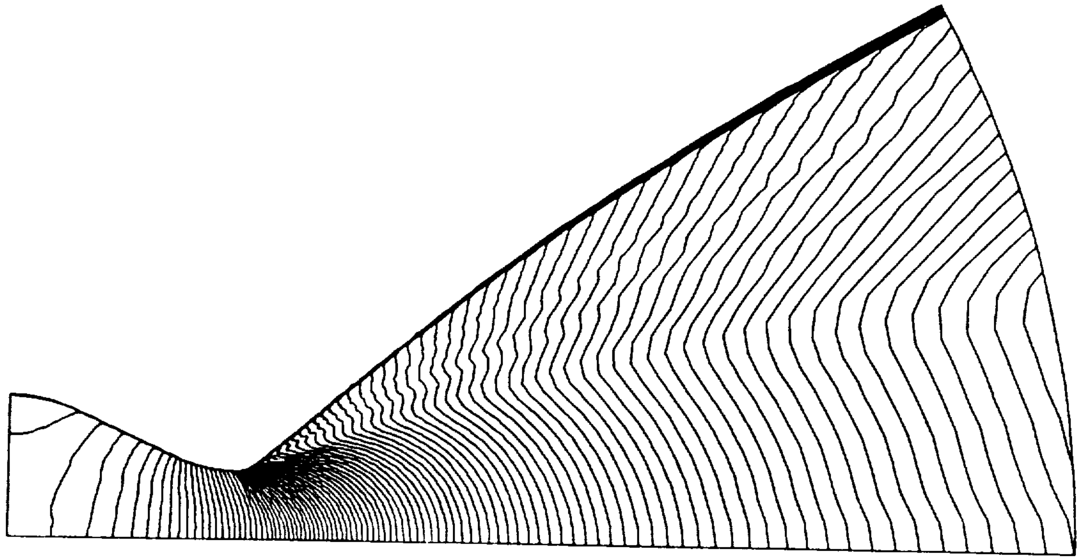

Figure 3. Mach contours near throat from computation.

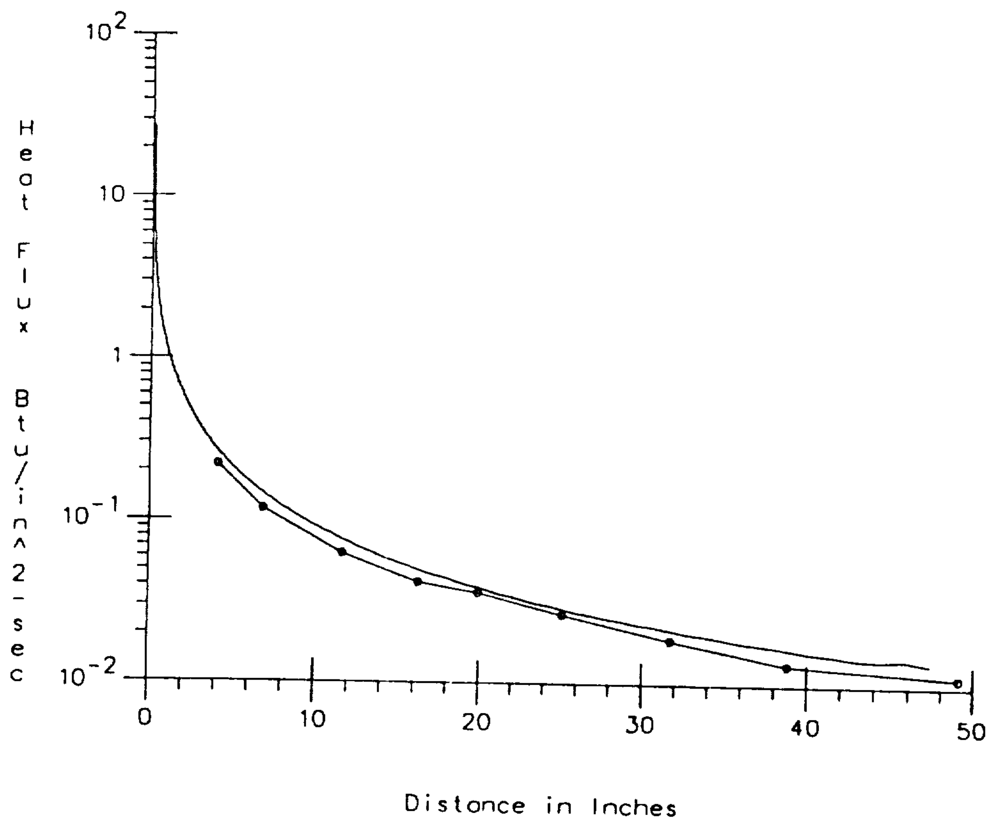

Figure 4. Experimental and predicted wall heat flux at a chamber pressure of 360 psia. (Smith, 1988)