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DEVELOPMENT OF CFD MODEL FOR AUGMENTED CORE TRIPROPELLANT ROCKET ENGINE

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INTRODUCTION

Many experts feel that the future of mankind depends on the economic accessibility to and effective use of space. Regardless of the approach taken, all human explorations will require delivery of substantial mass(personnel and equipment) to a low earth orbit. Therefore, it will be imperative that a dependable, affordable propulsion system be developed that will allow the economical exploitation of space to proceed in a rapid and orderly manner. The new propulsion systems will require the use of our best technologies, avoid immature or marginal designs, and a streamlined program that will bring rocket propulsion to the maturity of jet propulsion[1,2].

Early rocket pioneers recognized that the ultimate earth to orbit launch vehicle would consist of only a single stage, which discarded only propellants. Ideally, the launch vehicle would be all propellant; the airframe, tankage and subsystems would weigh almost nothing. As early as 1971, the mixed mode principle was suggested as a means of achieving single stage-to-orbit (SSTO)[1]. This principle involves the use of a dense propellant combination at liftoff, followed by a less dense, but higher performing combination at altitude. The benefit of the mixed mode system is that about 50 percent of the propellant would be spent in achieving 15 percent of the orbital velocity. When a high density propellant combination is used in the initial phase of flight, the resultant vehicle size and dry mass is less for a fixed payload mass. However, at this point in time the technology base for propellants or engines had not advanced to the point of achieving SSTO.

The Space Shuttle era has made major advances in technology and vehicle design to the point that the concept of a single-stage-to-orbit (SSTO) vehicle appears more feasible. In fact the recent NASA study "Access to Space" supports the concept that SSTO rockets could be demonstrated in the near term[3]. NASA presently is conducting studies into the feasiblility of certain advanced concept rocket engines that could be utilized in a SSTO vehicle. One such concept is a tripropellant system which burns kerosene and hydrogen initially and at altitude switches to hydrogen. This system will attain a larger mass fraction because LOX- kerosense engines have a greater average propellant density and greater thrust-to-weight ratio.

This report describes the investigation to model the tripropellant augmented core engine. The following sections will discuss the physical aspects of the engine, the CFD code employed and results of the numerical model for a single modular thruster.

PHYSICAL AND NUMERICAL MODEL

The tripropellant engine is composed of a main H_2/O_2 engine with smaller RP_1/O_2 thrusters arranged around the base of the main engine. An extended nozzle encloses the thrusters and main engine to form an bell/annular engine as seen in Figure 1.

The physical arrangement of the engine dictates the modeling process. Since the engine components are symmetric about the centerline of the main H_2/O_2 engine, the numerical model can treat the physical problem as axisymmetric. This means only one thruster will have to be modeled. Therefore, the engine can be modeled as three different problems: (1) thruster, (2) main engine and (3) combined exhaust nozzle. The solutions for the thruster and the main engine will be totally independent; however, the numerical model for the combined exhaust nozzle will require the converged solutions of the other two regions to start its solution process.

The numerical model has to be more than just a flow solver. The model will require that a chemistry model be incorporated to simulate the combustion process. A turbulence model will also be required to simulate the boundary layer at the walls as well as the shear layer that will be formed in the combined exhaust nozzle. In addition, a computational mesh must be generated such that the grid points are clustered in regions that will capture flow distrubances and form a well defined boundary layer.

NUMERICAL METHOD

The computational fluid dynamics (CFD) code GASP version 2.3 was used to model the Tripropellant Augumented Core Engine. The code solves the integral form of the time-dependent, three-dimensional Reynolds-Averaged Navier-Stokes (RANS) equations and its subsets subject to boundary and initial conditions. The code is a cell-centered, finite volume formulation with upwind based spatial discretization. GASP employes both explicit and implicit time integration methods to obtain solutions. GASP is a fully conservative shock capturing CFD code. In addition, the code gives the user the option of selecting from several turbulence, thermodynamic and chemistry models. For the high Reynolds number option, the two-equation k-epsilon turbulence model uses wall functions instead of artifical damping terms.

The GASP software does not include a complete grid generation package. Therefore, the user is responsible for importing a grid suitable for use by GASP. For this investigation, the computational mesh was generated by the computer code GENIE ++, which is a grid generation system developed by Mississippi State University.

RESULTS

The initial phase of this study was concerned with learning GASP. This was accomplished by developing solutions for each section of the engine using a perfect gas model for air on an unrefined grid. This solution process was accomplished on a Convex computer. This procedure allowed the user to develop a feel for the sensitivity of the code to CFL, boundary conditions and other user specified parameters.

The modular thruster was modeled with the computational grid shown in Figure 2. The thruster could not be treated axisymmetrically, therefore the entire thruster had to be modeled. The total computational grid was 81 axial by 31 radial points. Hyperbolic tangent stretching was used to cluster grid point axially at the inlet plane, throat region and the exit plane. Hyperbolic tangent stretching was also employed to cluster points radially at the nozzle wall. A nine species chemistry model, "NASP3", was used to model the RP_1/O_2 combustion [5].

The solution was initialized by running the coarse grid (41 axial by 16 radial) for approximately 1000 iterations. Then the solution was converted to the fine grid. It was discovered that if one tried to switch a well converged solution to the fine grid, the code's grid sequencing option just "bogged down" and the code would stop.

The thruster solution was started using frozen flow with a laminar boundary layer. The residual for this solution was dropped three orders of magnitude. The laminar solution was then used to restart a new solution using a Baldwin-Lomax algebraic turbulence model. The residual for this case was again allowed to drop three orders of magnitude. The Baldwin-Lomax solution was then used to restart a new solution using the low Reynolds number, Lam-Bremhorst k-epsilon turbulence model. This solution converged for two orders of magnitude and then began to diverge very rapidly. The solution was stopped. The Baldwin-Lomax solution was then used to restart a new solution using the high Reynolds number k- epsilon model with wall functions. As with the low Reynolds case, the residual initially dropped. As the subsonic section cleared, the residual began to oscillate every other iteration by three order of magnitude while continuing to drop.

The k-epsilon turbulence model is known to be difficult to initialize. But this does not appear to be the reason for the solution to go unsteady. An investigation of the grid points near the wall showed that the Y^+ was in the appropriate range for both cases. One possible explanation could be the thruster's geometry. The thruster's throat is skewed and the flow has to make an abrupt turn into the diverging section of the nozzle. Snapshots of the flow at different iterations indicate that this region could be causing disturbances as the flow tries to expand. Thus, causing the solution with the k-epsilon model to become unsteady.

Finally, the Baldwin-Lomax solution was used to restart the finite rate chemistry solution. This residual for this solution was dropped seven orders of magnitude. A Mach contour for this case is given in Figure 3. This solution process was carried out on a CRAY YMP computer.

CONCLUSIONS / FUTURE WORK

A CFD model for the modular thruster has been developed using GASP. The model incorporates the thermodynamics, turbulence and chemistry required to simulate the RP_1/O_2 combustion process. Initial results indicate that model can be used to analyze thruster contours and operating conditions. This model can now be extended to the main engine and combined exhaust nozzle.

Future tasks should include: 1) Modify thruster grid and determine if the grid spacing near the wall is the cause of the turbulence model going unsteady. 2) Using results of thruster grid investigation, modify main nozzle grid and develop solution. 3) Develop solution for the combined exhaust nozzle. 4) Investigate the slipstream formed in the combined exhaust nozzle and determine if it affects the engine's performance like that reported in Reference 6. 5) Finally, investigate the "engine out problem" when the modular thruster are shutdown at altitude.

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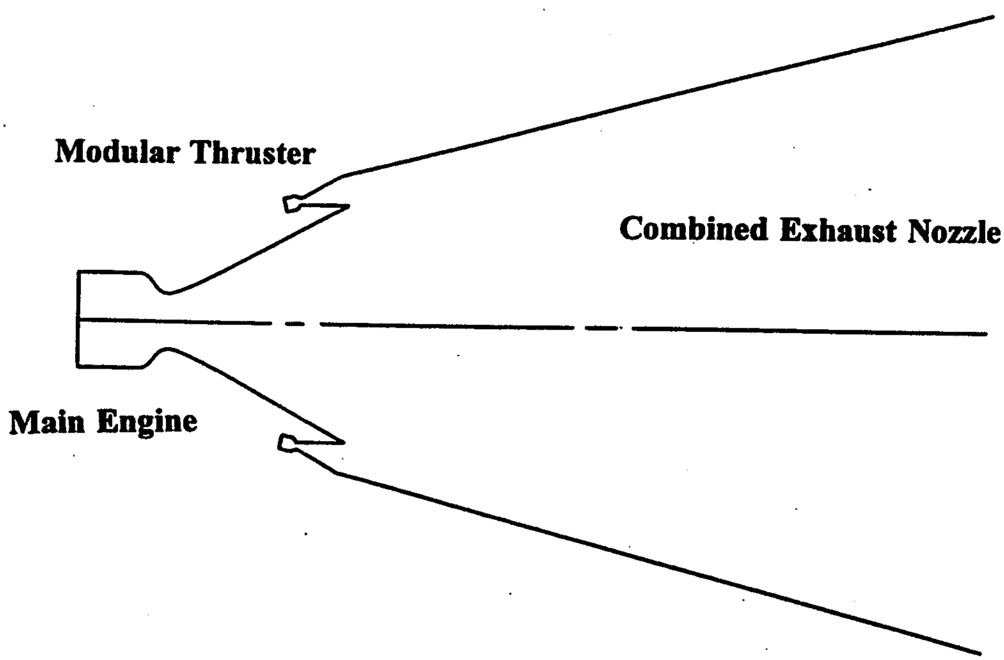

Figure 1: Layout of Augmented Core Engine

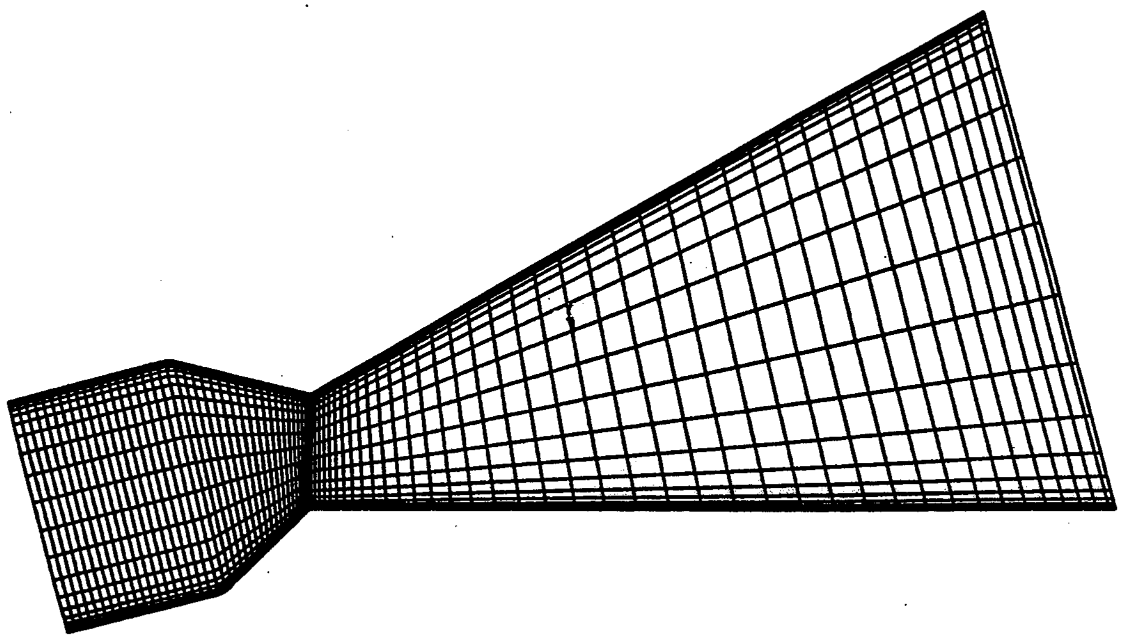

Figure 2: Computational Grid for Modular Thruster

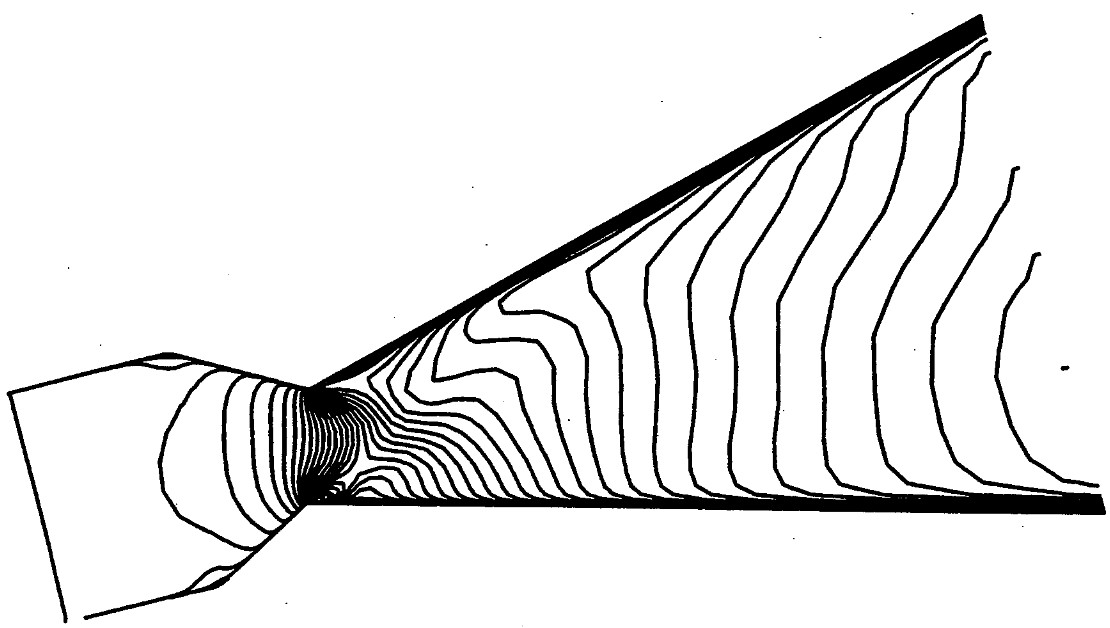

Figure 3: Mach Contours for Thruster with Turbulence and Finite-rate Chemistry