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Three Dimensional Numerical Simulation of Rocket-Based Combined-Cycle Engine Response During Mode Transition Events

Jack R. Edwards, D. Scott McRae, and Ryan B. Bond
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March 2003

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This report contains preliminary findings, subject to revision as analysis proceeds.

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Overview

The GTX program at NASA Glenn Research Center is designed to develop a launch vehicle concept based on rocket-based combined-cycle (RBCC) propulsion [1]. Experimental testing, cycle analysis, and computational fluid dynamics modeling have all demonstrated the viability of the GTX concept, yet significant technical issues and challenges still remain. Our research effort develops a unique capability for dynamic CFD simulation of complete high-speed propulsion devices and focuses this technology toward analysis of the GTX response during critical mode transition events. Our principal attention is focused on Mode 1/Mode 2 operation, in which initial rocket propulsion is transitioned into thermal-throat ramjet propulsion. A critical element of the GTX concept is the use of an Independent Ramjet Stream (IRS) cycle to provide propulsion at Mach numbers less than 3. In the IRS cycle, rocket thrust is initially used for primary power, and the hot rocket plume is used as a flame-holding mechanism for hydrogen fuel injected into the secondary air stream. A critical aspect is the establishment of a thermal throat in the secondary stream through the combination of area reduction effects and combustion-induced heat release. This is a necessity to enable the power-down of the rocket and the eventual shift to ramjet mode.

Our focus in this first year of the grant has been in three areas, each progressing directly toward the key initial goal of simulating thermal throat formation during the IRS cycle:

- CFD algorithm development
- Simulation of Mode 1 experiments conducted at Glenn's Rig 1 facility
- IRS cycle simulations

The remainder of this report discusses each of these efforts in detail and presents a plan of work for the next year.

CFD Algorithm Development

The key tool used in this research is a validated Navier-Stokes solver for unsteady reactive-flow calculations on massively parallel machines [2–4]. Several enhancements have been added to this solver during the past year to make it more suitable for simulations of the complete GTX engine flowpath. These include:

1.) **Addition of modified Menter and Wilcox '98 turbulence models.** The original code used the Menter hybrid k-ω model with the shear-stress transport (SST) modification as its baseline turbulence model. In comparing with cold-flow Rig 1 data (discussed later), it was found that this model tended to produce larger regions of flow separation than indicated in the experimental data. This is a known weakness of the SST modification, but the obvious fix of simply removing the SST modification lead to excessive growth of turbulence kinetic energy in corner regions, where the grid spacing was not (apparently) small enough to allow for the proper near-wall behavior for the turbulence frequency ω. A solution was found by restricting the SST modification to the laminar sublayer of the boundary layer. This prevented excessive growth of turbulence kinetic energy while alleviating the tendency of the original SST model to reduce eddy viscosity within separated-flow regions. Before this simple fix was identified, we also coded the Wilcox '98 k-ω model, which is supposedly less-sensitive to free-stream values than the original k-ω model. Our results with this model were decidedly inferior to those of the Menter variants and as such, this addition was not pursued further.

2.) **Addition of a patched grid procedure.** The requirement to resolve details of the fuel injection ports located around the perimeter of the cowl required the addition of a grid patching procedure to allow better circumferential resolution of these features. This implementation follows from general procedures developed in [5], and it allows for conservative interpolation of data among patched grid regions. A procedure for integrating these regions within the MPI message-passing framework was also developed.

3.) **Addition of better implicit coupling among blocks**: One of the issues in applying implicit methods to solve large multi-block problems is a loss in effectiveness as the block numbers increase. This is due to the fact that there is typically no mechanism to couple solution corrections generated on one block with those generated on a neighboring block. We have devised a sub-iterative strategy that provides better information transfer among blocks and thus more rapid information transfer throughout the domain. The sub-iteration procedure attempts to solve the linear system formulated at each block more accurately while also accounting for coupling with neighboring blocks. Though more work per iteration is required, the additional robustness provided by this procedure makes it a useful addition.

4.) **Addition of time-derivative preconditioning**. Some of the GTX conditions result in large pockets of low Mach number flow in the engine. One way of alleviating time-step restrictions and enhancing the overall convergence rate is through the use of time-derivative preconditioning [6,7]. We have implemented a preconditioning algorithm described fully in Ref. [8] and are currently examining its effectiveness in accelerating the GTX calculations.

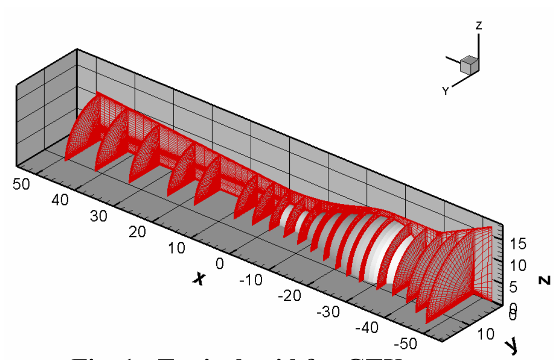

Fig. 1: Typical grid for GTX calculations

Other aspects of the simulations are as follows. GridGen Version 14 is used to generate grids for the GTX engine. These are somewhat complex due to the need to blend the circular rocket exit plane with the semicircular combustor section. Half-plane symmetry is invoked for all calculations. The grid sizes used in this investigation range from about 0.4 million grid cells to about 3.3 million grid cells, depending on the case. A sample grid is shown in Figure 1. As of now, we are using Tecplot Version 9 for visualization. Later this year, we will receive $10,000 from the NCSU College of Engineering as part of the cost-sharing agreement for this grant. We will use this to purchase a high-end Dell graphics workstation and will load the Ensight Gold visualization package on it. We expect that this will facilitate the production of quality animations of time-dependent flowfield responses during later stages of this investigation.

Simulation of Mode I Experiments in the Rig 1 Test Facility

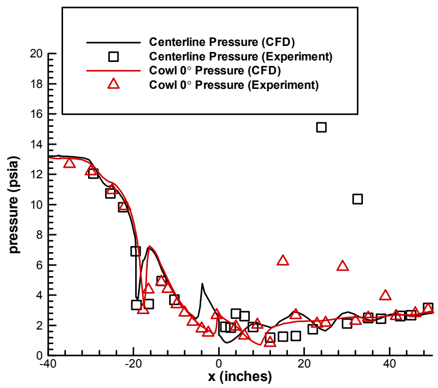

Fig. 2: Pressure distributions for ESP #39 case

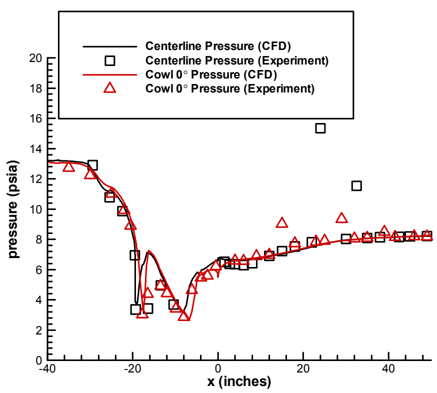

Fig. 3: Pressure distributions for ESP# 39 case

Simulations of two cases were carried out with no rocket or secondary fuel injection. These two cases corresponded to ESP #41 and ESP #39 from the Rig 1 test matrix. Both cases correspond to 9.94 lbm/s of air flow and an inlet total temperature of 547 R. The difference between the two cases is the back pressure, which is 3.1 psi for the ESP #39 case and 8.2 for the ESP #41 case. These cases were performed on an incorrect version of the geometry prior to the 2002 JANNAF conference, so they were repeated with geometry corrections and the changes to the turbulence model outlined above. The total number of grid cells is about 2.2 million. Figure 2 and Figure 3 present pressure distributions along the centerbody/flat plate surface and along the cowl surface at the y = 0 plane. The scale of the x-axis is referenced to station #3. Both cases resulted in a transition to supersonic flow at the minimum area location (station #2), followed by a compression and expansion region resulting from the changing flowpath area profile. Downstream of about x = -15, the two solutions differ. Both cases contained large regions shock-induced separated flow. In order to correctly model the exit plane, the grid was extended to x = 100 inches by extruding the exit plane as a straight semicircular duct. The ESP #39 simulations matched the experimental pressure along the cowl surface, but along the centerline, there were deviations from the experimental data caused by a difference in shock and expansion wave locations. The ESP #41 case matched the experiment almost exactly along both surfaces for the entire length of the rig.

Several ejector-ramjet (ER) runs have also been completed to date. ER operation corresponds to an operating rocket but no secondary fuel injection. Of the ER cases completed, only case ER 8687 had experimental data with which solutions could be compared. This case corresponded to an inlet mass flow rate of 10.016 lbm/s, inlet total temperature of 413 °F, and inlet total pressure of 13.4 psi. The rig back pressure is 2.6 psi, and the chamber pressure of the rocket is 750 psi. The oxidizer-to-fuel ratio of the rocket is 4 (4 parts oxygen by mass to 1 part hydrogen by mass), and the total number of mesh cells is about 3.3 million.

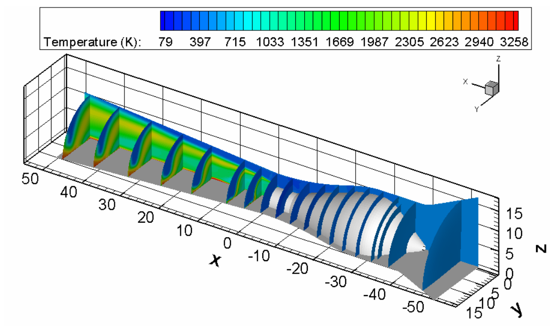

Fig. 4: Temperature contours for ER 8687

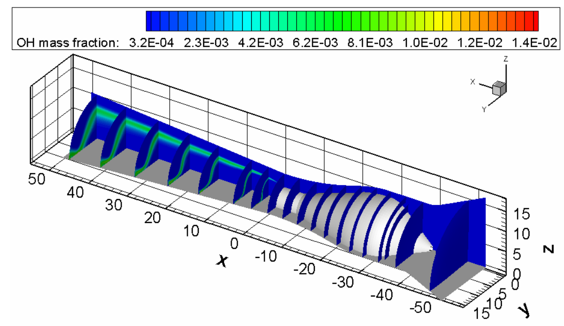

Fig. 5: OH mass fraction contours for ER 8687

Figure 4 shows temperature contours for this case. Temperatures of around 2500 K are found in the flame front, while temperatures of around 3200 K are found where the rocket plume impinges on the flat plate. The heat release is not enough to form a thermal throat, which is what would be expected for ER operation. The rocket exhaust enters the combustion chamber at approximately 1460 K, and the additional fuel in the exhaust ignites almost immediately. The flame front can be seen in the OH contours, shown in Figure 5. The mixing is incomplete, so the mixing layer does not extend through the entire air stream at the combustor exit.

Figure 6 compares the centerline and cowl pressures with the experimental data. The results are in good agreement with the experiment except for minor discrepancies along the cowl in the latter half of the combustor section. The higher pressures are attributed to an over prediction in the amount of heat release.

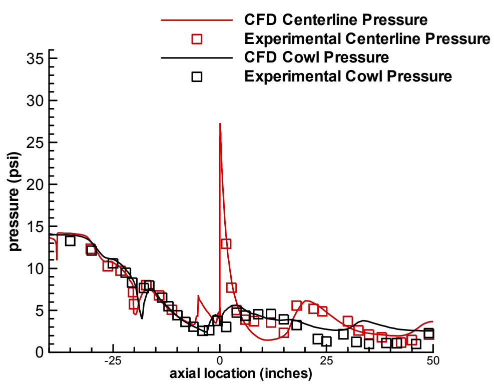

Fig. 6: Pressure distributions for ER 8687

IRS Cycle Simulations

Due to time limitations and other difficulties, no experiments corresponding to IRS cycle conditions were conducted in the Glenn Rig 1 facility this year. Over the last several months, we have attempted to simulate portions of the Rig 1 test matrix corresponding to IRS cycle conditions in an effort to determine strategies that lead to the proper response – the formation of a thermal throat in the secondary air stream followed by an expansion of hot reaction products to supersonic speeds. We have considered the Rig 1 geometry as the starting point and have used a smaller mesh of 0.8 million cells to allow for more rapid turn-around time. Our initial efforts in simulating the IRS cycle were hindered by our specification of a constant mass flow rate at the inlet of the geometry. While this is consistent with the manner in which the Rig 1 tunnel is operated, it does not appear to allow the formation of a stable thermal throat. Rather, choke points are alternately established at the physical throat of the engine, at a location midway along the combustor, and at the combustor exit. The solutions never converge as a result. Our most current efforts follow Steffen and Yungster [9] by enforcing a constant total pressure, rather than a constant mass flow rate, at the inlet. This allows the mass flow to adjust in response to heat release, and as a consequence, we have made some progress in simulating IRS operation. The current procedure for establishing a ramjet-type mode of operation in the secondary stream is as follows:

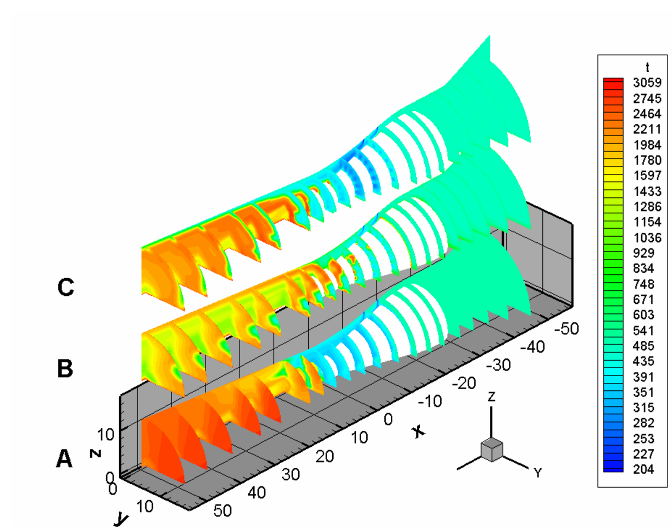

Fig. 7: Temperature contours during IRS cycle calculation

1.) Obtain a partial solution of the engine flowfield under purely subsonic conditions by enforcing a back pressure only slightly less than the enforced total pressure. As shown in the temperature contours of Fig. 7a and the hydrogen mass fraction contours of Fig. 8a, this leads to a very fuel rich combustion process, with a flame extending across to the cowl.
2.) Drop the back pressure to levels low enough to force an acceleration to sonic flow somewhere in the device.
3.) Restart the solution and run until the mass flow rate stabilizes.

Due to our current choices for the secondary fuel injection conditions, the last process actually results in flashback upstream of the physical throat. As the calculation converges, the flame position moves forward. Figures 7, 8, and 9 present temperature, hydrogen mass fraction, and Mach number at different stages of the calculation. After the flashback event, the flame front moves forward toward the nozzle exit. (Figs. 7b, 8b, and 9b). The combination of an effective area decrease due to volumetric expansion of the rocket plume, combustion induced heat release, and combustor wall divergence results in a gradual transition to sonic and then supersonic flow in the secondary air stream. This is nearly a ramjet mode response, with most of the combustion taking place upstream of the rocket exit plane and the combustor section serving as a nozzle for the expansion of hot products. As the calculation continues, the mass flow rate increases, and the thermal choke point shifts downstream toward the combustor exit. (Figs. 7c, 8c, and 9c) A secondary choke point is established downstream of the fuel injector locations. As of this writing, this calculation has not yet converged and it is unclear what the final state will be.

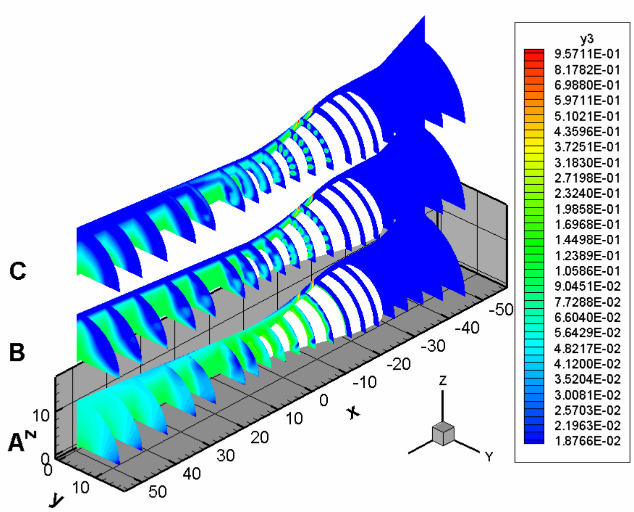

Fig. 8: Hydrogen mass fraction contours during IRS cycle calculation

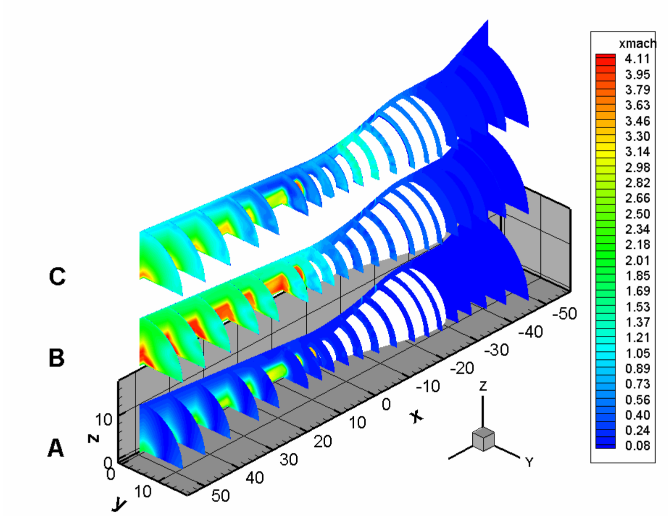

Fig. 9: Mach number contours during IRS cycle calculations

Plan of Work for Year 2

As we have not yet been able to firmly establish proper IRS cycle operation, our attention for the remainder of this year will be focused toward this goal. If simulations based on the Rig 1 conditions continue to be problematic, we will focus toward other points in the trajectory nearer to the point in which cycle analysis indicates thermal choking should first occur. We will also investigate the effects of other reaction mechanisms for hydrogen combustion. Steffen and Yungster were able to achieve a stable thermal choke point in their axisymmetric simulations using Jachimowski's hydrogen oxidation mechanism. [10] Once we have been able to converge to a thermally-choked solution, we will

1.) Investigate modulation of the thermal throat position by varying the fuel injection conditions. Cycle analysis indicates that this is a key for high propulsive efficiency. The types of modulation may include shutting off portions of the injector banks or simply changing the injection pressure. We will start from a converged solution for a particular positioning of the thermal throat and will vary the fuel injection conditions to shift the position upstream or downstream as necessary. We will conduct quasi-steady simulations of this type and if necessary, time dependent simulations. We also hope to develop a better means of initializing the IRS simulations so that flashback effects are avoided and will repeat successful simulations on finer grids to gain an understanding of the grid-independence of the predictions.

2.) Investigate power-down of the rocket and flame stabilization during the shift to ramjet mode. This will entail starting from a steady condition with an upstream positioning of the thermal throat, then powering down the rocket in a time-dependent manner. Detailed animations of the response of the flowfield under power-down conditions will be extracted, and the results made available for use in designing future Rig 1 mode transition experiments.

Publications

Two publications have resulted from this work this year:

1.) Steffen, C.J., Bond, R.B., and Edwards, J.R. "Three-Dimensional CFD Analysis of the GTX Combustor," Proceedings of the 26th JANNAF Interagency Propulsion Committee Meeting, JANNAF/CPIA, Destin, Fl. 2002.

2.) Bond, R.B. and Edwards, J.R. "CFD Analysis of an Independently Fueled Ramjet Stream Operation in a Rocket Based Combined Cycle Engine," Abstract accepted for presentation as AIAA Paper 2003-0017 at the 41st Aerospace Sciences Meeting, Reno, NV, Jan. 2003

Honors, Awards, and Other Miscellaneous Information

Ryan Bond was awarded the Ziglar Graduate Fellowship in Aerospace Engineering for the first and second years of his Ph. D. study. Ryan also tied for first place in the Ph.D category in the third annual MAE Department Graduate Student Research Posters Competition. His poster emphasized his work on the GTX project. Ryan also managed to pass his written Ph.D. preliminary exams and spent three weeks during the summer at NASA Glenn, working closely with the GTX team. Jason Norris will be joining the NCSU team in January of 2003. Jason obtained his Master of Science degree in Aerospace Engineering from NCSU in 1997and has been employed by Pratt & Whitney for the past five years. Jason will pursue a Ph.D in the general area of high-speed aeropropulsion, and it is hoped that his work might be funded under the present contract.

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[10] Jachimowski, C.J. "An Analytical Study of the Hydrogen-Air Reaction Mechanism with Application to Scramjet Combustion," NASA TP–2791, Feb. 1988.

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13. ABSTRACT ***(Maximum 200 words)***

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