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Numerical Simulation of Fluidic Modulation of Nozzle Thrust

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Abstract

Methods of modulating exhaust thrust in a converging-diverging nozzle by secondary fluidic injection are under investigation numerically. Modulation of thrust by fluidic means can be of significant benefit in some cases like the solid fuelled rockets which are difficult to throttle through conventional means. Two methods of Fluidic Thrust Modulation (FTM) are investigated. These are the Shock Thrust Modulation (STM) and the Throat Shifting Thrust Modulation (TSTM). In the STM method, shocks are induced in the supersonic flow by secondary injection in the diverging section of the nozzle to reduce the total pressure of the flow. In the TSTM method symmetric secondary injections are made near the nozzle throat to control the nozzle primary mass flow. While the initial STM simulations demonstrated the ability to modulate thrust, the configuration used was not able to reduce it. For the STM method the result indicated that pressure thrust was the dominant term when modulating the thrust. Total thrust, which is the sum of momentum thrust and pressure thrust, increased as the reduction in momentum thrust was much less than the increase in pressure thrust. For the TSTM case, the effect of slot size to throat size and the interaction of the parameters with NPR and injection angle had significant effects on the performance of the nozzle. Increased secondary to primary mass flow ratio increased the modulation of thrust in the TSTM method.

Introduction

One of the most important issues regarding the performance of a military aircraft is its manoeuvrability, the ability to alter its trajectory and speed using the least amount of time [1][2]. A conventional aircraft uses aerodynamic surfaces to alter its trajectory and these surfaces are usually located in specific areas of the aircraft such as the nose, wings and tail. A traditional aircraft's manoeuvrability is limited by aerodynamic constraints and cannot work in all conditions such as at high angle of attack or at low speed. To overcome these limitations, other ways of altering the force balance has been explored. One of these techniques is Thrust Vector Control (TVC). In the TVC method the exhaust flow of the propulsive jet is deflected from the centre line to transfer some exhaust momentum to the transverse axis, thus creating a force imbalance that allows the desired change in attitude and trajectory. Thrust vector control systems for aircraft generally use heavy actuators to vector the thrust. The actuation systems modulates the magnitude and direction of the force that vectors the thrust and can be either mechanical or fluidic [3].

Apart from vectoring, another aspect of thrust control is the modulation of thrust. Liquid fuelled aircraft engines modulate the thrust by varying the amount of fuel that enters the combustion chamber and by mechanically altering the throat and exit diameter of the nozzle. In liquid fuelled rocket this is achieved by simply controlling the flow of propellants into the combustion chamber. On the other hand solid fuelled rockets are very difficult to control. Once started, the combustion of solid fuel is very difficult to stop or slow down and usually burns at a predetermined rate [4]. Some solid fuelled rockets use a star shaped propellant core area so that at the start, a large surface is available for burning and later this surface area is reduced and less propellant is burned, but this cannot be changed in flight. Some other solid fuel rocket engines have used hatches on their sides to release the chamber pressure and control thrust. Some others use mechanical spoiling of the thrust [4]. Solid fuelled rockets have some advantages, like high thrust to relatively low cost, simplicity, compactness and reliability. The modulation of thrust in solid fuelled rockets is still very difficult and complicated, and hence motivates the current research. The practical application of a system that not only controls thrust direction but also magnitude would therefore be of considerable benefit for solid fuel rocket applications where attitude control is required or a degree of modulation is desired. Methods of modulating exhaust thrust in a converging-diverging nozzle by secondary fluidic injection are under investigation numerically and experimentally at UNSW@ADFA. This paper describes the initial numerical studies.

Fundamental Concepts

Thrust is the force that moves the rocket or aircraft through the air or space and is produced by the vehicle's propulsion system. The equation for total thrust for a rocket propulsion is

$$F = \dot{m}V_e + (P_e - P_a)A_e \qquad (1)$$

Here the first term in the right hand side is the momentum thrust and the second term is the pressure thrust. $\dot{m}$ is the mass flow rate, V_e is the axial nozzle exhaust velocity, A_e is the nozzle exit area, P_e is the exit pressure and P_a is the ambient pressure. From Eq. 1 we can see that the amount of thrust produced by a nozzle depends on the mass flow rate through the nozzle, the exit velocity of nozzle exhaust and the pressure of the nozzle exit plane. For fluidic thrust modulation we need to find an optimum combination of these parameters so that the thrust can be reduced when required.

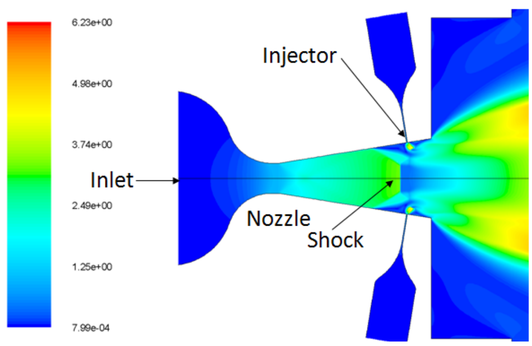

Figure 1: Mach no contours for secondary injection across the supersonic flow, clearly showing the shock formations.

Two methods of the Fluidic Thrust Modulation (FTM) are being investigated, Shock Thrust Modulation (STM) and Throat Shifting Thrust Modulation (TSTM). In the STM method secondary flow is injected axisymetrically into the supersonic flow in the diverging section of the nozzle and a normal shock is created by the interaction of the oblique shocks (Figure:1). Across a normal shock fluid flow becomes subsonic and total thrust is lost. This reduction in flow velocity has the potential to reduce the momentum thrust and total thrust.

In the TSTM method, thrust control occurs by shifting the throat of the nozzle. This method is based on controlling the throat area by symmetric injection near the throat for jet area control for modulation of the thrust.

Numerical Set Up

Computational fluid dynamics has been used successfully to study fluidic thrust vectoring [1][2]. NASA has conducted a number of computational investigations of fluidic thrust vectoring using their in house code PAB3D which can accurately predict propulsive flows with mixing separated flow regions and jet share layers [1]. To properly model the any experimental set-up conditions using CFD, a proper turbulence model needed to be used. A one-equation Spalart-Allmaras turbulence model was used for its computational efficiency. Neely et al. [5] effectively used this model to study SVC thrust vectoring in similar geometry. Ansys Workbench Suite is used for geometry creation (Ansys Design Modeller), meshing (Ansys Meshing) and flow simulation (Fluent).

Results

In the STM method, shocks are induced in the supersonic flow by secondary injection in the diverging section of the nozzle. The Nozzle geometry and stagnation conditions selected were representative of a small solid rocket. The nozzle had a divergent section length of 0.038m, a throat diameter of 0.007m and an exit diameter of 0.02m. The computational domain is shown in Figure 2. To improve computational efficiency half the domain was modelled and axisymmetrical solver was used. Simulations were performed for the no-injection case and these calculated thrusts were in good agreement (within 1%) with the experimental data for a small solid rocket with a matching nozzle and operating pressure ratio. Realated simulations on a previous study of fluidic thrust vectoring were validated against experimental measurements [5] but no matching experiments have yet been performed for the fluidic modulation case. An Experimental rig is being readied to perform these measurements.

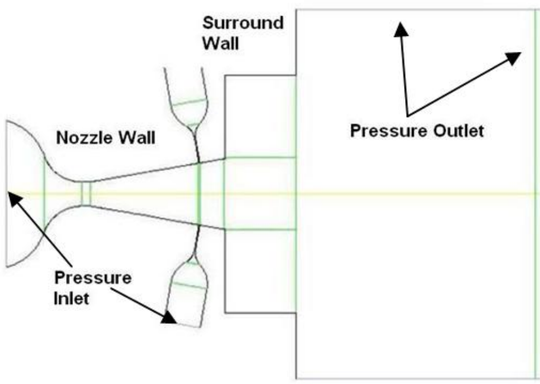

Figure 2: Geometry configuration and boundary conditions for STM method.

A grid dependency study was performed to verify the simulations. For an initial cell count of 71323 cells the momentum thrust was 614.27 N and for a refined cell count of 1206652, which was 17 times the initial grid size, the momentum thrust was 616 N. As the difference was less than 0.25%, the simulations were sufficiently grid independent. A circumferential injector of 1 mm was placed (Figure 4) halfway along the divergent section of the nozzle. A range of injection pressures were used to alter the depth of the jet penetration and by doing so alter the oblique shock generated and hence the shock -shock interactions, normal shock location and size. The simulations were conducted axisymmetrically, which created the injector effectively as an annulus around the entire nozzle. This initial model was found to actually increase the net thrust rather than decrease it. It was also found that the injector was located too far upstream as the flow had the opportunity to recover to supersonic speeds as shown in Figure 4. This situation was not ideal for the modulation of thrust as the total thrust produced is a function of both mass and nozzle exit velocity.

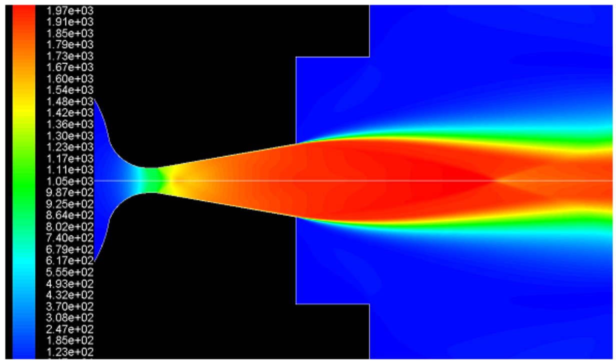

Figure 3: Velocity contours of axisymmetric nozzle flow (NPR=12).

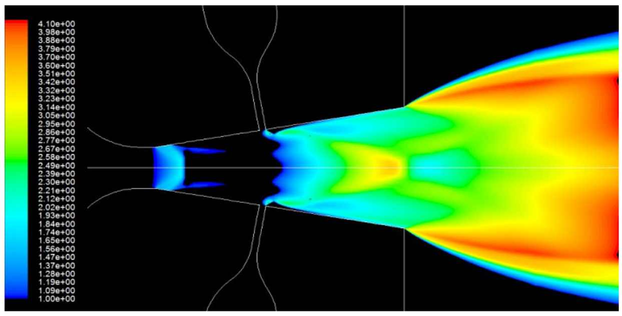

Figure 4: Mach number contours of initial axisymetric STM model. Subsonic flow in black.

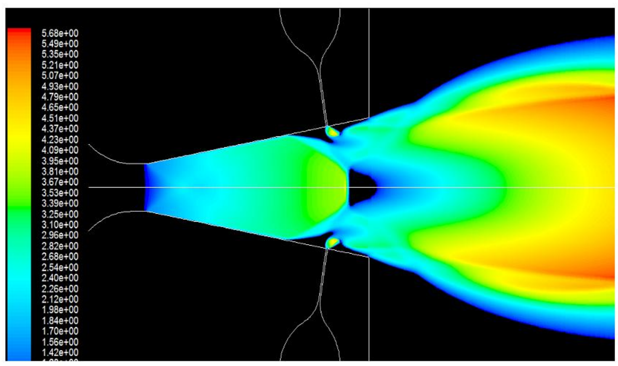

Figure 5: Mach number contours of the modified axisymetric model. Subsonic flow in black. (NPR=12)

The geometry (Figure 5) was modified by reducing the injector length by 75% and moving the injector from half way of the diverging section to 0.18 length of the diverging section from the nozzle exit plane resulting a total area reduction of 65% and injector area to throat area ratio of 0.38. This was done to reduce the amount of secondary mass flow as well as preventing flow velocity after the normal shock to gain supersonic speed before leaving the nozzle.

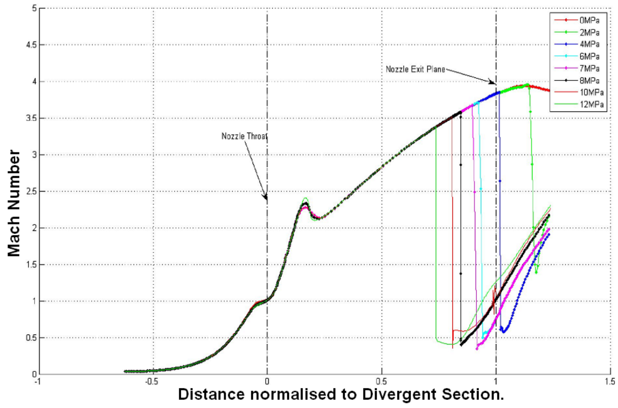

Figure 6: Mach number and location indicating shock locations.

A constant NPR of 12 was used in all simulations and injection pressure was increased in 2 MPa intervals upto 12 MPa. Injection pressure variation was used to adjust the location and size of the normal shock generated by the interaction of the oblique shocks. Figure 6 shows the Mach number along the nozzle axis. The flow was choked at the throat indicated by the sonic velocity at the throat. The normal shock was indicated by the rapid change of flow velocity form supersonic to subsonic. The location of the normal shock was dependent on injection pressure. Increased injection pressure caused the normal shock to move further upstream (Figure 5).

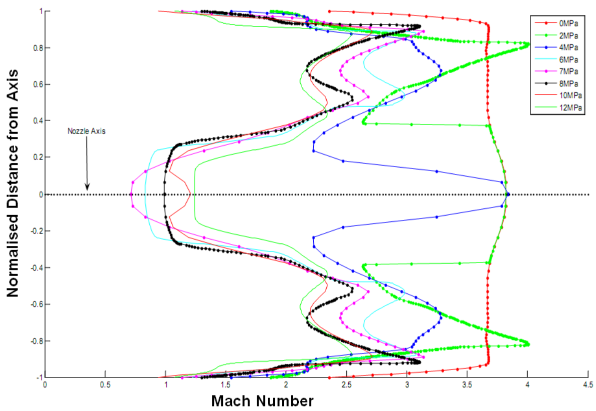

Figure 7: Mach number distribution over nozzle exit plane

Along with the presence of a normal shock , the velocity profile at the nozzle exit plane is also important when attempting to control the momentum thrust. There is a significant velocity change in the flow after passing the shock that significantly impact the thrust production. Figure 7 shows the Mach number distribution at nozzle exit. The greatest reduction in momentum thrust was found for the injection pressure that have regions of lowest Mach number at the nozzle exit. Maximum reduction in momentum thrust (about 4.4%) was found for a secondary injection pressure of 7MPa. Beyond 7MPa secondary pressure momentum thrust began to increase due to the added mass flow to the nozzle. From the thrust equation (Eq. 1) it was apparent that static pressure at the nozzle has the potential to be a significant contributor to the development of total thrust. Along the exit plane it is the integral of static pressure that provided the pressure thrust component of the total thrust produced. Figure 8 shows the planar distribution of static pressure at nozzle exit. Initially it was expected that the pressure thrust would be negligible as in the case of no injection scenario. But this was not the case. The initial amount of pressure thrust using no injection was 6.02N which increased consistently with greater injection pressure until a net pressure thrust of 103.95N was reached with maximum injection pressure, that being the same as the used operating pressure of the nozzle at 12MPa. Though there is notable decrease in momentum thrust there was a constant increase in the pressure thrust as shown in Figure 9. The increase in pressure thrust is significantly greater than any of the decreases in the momentum thrust. For this reason total thrust was not decreased in any test cases of secondary injection.

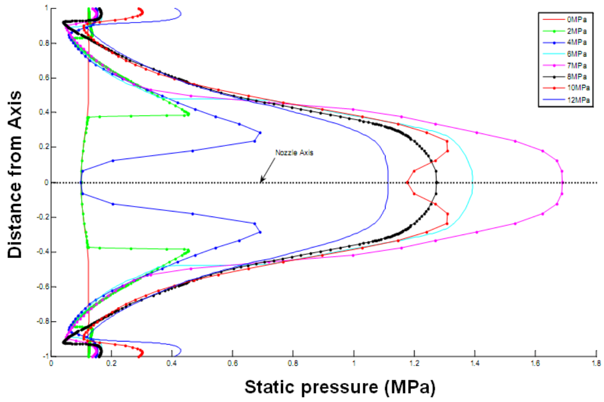

Figure 8: Static pressure distribution at nozzle exit plane.

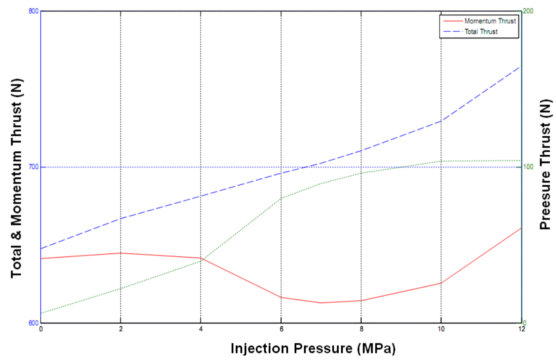

Figure 9: Thrust produced vs injection pressure.

In the TSTM method symmetric secondary injections were made near the nozzle throat. The injection of fluid at the throat causes separation of the primary flow from the nozzle wall at the throat of the nozzle. The injection of secondary fluid induces a high velocity shear layer in the region of the throat and effectively squeezes the primary flow to a smaller throat cross section and hence reduced the primary mass flow rate. Initially Design of Experiments [6] method was used to determine the significance of the factors affecting the modulation. Injector size, location, angle, and injection pressure ratio were the factors that were altered in our investigation. Factors were selected based on Deere et al. [1] highest achieving TS FTV nozzle. The initial TSTM simulations were performed with the symmetry line modelled as a 2D line of symmetry. This mean that the model was effectively reflected on the line of symmetry. The net thrust reduction was obtained by finding the percentage difference in thrust before and after fluidic injection $(T_0 - T_{inj})/T_o$.

The measure of effectiveness is given by percentage change in thrust divided by the ratio of primary mass flow to secondary mass flow.

$$\eta_{FTM} = \frac{NetThrustReduction}{MassFlowRatio} = \frac{T_0 - T_{inj}}{T_0} \div \frac{\dot{m}_s}{\dot{m}_p} \qquad (2)$$

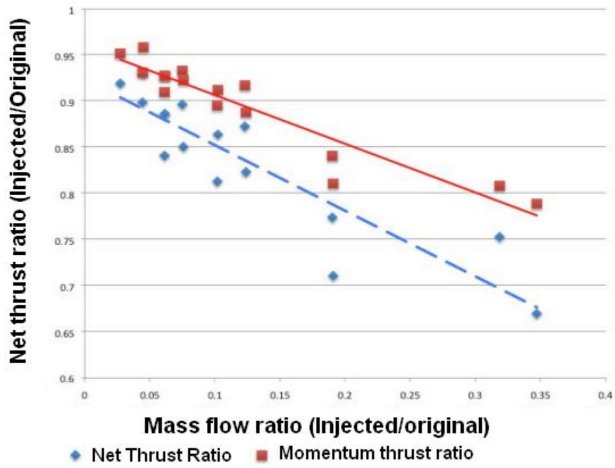

Figure 10: Effect of mass flow ratio on thrust ratio.

The average ηFTM level achieved was 1.66/%-injection. The highest ηFTM was 2.96/%-injection. Net thrust ratio against injected mass flow ratio plot is shown in Figure 10. The plot shows that the momentum thrust is decreased with increased injection, due to the constriction of the throat, which effectively reduced the primary mass flow rate. This result also shows unlike the STM method, the two effects of adding additional mass-flow into the system through secondary injection and the re-acceleration of the flow after the throat skewing is not significant enough to equal the decrease in momentum thrust in TSTM method.

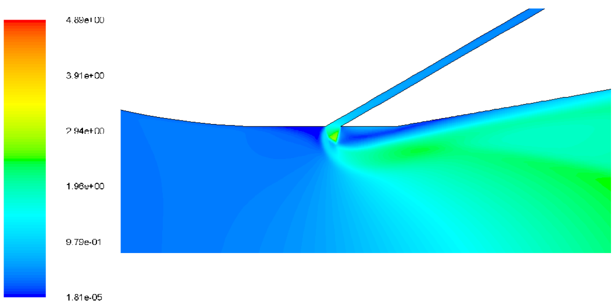

Figure 11: Mach number contours for secondary injection at throat.

As our objective was to use a combination of STM and TSTM method to modulate the thrust, another study of TSTM method with the same geometry configuration and boundary conditions and operating pressures of STM method were performed. Half the domain was modelled and axisymmetrical solver was used. A circumferential injector of 0.25 mm (Figure 11) was placed at the throat. The injector area to throat area ratio was 0.20. The injection angle used was 30^0. A constant NPR of 12 was used in all simulations and various secondary injection pressure was used. Figure 12 shows the thrust produced for different injection pressure ratio. It shows that unlike STM method the pressure thrust is not increasing with the increase of injection pressure ratio. Though in our 2D TSTM study and in other previous study the injection pressure ratio was few times higher than primary pressure, putting injection pressure ratio that high may not be practical in a rocket nozzle. In our axisymmetric TSTM study we only used a maximum sInjection pressure ratio of 2. The highest ηFTM level achieved was 1.88/%-injection for the secondary injection pressure of 10 MPa. Below 8Mpa secondary injection pressure, the injection flow was lower in pressure than the local pressure in the nozzle throat, hence there was a reverse flow in the secondary injector. While TSTM increased area ratio which increased exit velocity and can decrease pressure thrust and can even lead to over expansion which results in a pressure drag. Although in this case is it was two order of magnitude smaller.

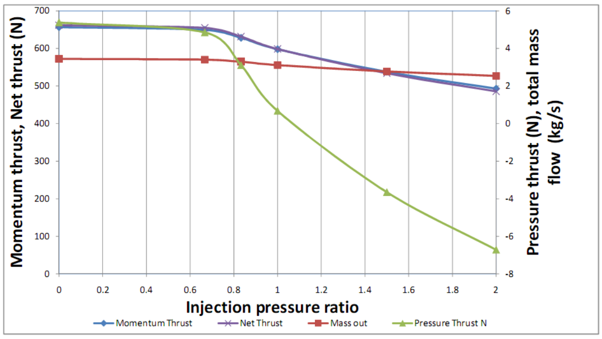

Figure 12: Thrust produced vs injection pressure

Conclusions

The STM method study is still ongoing and the conclusion that can be made based on the work done so far is that development of thrust may be modulated in a rocket nozzle using fluidic injection based on a sound understanding of the fluid physics involved. Three main parameters can be identified that impacted on the ability to change the thrust production. Secondary injection pressure regulates the thrust production for a fixed geometry and operating pressure. Another significant factor that strongly affect the thrust is the size of the injector and as such the added mass flow. The position of the injector also has a dramatic and significant impact on the generation and alternation of thrust production particularly in the presence of a generated normal shock. If injection occurs too far upstream flow velocity recovers with expansion and hence the optimal reduction in flow velocity at the exit plane is not achieved. Further parametric study is in progress to validate or disprove the method of fluidic injection for thrust modulation in rocket nozzle. For the TSTM case, the effect of slot size to throat size and the interaction of the parameters with NPR and injection angle had significant effects on the performance of the nozzle. Increased secondary to primary mass flow ratio increased the modulation of thrust in the TSTM method. Combination of the STM and the TSTM methods are currently under investigation.

References

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