Engineering Notes

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Application of Compressibility Correction to Calculation of Flow in Inlets

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HIGH local surface Mach numbers are encountered on the internal surfaces of inlets for V/STOL and CTOL aircraft. Unfortunately, accurate general compressible solutions are not readily available for inlets. However, incompressible potential flow solutions are generally available. Hence a simple compressibility correction suitable for inlets is needed to correct the incompressible potential flow solution to obtain accurate calculation procedures for inlet design. A compressibility correction for internal flow solutions was proposed in Ref. 1, as

$$V_c = V_i(\rho_i/\bar{\rho}_c)^{V_i/\bar{V}_i} \tag{1}$$

where V_c = the local compressible velocity, V_i = the local incompressible velocity, $\bar{V}_i$ = the average incompressible velocity across the flow passage, ρ_i = incompressible density, $\bar{\rho}_c$ = average compressible density across flow passage. This expression was developed by comparing exact compressible and incompressible solutions for flow through a turbine blade passage. However, the correction was believed to be applicable to other geometric configurations such as engine nacelle inlets. This paper presents an application of the compressibility correction of Ref. 1 to the calculation of flow in axisymmetric inlets and a comparison of the results with experimental data from wind-tunnel model tests.

The inlet geometry studied is illustrated in Fig. 1. This configuration is a conventional subsonic inlet with a NACA series one external cowl shape and a two-to-one ellipse internal lip. The contraction ratio (highlight area/throat area) is 1.30 and the diameter at the diffuser exit is 13.97 cm.

A comparison of the theoretical internal surface static pressure distribution with experimental data is shown in Fig. 2 for static conditions. The pressure distributions are presented from the inlet highlight ($X/L = 0$) to the diffuser exit ($X/L = 1.0$). Three theoretical curves are shown: 1) the incompressible potential flow solution (calculated by the method of Ref. 2); 2) the incompressible solution corrected for compressibility Ref. 1); and 3) the corrected solution with boundary-layer displacement thickness taken into account. The incompressible potential flow solution overestimated the static pressures because of the high surface Mach numbers (up to 0.9). The incompressible potential flow solution with the compressibility correction compared well with the experimental data in the first half of the inlet. In the aft portion of the diffuser the theoretical static pressures were on the average 2% higher than experimental data. However, when both the compressibility correction and boundary-layer displacement thickness (calculated by method of Ref. 3) were included, the theoretical and experimental static pressures were in good agreement over the entire length of the diffuser.

Further validity of the compressibility correction is illustrated in Fig. 3 where experimental data and theory are compared for angles of attack of 20° and 40°. [Because of the high lift coefficients and low speeds necessary for takeoff and landing operations of STOL aircraft, the engine inlet will be exposed to larger upwash angles than conventional aircraft (Ref. 4)]. The pressure distributions on the windward side of the inlet are presented for a free-stream velocity of 32 m/sec and at mass flow rates of 90 and 100% of design. The theoretical surface static pressures generally compared well with the experimental data. Good agreement was obtained for local Mach numbers as high as 1.3 ($p/P_\infty = 0.36$). However, at the inlet highlight ($X/L = 0$) the theoretical static pressures were lower than

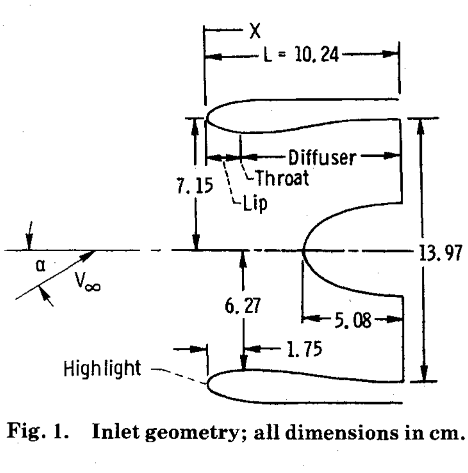

Fig. 1. Inlet geometry; all dimensions in cm.

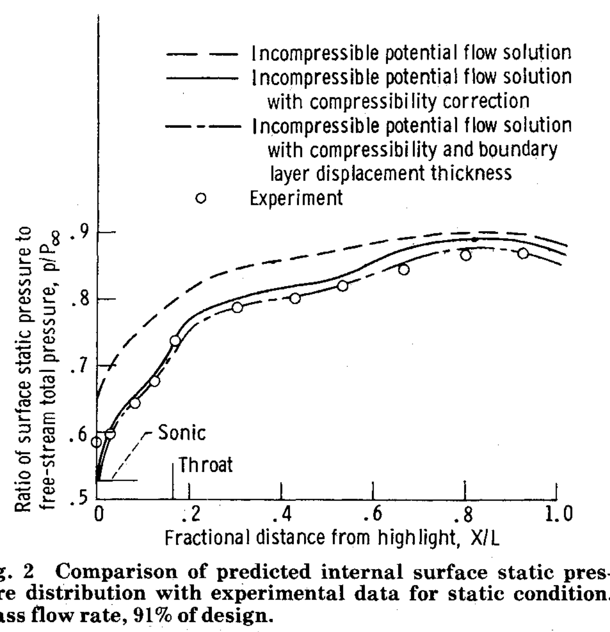

Fig. 2 Comparison of predicted internal surface static pressure distribution with experimental data for static condition. Mass flow rate, 91% of design.

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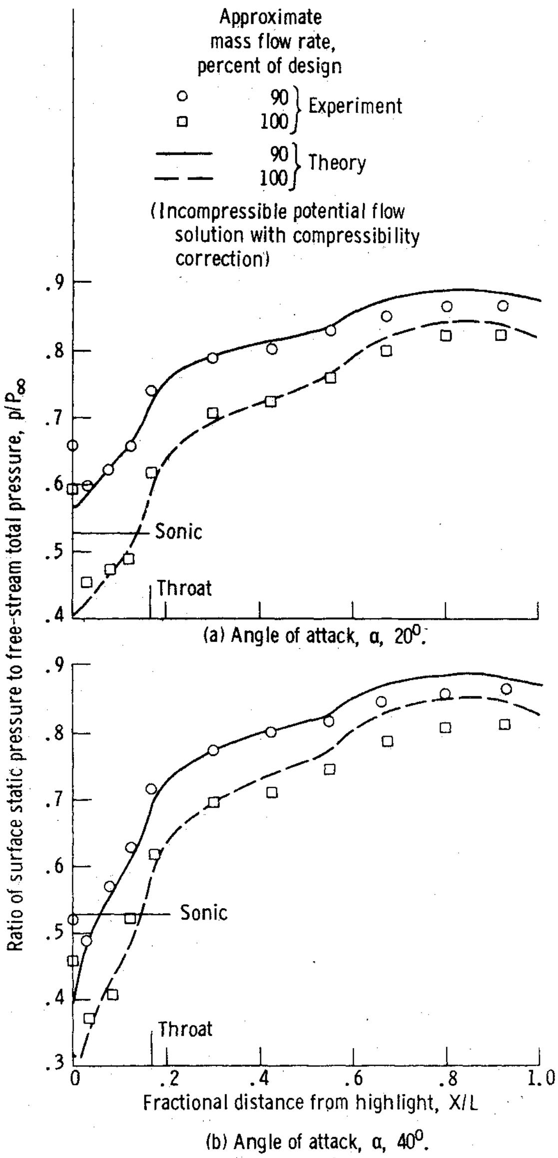

Fig. 3 Comparison of predicted internal surface static pressure distribution with experimental data for angle-of-attack conditions. Freestream velocity, V_∞, 32 m/sec.

experimental data. The possibility of local shocks and/or short separation bubbles and reattachment (often encountered on airfoils at angle of attack) near the inlet highlight could account for the experimental static pressures being higher than the theory. The discrepancy in the aft portion of the diffuser can be attributed to the neglect of the boundary-layer displacement thickness in solution which was discussed in the previous figure.

These preliminary results illustrate that the proposed compressibility correction of Ref. 1 gives a relatively good approximation to the internal compressible flow behavior and thus should be useful in the design and analysis of engine nacelle inlets. Further verification of the applicability of the compressibility correction to inlets is given in Ref. 5.

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An Automated Procedure for Determining the Flutter Velocity

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Introduction

EFFICIENT optimal design programs for aircraft structures which are subject to constraints on the flutter velocity require a rapid and automatic method for evaluating the flutter velocity. Lawrence and Jackson[1] described and compared three traditional methods for finding the flutter velocity of aeroelastic systems, the American approach, the British approach, and the Richardson approach. Repetitive solutions of a complex eigenvalue problem or a determinant for a large number of reduced frequencies or velocities are required for each flutter solution when traditional flutter solutions procedures such as the k-method or the p-k method[2] are employed. Bhatia[3] developed a method for solving the flutter equation and finding the match point by use of a Leguerre iteration technique. Desmarals and Bennet[4] recently developed an automated procedure for implementing the traditional V-g method of flutter solution, their method utilizes a cubic spline for interpolating generalized aerodynamic forces. In this paper a computationally efficient method for finding the flutter velocity is presented. The method utilizes derivatives of the eigenvalues with respect to the reduced frequency in a curve fitting scheme for finding the critical roots of the flutter equation. The method is unaffected by the coalescence of any of the eigenvalues.

Description of the Solution Method

The flutter equation may be written in the form

$$[K - \lambda_j(M + A)]U_j = 0 \tag{1}$$

where K, M and A are the stiffness, mass and aerodynamic force matrices respectively. λ_j is an eigenvalue and U_j is the corresponding eigenvector.

The aerodynamic force matrix is a function of the air density, Mach number, semichord, and the reduced frequency. The reduced frequency k may be defined by the relation

$$k = b\,\omega/V \tag{2}$$

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