Figure 3 illustrates the computed entry vehicle pitch rate history. This together with the other computed rates are of importance to the guidance and control system and have been found, from case studies, to be dependent on numerous factors, many of which are concerned with parachute initial conditions and subsequent inflation motion. This demonstrates one important problem area in the simulation process—that of determining the parachute aerodynamic input. Little data are available for an inflating parachute trailing in the wake of a large diameter forebody under supersonic and transonic conditions. While sensitivity studies using the simulation program may be useful, wind-tunnel tests to determine the aerodynamic coefficients at the proper Mach numbers would contribute greatly to accurate computer simulations.

Conclusions

An advanced two-body six-degree-of-freedom computer model employing an indeterminate structures approach has been developed for the parachute opening process. The program determines vehicular responses and trajectories based on known input. Where uncertainty exists in the input, sensitivity analysis may be performed to isolate the important variables. The program has demonstrated the importance of suspension-line elasticity and parachute aerodynamics in the parachute opening process.

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Jet Penetration at a Sonic Throat

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Nomenclature

a = sonic velocity
$A(M)$ = area ratio function for one-dimensional isentropic flow
B = duct height
C = injection parameter defined in Eq. (2)
d = jet slot width
F_x = x-component of force acting on jet face
h = depth of jet penetration
m = mass flowrate
M = Mach number
p,P = static and stagnation pressure, respectively
V = velocity
y = coordinate of jet face normal to duct wall
α = angle of injection (positive for upstream injection)
γ = ratio of specific heats
$\pi(M)$ = pressure ratio function for one dimensional isentropic flow

Subscripts

j,m,o = jet, mainstream and initial quantity, respectively
r = reference mainstream condition (without jet flow)

Superscript

$*$ = sonic conditions

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THE problem of a transverse jet issuing into the sonic throat of a nozzle for the purpose of throttling the primary flow has been studied by numerous investigators. Reviews of these works are available in the literature.[1] Nunn and Brandt[2] have presented an analysis for the two-dimensional case that assumes that a normal shock occurs in the jet flow at the point of maximum jet penetration. This simplification does not take into account the effect of the curvature of the jet plume upon jet shock location and is, therefore, a poor model of the actual jet penetration process. This note describes an improved analysis that avoids the explicit description of the jet shock. In addition, new experimental data are given for increased values of the jet/freestream total pressures, P_j/P_m.

A two-dimensional control volume (dashed line in Fig. 1) is defined as bounded by the windward face of the jet plume from the point of injection to the jet sonic line (dotted line in Fig. 1), the jet sonic line to the wall, and the plane of the wall. A momentum balance yields

$$F_x - p_j{}^* h = m(a_j + V_{jo} \sin\alpha) \tag{1}$$

where F_x is the x-component of the force per unit depth acting on the windward face of the jet and will be evaluated in subsequent paragraphs. Following simplifications similar to those utilized by Barnes et al.[3] (the jet is assumed to behave as an ideal gas undergoing an adiabatic process), Eq. (1) reduces to

$$\frac{F_x}{P_j d} = \frac{1}{A(M_{jo})}\left(\frac{2}{\gamma+1}\right)^{\gamma/\gamma-1}\left[\gamma(1 + M_{jo}\sin\alpha) + 1\right] \equiv C \tag{2}$$

Neglecting shear forces acting on the windward face of the jet,[1,2] the remaining pressure force may be written

$$F_x/P_j d = 1/P_j d \int_0^y p\, dy = C \tag{3}$$

It is assumed that the pressure p acting on the jet face can be reasonably approximated by the static pressure, p_m, calculated on the basis of one-dimensional isentropic mainstream flow. This assumption is clearly justifiable near maximum jet penetration where jet and mainstream flows are parallel, but is less

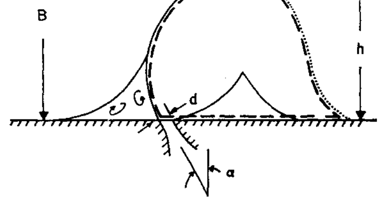

Fig. 1 Control volume.

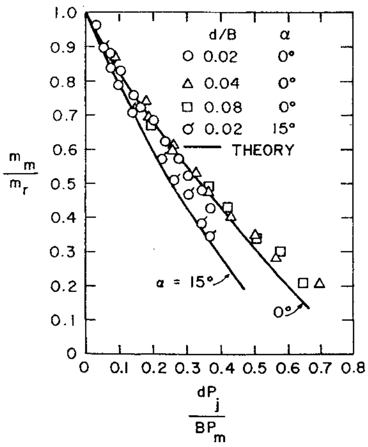

Fig. 2 Experimental results.

credible when applied to regions near the base of the jet where stagnation conditions are to be expected. By defining a pressure weighting function, it was found that the results of the analysis were insensitive to the assumption that mainstream static pressure acts at all points along the jet face. In retrospect, this result could be expected since, for large jet penetrations, the throttling effect reduces mainstream velocities near the point of injection to values that lead to small differences between static and stagnation pressures.

Application of mass continuity in the mainstream allows an exchange of the mainstream Mach number, M, for the height y in Eq. (3) (see Ref. 1 for details). The result is

$$C = \frac{B}{d}\frac{P_m}{P_j}\int_{M_{mo}}^{1} \pi(M)\frac{A(M)}{A(M_{mo})}\left(\frac{1}{M} - \frac{[(\gamma+1)/2]M}{1+[(\gamma-1)/2]M^2}\right)dM \tag{4}$$

where $\pi(M)$ and $A(M)$ are the familiar Mach number functions for pressure ratio (p_m/P_m) and area ratio (A/A^*) in locally isentropic flow. With specified values for $C(B/d)(P_m/P_j)$, Eq. (4) is easily solved for M_{mo}.

The throttling effect of the jet is given by

$$m_m/m_r = 1/A(M_{mo}) \quad \text{and} \quad m_j/m_r = (d/B)P_j/P_m \tag{5}$$

Data were obtained[1] for values of d/B in the range 0.02 to 0.08 and for P_j/P_m from 2 to 20. The experiments utilized air flows under conditions such that the jet was sonic at injection Results for $\alpha = 0°$ and $15°$ are shown in Fig. 2. The analysis accurately predicts the measured flow rates for conditions in which the mainstream is reduced to approximately 40% of its unthrottled value. For conditions of upstream injection, the agreement with experiment should be given only qualitative credence since in the experiments the actual angle of injection was difficult to determine. The deviation of theory from experiment at very large injection rates is thought to be due mainly to the neglect of viscous effects.

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Turbulent Interference Heating on Several Small Fin Configurations

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Nomenclature

Q = heating rate
M = Mach number
Re = Reynolds number

Subscripts
c = conditions on the cone at the fin station
e = boundary-layer edge at the fin station
x = surface distance on the cone
LE = conditions on the stagnation line of the fin leading edge

Introduction

When aerodynamic control fins are used on high velocity vehicles, regions of increased heating occur on the vehicle surface in the vicinity of the fin and also on the body of the fin itself. The flowfield in such regions is complex and does not yield itself to a straightforward analytical description. It is therefore necessary to investigate this type of problem experimentally. This Note presents turbulent heating distributions measured in the interference region on and around several small, highly-swept fin configurations mounted on a cone.

Test Program

The tests were conducted in the LTV Aerospace Corp.—Vought Aeronautics Div. High Speed Wind Tunnel over a range of freestream Mach numbers from 2.71 to 4.97 and freestream Reynolds numbers from 1.1 to 2.8 × 10⁷/ft. The 3-in. long fins were mounted on the aft end of a 6° half-angle, 10-in. base diameter cone. Two noses were used, one sharp and one 40% blunt, to vary local flow conditions on the fins. Boundary-layer thicknesses were about one-third the fin height and no evidence of separation was observed. Heat-transfer data were obtained on the fins and in the interference region on the cone using the transient thin skin technique. A complete description of the test program and results is given in Refs. 1 and 2. The results for the fin leading edge stagnation line were analyzed and discussed in Ref. 3.

Results

The experimental heat-transfer distributions around the leading edge and along the side of several 60° swept, cylindrical leading edge fins are shown in Fig. 1. The data represent a range of local Mach numbers, M_e, from 2.13 to 4.71 and local Reynolds numbers, Re_x, from 1.4 to 11.0 × 10^7. The heating rate decreases around the leading edge to about 40%–60% of the stagnation line value at the tangency point and decreases further to about 20%–45% along the fin side. A comparison of these data with the prediction obtained using the expression of Beckwith and Gallagher[4] for the turbulent circumferential heating for a 60° swept cylinder in a freestream with a Mach number range from 4.71 to 2.13 is given in Fig. 1. The comparison indicates that the data on the fin face forward of the tangency point generally lie above the prediction range.

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