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RAMSCRAM—A Flexible Ramjet/Scramjet Engine Simulation Program

Leo A. Burkardt and Leo C. Franciscus
Lewis Research Center
Cleveland, Ohio

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Leo A. Burkardt and Leo C. Franciscus
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

ABSTRACT

With the resurgence of interest in high supersonic and hypersonic flight there is a need to simulate airbreathing engines which may be used in this flight regime. To meet this requirement the RAMSCRAM code was developed. The code calculates one-dimensional flow properties at each component interface and the overall performance of the engine. It uses equilibrium thermodynamics which accounts for dissociation and allows for any fuel or combination of fuels. The program can simulate ramjet, scramjet, rocket, and ducted rocket engines.

INTRODUCTION

In the past few years, there has been a resurgence of interest in high speed flight. Such programs as the High Speed Civil Transport and the National Aerospace Plane have highlighted this interest. Ramjet and scramjet engines perform very favorably in the high supersonic and hypersonic flight regimes. Ramjets can be effectively used from about Mach 2 to Mach 6 and scramjets give reasonable performance at Mach 6 and above.

The RAMSCRAM program described herein was developed at NASA Lewis Research Center to meet the need to calculate the performance of ramjet and scramjet engines. The goal in developing this program was to obtain high fidelity simulation while maintaining ease of use, versatility and reasonable calculation times. In pursuit of this goal, performance is calculated by marching from front to back through the engine flow path in a stepwise fashion. One-dimensional calculations are performed at each step with efficiency factors to account for flow path losses. Program input is made easy through the use of Fortran Namelist input.

Equilibrium thermodynamics is included to account for high temperature real gas effects. The general nature of the thermodynamics subroutine, together with the flexible mixer and combustor components, gives the RAMSCRAM code the ability to simulate a large variety of engine/propellant combinations. Some examples include: any chemical rocket or ducted rocket, multifuel ramjet engines, oxygen enriched scramjet engines.

This paper presents a general description of the calculation methods used in the code and a demonstration of some of its capabilities through the use of examples.

FLOW PATH DESCRIPTION

Figure 1 is a schematic of the simplest flow path used in the RAMSCRAM program. Calculations are performed at each flow station shown in the schematic. Station 0.0 is the beginning of the forebody (external compression surface) and has free stream conditions associated with it. Station 1.0 defines the beginning of the inlet (internal compression portion). Station 2.0 defines the throat of the inlet or the end of the mixer if a mixer is to be included. Station 3.0 is the diffuser exit/combustor entrance location. Station 4.0 is the combustor exit. Station 4.5 is the nozzle throat. Station 5.0 is the nozzle exit.

THERMODYNAMIC ROUTINE

Due to the high temperatures that can be reached in ramjet and scramjet engines, we felt that equilibrium thermodynamics would be needed to maintain good accuracy. At the time we were developing this code, the CET86 (1) general equilibrium code was being modified to run as a subroutine to the NNEP (2) turbine engine simulation code. It was convenient therefore to use this version of the CET86 code for the RAMSCRAM program also.

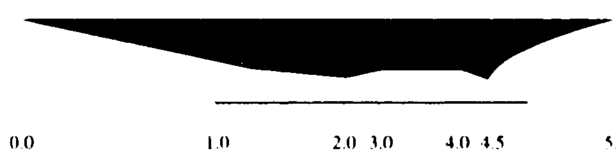

FIG. 1 - RAMJET STATION LOCATIONS.

In addition to the modifications made to CET86 for use with NNEP, further modifications were made to allow more than a single fuel/oxidizer combination to be specified. Up to 24 reactants can be specified as possible fuels or oxidizers during a single run with the appropriate combination of those reactants specified at each location in the engine. This permits calculation of propulsion systems such as ducted solid rocket.

Forebody

A conical, two-dimensional wedge or no forebody can be specified. Figure 2 presents a schematic of the forebody.

In the case of a cone, the conical half angle is specified and a conical shock is calculated. The conical flow field is integrated across the station 1.0 inlet height and a mass weighted average is calculated to obtain one-dimensional flow conditions at that location.

If a two-dimensional wedge is specified, up to 20 ramps can also be included. This allows an isentropic compression surface to be approximated. The shock off each ramp is calculated. If the combination of Mach number and ramp angle at any ramp will not support an attached oblique shock, then a terminal normal shock is calculated at that point and the subsonic flow field is assumed to persist down the rest of the forebody.

The inlet can be specified to be either started or unstarted. If it is started, the conditions after the oblique shocks specify the flow field at station 1.0. If the inlet is specified as unstarted, then a further normal shock calculation is performed as part of the forebody calculation to give the properties of the flow a station 1.0.

Inlet

In the simplest case, the internal inlet consists of everything from station 1.0 to 3.0. The flow characteristics at each station are based on those calculated at the previous station and the flow area and efficiency terms given for the current station. The efficiency terms are always relative to conditions at the previous station. If no efficiency factor is provided an isentropic calculation is performed.

The flow field at station 1.0 defines what is coming into the inlet. An efficiency factor can be applied to the flow at this point to account for any losses not accounted for in the forebody calculations. The efficiency factor can be in one of three forms: total pressure recovery, kinetic energy efficiency or process efficiency. Process efficiency is similar to kinetic energy efficiency, except that it is referenced to diffuser exit static conditions rather than total conditions. If desired, it is entirely possible to leave the forebody undefined and account for all losses at this point using the efficiency factor. Two of three parameters, airflow, Mach number or flow area, are input to permit calculation of the flow conditions and geometry at this location. If only airflow or flow area are given, then the Mach number is assumed to be the terminal Mach number from the forebody calculation or the free stream Mach number if no forebody is included. If flow area and airflow are input, then the subsonic or supersonic solution is chosen to match the condition of the air just upstream of this station. It can also be forced to either solution by entering a subsonic or supersonic Mach number which will be used as a first guess in the calculation procedure.

Station 2.0 can be used to determine the flow field at an intermediate point in the inlet, such as the throat of a ramjet inlet. The same efficiency factor and either Mach number or flow area are input to calculate the flow conditions. If desired, a normal shock can be included in the calculation at this point when the inlet is started.

The flow conditions at station 3.0 are calculated in the same manner as described for station 2.0, with the exception that there is no provision for a normal shock calculation.

Mixer

The inlet can be configured to perform as a mixer in the case of a ducted rocket or ejector ramjet simulation. In this case, station 1.0 becomes the airstream entrance to the mixer and station 2.0 becomes the mixed plane. If it is not convenient to use station 1.0 as the mixer entrance, then another station can be included to follow station 1.0. This station is designated station 1.5 and it has the same characteristics as described for station 2.0 in the inlet section above. The rocket stream enters the mixer at station 1.2.

The mixer model is defined by a control volume as shown in Fig. 3. Streams A (air stream) and B (rocket exhaust) enter the control volume at flow station 1.5 and 1.2, respectively, and mix inside the control volume. The flow is assumed to be fully mixed and departs the control volume at station 2.0.

The mixed flow conditions at station 2.0 are calculated by simultaneously solving the equation of state and the conservation of mass, momentum and energy equations. This leads to a quadratic equation. The two solutions represent a low and a high entropy gain. The high entropy solution always results in subsonic flow and the low entropy solution normally results in supersonic flow. The code allows the choice of either solution. Normally the subsonic solution would be chosen.

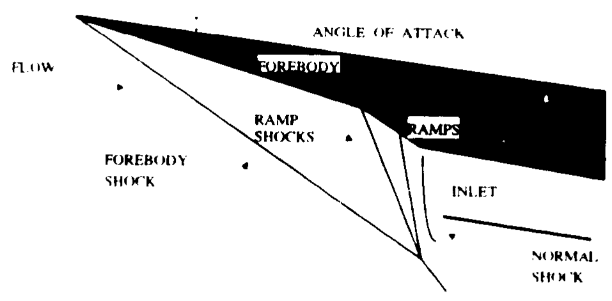

FIG. 2 - FOREBODY DIAGRAM.

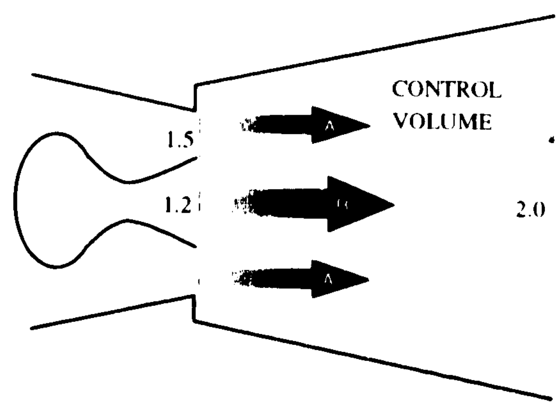

FIG. 3 - MIXER MODEL.

The mixer calculation allows the inclusion of efficiency factors to account for losses in the mixing section. These factors include a momentum loss term for either incoming stream, a wall friction loss term and a total or static pressure loss term. The pressure loss term is referenced to mixed pressure before the loss is calculated.

The mixing section does not have to be constant area. If a diverging or converging area is defined, then a pressure-area term must be accounted for on the walls. This term is approximated by assuming that the pressure acting on the wall is the average of stream A and mixed stream static pressures.

The use of equilibrium thermodynamics also accounts for any burning occurring in the mixing process by maintaining equilibrium chemistry at the mixed plane.

Combustor

The combustor model is essentially the same as the mixer model. In this case, the fuel stream replaces the rocket exhaust stream of the mixer. Two fuel streams are permitted to allow the injection of two different fuels simultaneously. Also, up to ten sequential fuel injection stations are permitted along the length of the combustor. A ramjet combustor is represented by specifying the subsonic solution and a scramjet combustor is represented by specifying the supersonic solution. Using this model for the combustor allows the momentum of the fuel to be accounted for as well as its energy. The fuel momentum can be a significant contributor to scramjet thrust.

Incomplete combustion is represented by including "inert fuel" in the fuel composition. Currently "inert hydrogen" and "inert methane" are available, but others can be easily added as needed. Other combustor efficiency factors are similar to those used for the mixer.

Nozzle

The nozzle calculation is an isentropic expansion. A throat calculation is provided whenever the velocity coming out of the combustor is subsonic. The nozzle can be allowed to expand fully to free stream pressure or it can be terminate at a given area. No distinction is made between internal and external expansion sections of the nozzle. Two efficiency factors are provided, a velocity coefficient and a thrust coefficient.

EXAMPLE SIMULATIONS

Ducted Rocket

A ducted rocket was simulated and an analysis of its performance versus air flow to propellant flow ratio was completed. The rocket burns a composite solid propellant mixture of ammonium perchlorate oxidizer and ethylene oxide fuel. The net thrust was held constant while the air to propellant bypass ratio was varied. As the bypass ratio was varied, the mixture ratio of the propellant was also varied to give enough excess fuel to provide a stoichiometric fuel to air ratio. The specific impulse and air capture area of the rocket versus bypass ratio is plotted in Fig. 4. The capture area is normalized with thrust.

A bypass ratio of 6 was arbitrarily chosen as the maximum simulated. At some point, the solid propellant mixture ratio becomes such that it cannot support combustion. The specific impulse increases linearly with bypass ratio. However, as shown, the air capture area also increases with bypass ratio. This offsets the gain in performance, as the inlet and combustion duct will increase in size and weight. The ultimate compromise between the two offsetting trends must be accomplished by doing a vehicle analysis using the information provided by the engine performance code.

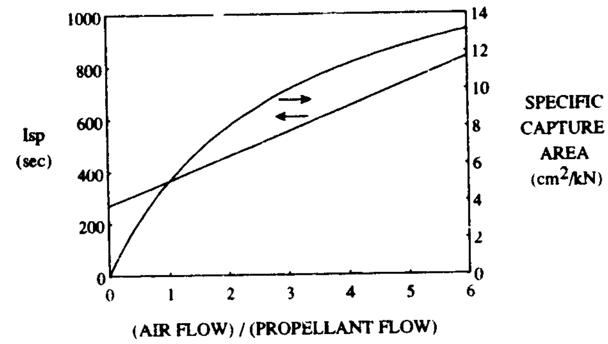

FIG. 4 - DUCTED ROCKET PERFORMANCE.

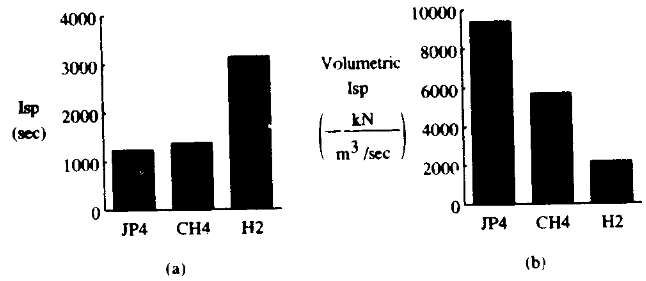

FIG. 5 - PERFORMANCE COMPARISON OF CANDIDATE RAMJET FUELS.

Ramjet Fuel Comparison

A ramjet was operated at Mach 5 using three different fuels. The general thermodynamics routines used in this program make this type of comparison a very simple procedure, requiring only a change in the specified fuel to air ratio and the fuel type. Figure 5 illustrates the performance of the engine with each fuel.

Figure 5(a) presents I_{sp} and Fig. 5(b) presents volumetric I_{sp}, which is defined as thrust divided by volumetric flow rate. The three fuels represented are JP4, liquid methane and liquid hydrogen. As shown, hydrogen gives by far the best performance based on mass flow rate, but it also gives by far the worst performance based on volumetric flow rate because of its very low density. The structural weight of the vehicle required to accommodate the fuel and the larger drags associated with the greater volume of hydrogen or methane play an important part in the tradeoff between fuels. Here again, the performance provided by the engine code would be used in a vehicle analysis to determine the best fuel for specific mission requirements.

Scramjet Sensitivity Analysis

A sensitivity analysis was performed for a Mach 10 scramjet. The inlet kinetic energy efficiency (η_{KE}), combustion efficiency (η_B), and nozzle thrust coefficient (C_{FG}) were varied. The baseline values used were: 0.97 for η_{KE}, 0.95 for η_B, and 0.96 for C_{FG}. Figure 6 presents the results of this analysis.

Each parameter was varied over a range of from approximately 0.9 to 1. Kinetic energy efficiency was varied with two different downstream constraints. In one case, the diffuser exit/combustor inlet geometry was held constant. In the other case the diffuser exit/combustor inlet Mach number was held constant. In the first case η_{KE} had only a small effect on engine

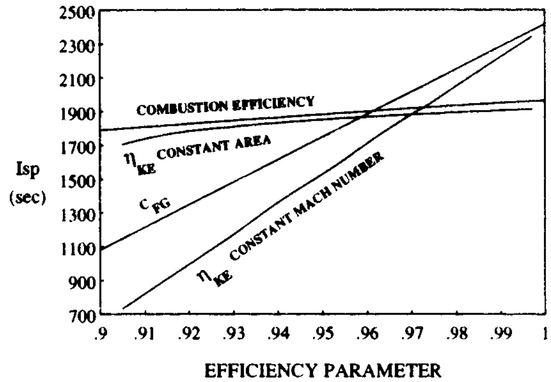

FIG. 6 - SCRAMJET SENSITIVITY ANALYSIS AT MACH 10.

performance because there was a self compensating effect of the burner Mach number decreasing as η_{KE} decreased. This resulted in lower Rayleigh losses in the burner. Combustion efficiency also had only a small effect on the performance over its range of variation. Both η_{KE} with constant burner Mach number and C_{FG} had large effects on the engine performance. This is because they both directly affect the gross thrust on about a one-for-one basis. Net thrust is only about 15 percent of the gross thrust and therefore any variation in gross thrust is magnified in the net thrust and specific impulse values.

CONCLUDING REMARKS

There has been a resurgence of interest in hypersonic propulsion. Air breathing flight in this regime requires the use of ramjet and scramjet propulsion systems. Vehicle and engine performance and tradeoff studies require a reasonably accurate, versatile and fast running propulsion simulation program.

The RAMSCRAM code fulfills this requirement. The general form of the geometry inputs allows the simulation of many engine configurations. Component efficiency inputs make it easy to study component performance effects on overall engine performance. And the general nature of the equilibrium thermodynamics routines allow the specification of nearly all conceivable propellant combinations.

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