Analysis of Liquid Rocket Engine Exhaust Plumes

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A comprehensive examination of liquid rocket engine exhaust plume flowfields has been made, and models for calculating specific regions of the plume developed. The combustion chamber flowfield is calculated by two programs, one which utilizes the viscous boundary-layer equations with equilibrium chemistry and one which uses an inviscid streamtube equilibrium chemistry concept. Two-dimensional finite-difference programs with chemistry effects and oxidizer/fuel gradients are used for the transonic analysis. The supersonic nozzle and plume flowfield is calculated by a method-of-characteristics program which includes chemistry effects, oxidizer/fuel gradients, and shocks. The free molecule flow regime is calculated where applicable by a source-flow calculation. The effects of the nozzle boundary layer and condensation within the plume are also included in the high-altitude plume analysis. Turbulent mixing of the plume with the external air stream is calculated by a finite-difference solution of the boundary-layer equations with finite rate chemistry. The results of the turbulent mixing analysis are then superimposed on the inviscid plume flowfield. Several liquid rocket engines have been analyzed by these programs and comparisons made with available experimental data.

Introduction

IN general, rocket engine plumes are high-energy multi-component flowfields which expand into the surrounding environment. Although the major portion of the plume flowfield is supersonic, there are also subsonic regions present. The basic structure of a low altitude plume with an external supersonic freestream is shown in Fig. 1. For flow into a vacuum, the internal shock pattern and the free shear layer no longer exist, but the additional phenomenon of transition from continuum to free molecular flow in the plume flowfield must be considered. The plume flowfield is ultimately dissipated by viscous mixing, by shock waves, or by dilution into a space environment. Previous publications[1-3] have described the state-of-the-art in general plume technology and have indicated areas where additional work is required.

The over-all plume description starts with an analysis of the thrust chamber flowfield, then continues through the transonic throat region and the nozzle, and finally considers the plume itself. The major effects initiated in the thrust chamber and/or nozzle that affect the plume [that is, shock waves, chemical reactions, radial oxidizer-to-fuel ratio (O/F) gradients, viscous mixing, boundary-layer effects, noncontinuum effects, and condensation] are included in the analysis.

Combustion Chamber

Prior plume analyses have ignored the effects of O/F gradients which may result from the injector design. An oxidizer/fuel gradient, which gives rise to nonisoenergetic flows, can have a large impact on the plume flowfield.[4]

The hostile environment of the combustion chamber has severely hampered experimental investigation of the details of the combustion process, making mathematical treatment of this region difficult. A major problem of interest to the plume specialist is the description of the propellant mixing and, in the case of liquids, droplet breakup and evaporation. These processes are highly engine-oriented and not easily modeled. Presently, it is necessary from the plume prediction standpoint to accept efficiency numbers from an engine performance calculation and to modify the inlet enthalpy appropriately.[5]

When the O/F gradient is known, an analysis of the combustion process is possible. The initial combustion process can be modeled by a multiple-stream tube approach[6] which yields the postcombustion conditions under the assumptions of inviscid flow with no lateral pressure gradient. Generally, it is advisable to determine the effect of mixing on the stream properties because there can be large velocity discontinuities between the streams. A ducted mixing program[7] which solves the turbulent boundary-layer equations with either equilibrium or finite rate chemistry can be used to perform these calculations. An appropriate eddy viscosity model for combustion chamber flows remains to be determined. The mixing calculation may be used from the postcombustion state throughout the remainder of the cylindrical combustor.

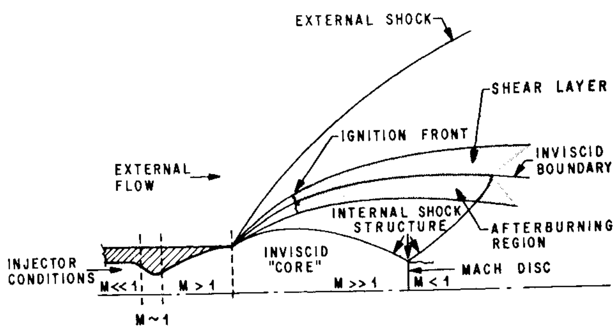

Fig. 1 Basic structure of low-altitude plumes.

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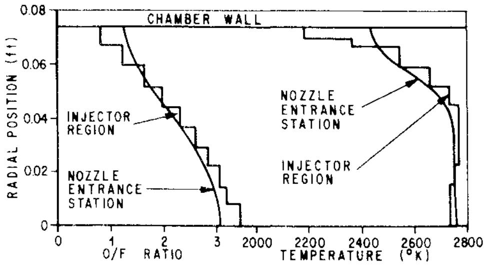

Fig. 2 Effect of mixing in an R-4D rocket motor.

In the converging section the shearing forces diminish, whereas the accelerations due to expansion increase, making it possible to discontinue the mixing solution and proceed with inviscid calculations. In addition, in most conventional nozzles the zero lateral pressure gradient assumption used in the mixing calculation becomes questionable at this point as the geometrical throat effects increase. An example of a mixing/combustion calculation for the R-4D rocket motor is shown in Fig. 2.

While the thermochemistry analyses used in current calculations is derived by free energy minimization principles, a new mathematical model[8] for equilibrium rocket engine combustion is being investigated in which the throat choking condition is introduced as a constraint in the equilibrium analysis. This alters the post-combustion condition significantly, and if this approach is verified, uncoupling the chemistry process from the fluid mechanics will no longer be possible in subsonic combustion chamber regions.

Transonic Region

In the transonic region, the gas dynamic equations of motion behave in a mixed elliptic-hyperbolic fashion. Two basic approaches to the solution of these equations have been taken. One approach neglects the radial momentum equation and includes finite rate chemistry and viscous effects. Neglecting the radial momentum equation causes the system of equations to be parabolic and allows the solution to be obtained by integrating forward from a known subsonic inlet condition. Some of the existing treatments are 1) an inviscid stream tube solution treating fuel striations,[6] 2) a viscous, equilibrium mixing-solution,[7] and 3) a viscous, finite rate chemistry solution.[7] The standard set of ICRPG recommended programs[5] is also available.

The second approach includes the radial momentum equation which results in a set of mixed partial differential equations. A practical solution to these equations precludes a finite rate chemistry analysis. Both unsteady[9-11] and relaxation[12-14] techniques have been applied to obtain finite-difference solutions to the transonic problem.

A typical transonic calculation is shown in Fig. 3[4] for both a one-dimensional and an axisymmetric solution for the R-4D engine. As the flow approaches the throat, the geometrical effects begin to dominate and the solutions deviate markedly.

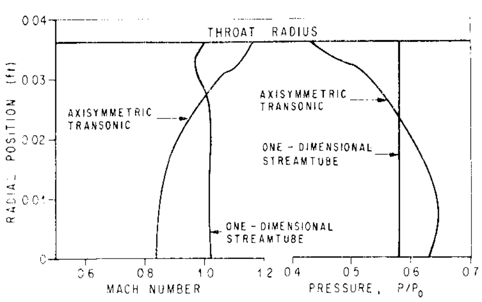

Fig. 3 Mach number and pressure distributions at the nozzle throat for the R-4D engine.

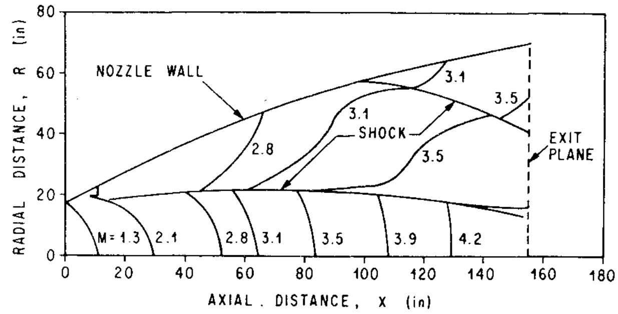

Fig. 4 F-1 engine shock structure and Mach number contours.

Nozzle and Plume

Inviscid Nozzle and Plume Flowfield

The analysis discussed in the previous two sections determines initial conditions for the nozzle calculations which must precede the plume calculations. A method-of-characteristics program,[15] which is a numerical solution of the governing hyperbolic partial differential equations, is used together with the nozzle wall boundary conditions to calculate the nozzle flowfield. This program can compute flows with equilibrium, frozen, or equilibrium/frozen thermochemistry. Shock waves initiated by geometric or flow phenomena are accommodated. Although only one shock may be considered in a single calculation, the program can be restarted to compute any number of shock waves.

Typical Mach number contours for a large liquid rocket engine (Saturn F-1) are shown in Fig. 4.[16] Bell nozzles of this type generally give rise to weak shocks which propagate through the flowfield, gradually increasing in strength as they approach the nozzle axis of symmetry. In addition to this shock, the discontinuity at the junction of the contour and the conical extension on the F-1 generates an additional shock wave (Fig. 4).

By using the method-of-characteristics program[15] with a pressure boundary instead of the nozzle wall contour boundary, the inviscid plume can be calculated. The plume boundary pressure is determined by Newtonian Impact Theory, a tangent-wedge approximation, or a quiescent back pressure.

Figure 5 shows the inviscid plume boundary and internal shock structure of a single F-1 engine. In all rocket engine exhaust plumes, a repetitive shock structure is present. For an axisymmetric plume, an incident shock wave approaching the plume centerline always results in a Mach disk. The shock structure consists of an incident shock wave joined by a

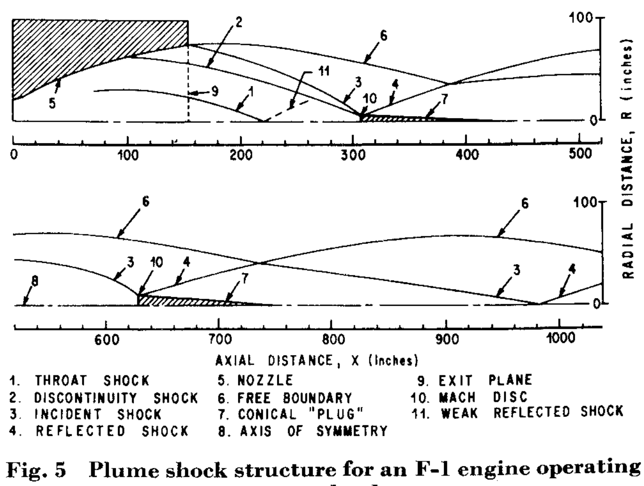

Fig. 5 Plume shock structure for an F-1 engine operating at sea level.

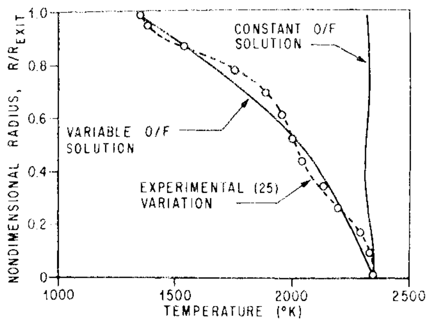

Fig. 6 O/F gradient effects on Atlas Vernier exit plane temperature.

strong shock leg (Mach disk) and a reflected leg which propagates to the plume boundary. Although no entirely satisfactory analytical solution for the Mach disk problem is currently available, several methods can be used to obtain approximate disk locations. These methods specify either the strong shock wave angle,[17] the pressure immediately behind the strong shock,[18] that the pressure downstream of the incident shock is a minimum,[19] or that the subsonic flow behind the strong shock becomes sonic at some point downstream.[20] The shock structure and slipline information obtained from a Schlieren or a regular photograph can be used to continue the calculation of the flowfield.[21]

Flowfield Chemistry

In the high-temperature, low-velocity regions of the flowfield, an equilibrium chemistry calculation[22] is a good assumption. However, as the flow accelerates through the nozzle, significant deviations from equilibrium conditions occur, and the flowfield chemistry usually approaches a frozen condition. This calculation is best handled by a two-dimensional finite rate chemistry analysis.[23]

If a two-dimensional finite rate analytical procedure either is not available or is not practical, an approximation can be made using an equilibrium chemistry analysis up to a certain point in the flow system and then changing to a frozen chemistry analysis. This "freeze" point can be determined by performing a one-dimensional finite-rate analysis[24] along typical flowfield streamlines.

Mixture Ratio Maldistribution

In most cases, the effects of the initial O/F gradient discussed in the section on combustion chambers are found throughout the entire nozzle and plume flowfield.[25] Mixture ratio gradients are included in the method-of-characteristics analysis by the use of Crocco's Theorem. The rotational term in the compatibility equation is modified to account for the nonisoenergetic effects. A typical example of the O/F gradient effect is shown in Fig. 6. Significant improvement results from including the mixture ratio effect in the calculation. Figure 7 shows O/F gradient effects on the exit plane

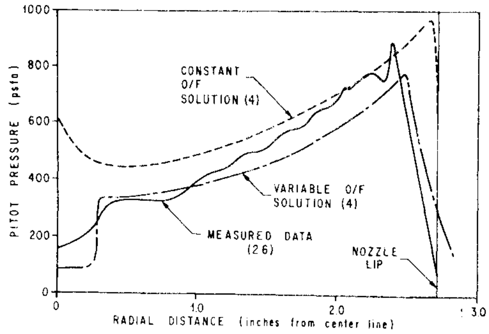

Fig. 7 O/F gradient effects on R-4D exit plane pitot pressure.

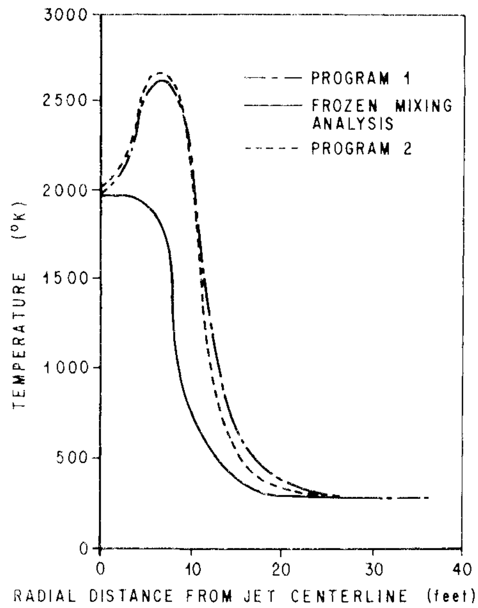

Fig. 8 F-1 engine shear layer temperature profiles at an axial position of 50 ft.

Pitot pressure distribution for a Marquardt R4D RCS engine.[26] O/F gradients also significantly affect engine performance.[27]

Free Shear Layer

The free shear layer is generated by turbulent mixing of the plume with the external air flow. There is in general no solution that is completely satisfactory for the entire viscous shear layer, but, by using the turbulent boundary layer equations with appropriate boundary conditions, a turbulent eddy viscosity model, and a chemistry model, portions of this region can be calculated. Use of the turbulent boundary-layer equations restricts these solutions to regions in the plume where the radial pressure gradient is small. This condition occurs near the exit plane where the shear layer is thin and in the plume far field. Two methods have been developed for this analysis.

One program[28] solves the boundary-layer equations with an explicit finite-difference technique. The chemical composition is calculated from a set of linearized finite rate chemistry equations. This program is the more realistic representation of the physical situation. The second program[29] solves only the boundary-layer momentum equation with an explicit finite-difference technique. The O/F radial profile is assumed to be similar to the velocity profile. An equilibrium table is then used to relate local composition and temperature to local O/F.

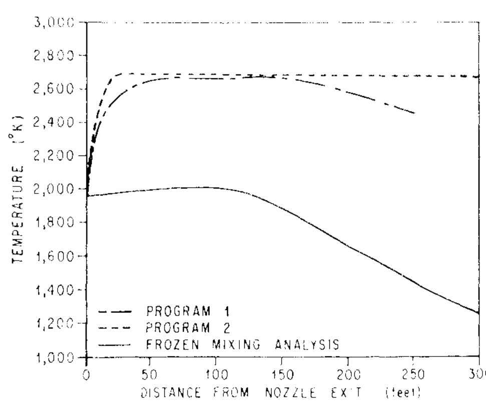

Fig. 9 Maximum shear layer temperatures for the F-1 engine.

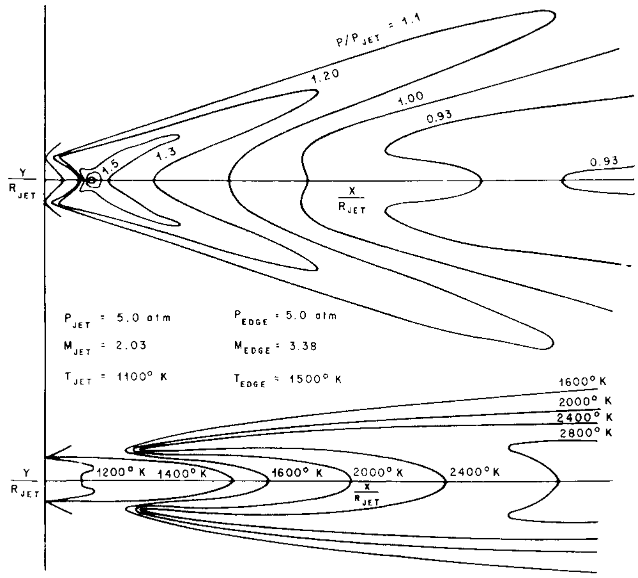

Fig. 10 Pressure balanced hydrogen jet exhausting into air.

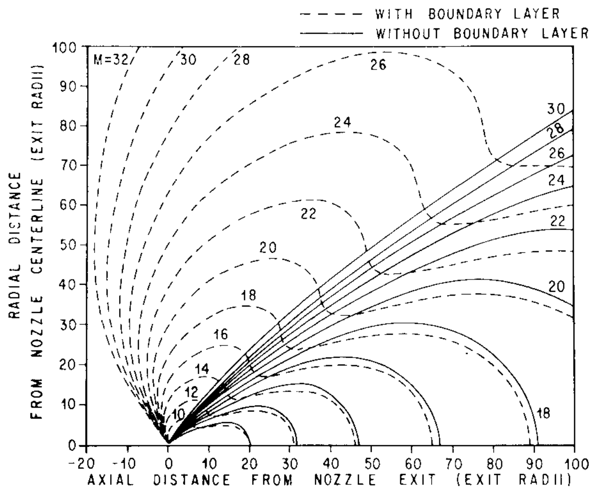

Fig. 12 Mach number contours for an inviscid plume with and without nozzle boundary layer.

The shear layer region of an F-1 engine operating at sea level was analyzed by these two methods.[30] The predicted radial temperature distribution is shown in Fig. 8. Figure 9 shows the maximum temperature within the shear layer at each axial location. In addition, a mixing analysis with frozen chemistry is included to illustrate the necessity for including chemical reactions in the shear layer analysis.

A program[23] has been written which uses the method-of-characteristics theory with viscous terms as forcing functions in the compatibility equations. Since this program includes the radial momentum equation, it is possible to calculate viscous flowfields with radial pressure gradients. The thermodynamic and chemical constituent properties are calculated using finite rate chemical reaction equations. Figure 10 illustrates the results of this program.[23]

The choice of a turbulent eddy viscosity model can have a significant impact on the results of the shear layer calculations. As a result of a lack of appropriate experimental information on reacting jet mixing, the most realistic eddy viscosity model cannot be determined at this time.

Nozzle Boundary Layer

In cases where turbine exhausts are injected into the nozzle or the engine is operated in a vacuum environment, the boundary layer can significantly affect the plume. A computer program,[31] developed for either laminar or turbulent boundary layers, solves the integral boundary-layer momentum equation with a stepwise numerical integration along the nozzle wall. The velocity profile and skin-friction formula used in this study program are based on Clauser's work.[32] A Crocco temperature distribution through the boundary layer and the mass injection scheme of Spalding[33] are used. The effects of the interaction of the viscous flow with the inviscid flow, along with experimental data, are shown in Fig. 11.

For engines operating at vacuum conditions, the boundary-layer effect can significantly influence the plume as shown in Fig. 12.[34] The plume with the boundary layer was calculated by assuming an inviscid expansion of the supersonic portion of the boundary layer at the exit plane. The increased expansion of the plume is due to the low Mach numbers in the boundary layer. Although this deviation was expected at the outer edge of the plume, the boundary layer was found to affect the entire plume to some extent.

Noncontinuum Effects

Two methods have been developed for predicting the rarefied region of the plume. One method "freezes" out the various energy modes of the gas molecules as the gas becomes increasingly rarefied. Since a large number of intermolecular collisions are necessary to maintain vibrational equilibrium and because the number of collisions steadily decreases with increasing rarefaction, vibrational equilibrium eventually can no longer be maintained. In a similar manner, the rotational, and eventually, the translational energy modes

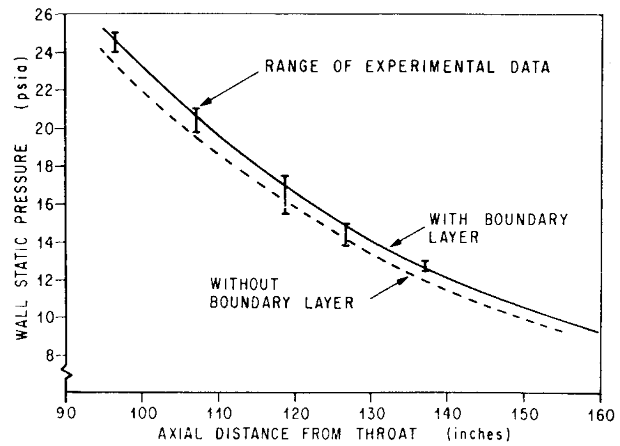

Fig. 11 Experimental and calculated wall pressure on F-1 nozzle extension.

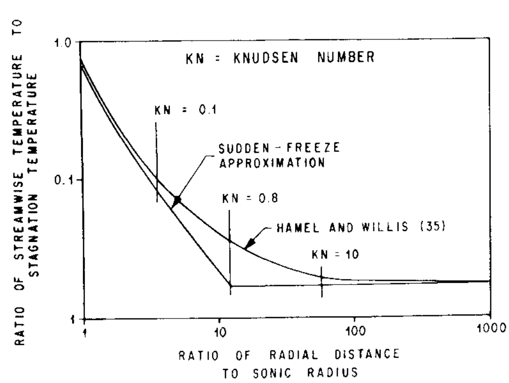

Fig. 13 Temperature comparison for a spherical source flow expansion.

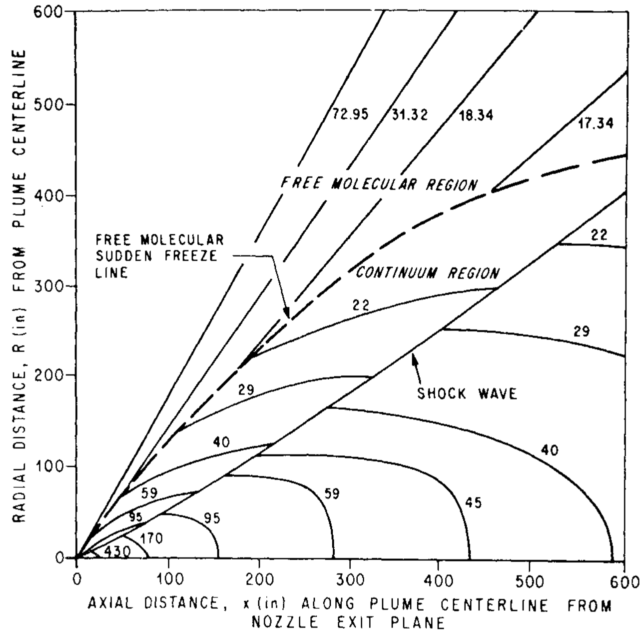

Fig. 14 Temperature contours (°R) in the R-1E engine plume flowfield.

also freeze. The flow is then free molecular and the calculations are continued along streamlines (considered straight because freezing occurs at hypersonic Mach numbers where the flow is sourcelike) with the temperature and velocity held constant. The density in this region varies inversely with the cross-sectional area of the streamtubes formed by adjacent streamlines. A local Knudsen number is used as the freeze criterion.

The results of this method are compared with the solution of the Boltzmann equation of Hamel and Willis[35] for spherical source flow in Fig. 13. The over-all agreement of these two results is quite good, although, as to be expected, there are significant differences at the freeze point. An example of a typical vacuum plume expansion is shown in Fig. 14 where only translational freezing has been considered.

A more rigorous approach to the translational nonequilibrium problem was also investigated.[36] Moments of the Boltzmann equation were used, together with the BGK approximation of the collision integral, to form a hyperbolic set of equations which was then solved numerically by the method of characteristics. The method was then applied to an axisymmetric plume with uniform parallel flow at the exit plane. The axial temperature distribution for both temperature components is shown in Fig. 15. For the case where the translational degree of freedom remains in equilibrium until hypersonic Mach numbers are reached and the streamlines become straight, there is little difference between continuum and noncontinuum calculations of density and velocity.

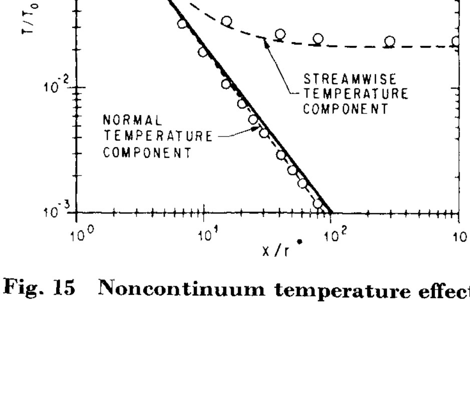

Fig. 15 Noncontinuum temperature effects.

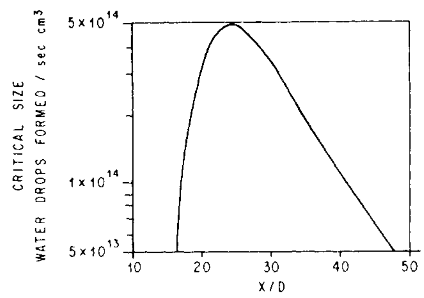

Fig. 16 Nucleation rate for the R-1E engine.

Condensation Effects

Operation of rocket engines at high altitudes usually results in supersaturation of one or more of the species present in the plume. Because of the high stagnation temperature, condensation usually does not occur in the expansion until hypersonic Mach numbers are reached. A finite rate analysis of the phase change process is required because of the rapidly changing gas environment of the condensed particles.

A program[37] which solves the one-dimensional flow equations with finite rate phase change for a single ideal gas has been extended to include a noncondensable carrier gas. This program was used to calculate the condensation of water and carbon dioxide for the center streamline of several rocket engine plumes exhausting into a vacuum. The results shown in Fig. 16 for the R-1E engine are similar to the results found for the other engines.

Unlike flows with relatively low stagnation temperatures, there is no discrete "condensation shock" in rocket exhaust plumes. Because the plume density is decreasing approximately as the square of the distance, the nucleation rate and the rate of growth eventually become negligible, leaving the gas supersaturated with a very large number of submicron size particles.

All analyses of condensing flows require experimentally determined drop properties and accommodation coefficients. Although these analyses have been successfully applied to flows with low stagnation temperatures, their application to rocket engine plumes remains to be verified.

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