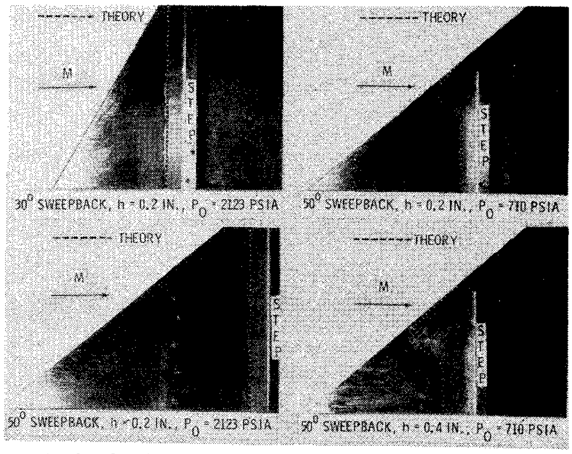

Fig. 3 Oil flow patterns ahead of steps on swept wings.

analysis is valid, then the actual separation line should resemble that indicated by the x's in Fig. 2. On the inboard portion of the wing, separation will occur along the turbulent line; on the outboard portion separation will occur along the laminar line. In between, the separated flow will be transitional, and the separation line will follow an "s" shaped curve such as that indicated in Fig. 2.

The existence of standing vortices, crossflow, and premature transition in a free shear layer make suspect the use of a two-dimensional strip-type analysis. However, these are secondary effects and do not invalidate the postulated method as a means for estimating qualitatively the effects of transition on the extent of separation ahead of controls on swept wings.

Experimental Program

Experiments were conducted in the High Reynolds Number Aerospace Research Lab, Mach 6 Tunnel. To ensure reasonable extents of both laminar and turbulent boundary-layer flows, it was opted to use "half wings" rather than full delta wings. Wings with machined sharp leading edges with sweepback angles of 0, 30, and 50° were fabricated. Steps of two different heights ($h = 0.2$ and 0.4 in.) were provided. The steps could be mounted either at the mid-chord or at the aft end of the wings. The steps are sealed to the wings, to prevent flow between the step and wing surface. The step heights are comparable to or larger than the undisturbed boundary-layer thickness along the inboard portions of the wings.

Oil flow and schlieren photographs were obtained for both step heights at both locations on the three wings for two tunnel flow stagnation pressure levels: approximately 710 and 2123 psia. The resulting freestream unit Reynolds numbers are approximately 9.4 and 27.2 million per foot (approximately 12 and 34 million based on the 50° wing root chord); the freestream Mach number is approximately 5.88 for both pressure levels.[11]

Sample oil flow photographs are shown in Fig. 3. At the higher pressure level, the flow over the wing was predominately turbulent and the oil accumulation lines indicate nearly a constant extent of separation ahead of the steps. For the lower pressure level, the oil accumulation lines are similar to the x line sketched in Fig. 2 for turbulent separation inboard, laminar separation outboard, and transitional separtion in between.

Comparison of Data with Theory

For turbulent separation, the measured oil accumulation lines indicated an average separation length of approximately $5.2h$. The theoretical value, indicated by the dashed lines in Fig. 3, is $4.9h$. We believe that the small discrepancy is caused by the subject step heights being comparable to the boundary-layer thicknesses on the plate surface.[6] For laminar-transitional-turbulent separation, the photographed oil accumulation lines closely resemble those predicted (dashed lines in Fig. 3) using the theoretical analysis previously described.

Conclusions

The postulated strip-type analysis correctly predicts, qualitatively, the shape of separated flow regions ahead of forward facing steps on swept wings. The proposed method is quatitatively correct, at least within the range of conditions for the subject experiments, as long as the location of boundary-layer transition on the swept wings is estimated correctly.

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Explicit Equations for Barometric Altitude Computations

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THE values of barometric altitude, Mach number, and various airspeeds are calculated by onboard air-data systems from measured pressures and temperatures. The equations that relate the measured values to the required quantities, although simple, involve noninteger exponents. But in supersonic range, the Mach number and airspeeds are related to measured values in an implicit fashion, thus requiring iterations for implementation in the onboard computer. To avoid such iterations, Bogel[1] has obtained explicit expressions for computation of supersonic Mach numbers and airspeeds. The expressions he obtained involve only square root and summations operations. Bogel also has im-

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Table 1 Computational errors in different ranges[a]

Altitude range (ft)	Maximum error (% of true value)	Maximum error (ft and true value)	Comments
−1000 to 32,500	±0.021	+7.1 at 32,500	Error at zero altitude (reference) is −0.0023 ft Error is less than ±0.01% of true value throughout, for altitudes less than 31,000 ft
32,500 to 45,000	±0.026	+8.52 at 32,500	Error is less than ±0.01% of true value between 34,000 ft and 45,000 ft
45,000 to 60,000	±0.01	−4.82 at 48,000	At all other points the error is much less than ±0.01% of true value

[a] Computations are made for U.S. Standard Atmosphere Tables.[2]

proved the existing explicit relations for Mach number and airspeeds in the subsonic range for implementation in the on-board computers.

With a view to bringing the barometric altitude computation into a form similar to those obtained by Bogel, explicit equations involving only square root and summation operations have been obtained here. Logarithmic form of altitude computation is preferred when mechanical analog mechanisms are used. However, with the requirements for implementation on digital computer, the logarithmic form does not offer any particular advantage.

Because the basic relationship between altitude and the corresponding atmospheric pressure is an exponential one, in general the altitude can be expressed as a power series. The power series can be either in terms of atmospheric pressure p itself, or in terms of the ratio of atmospheric pressure p to the reference standard mean sea-level pressure p_0. The latter case is required in the lower altitude ranges for landing purposes. Thus, it is possible to find an equation of the form

$$h = a + [b + cy + d(y)^{1/2}]^{1/2} \tag{1}$$

where h is the altitude to be computed, a, b, c, and d are constants, and y is either pressure or pressure ratio. For the purposes of computation, the altitude range of −1000 ft to +60,000 ft is divided into three overlapping ranges. Four suitable points were chosen in each range and the simultaneous equations were solved to obtain the values of a, b, c, and d in each range. The equations so obtained are

$$h_1 = 10.5323 - [-11.1767 - 5.5179y + 127.6249(y)^{1/2}]^{1/2} \times 10^4 \text{ ft} \tag{2}$$

for $-1000 \text{ ft} \le h_1 \le 32{,}500$ ft

$$h_2 = 7.4599 - [-13.6337 + 0.5633p + 9.5546(p)^{1/2}]^{1/2} \times 10^4 \text{ft} \tag{3}$$

for $32{,}500 \text{ ft} \le h_2 \le 45{,}000$ ft, and

$$h_3 = 12.3836 - [-20.4965 - 3.941p + 47.8316(p)^{1/2}]^{1/2} \times 10^4 \text{ ft} \tag{4}$$

for $45{,}000 \text{ ft} \le h_3 \le 60{,}000$ ft where $y = p/p_0$.

The errors in values of altitudes computed with these equations are shown in Table 1. The accuracy requirements for pressure altitude, as specified in ARINC Characteristic No. 545, range from ±2.5% to ±0.25% for different altitudes up to 60,000 ft. From Table 1, it is evident that the Eqs. (2-4) give required accuracy for computational purposes. The particular division of ranges i.e., −1000 ft to +32,500 ft, 32,500 ft to 45,000 ft, and 45,000 ft to 60,000 ft is done to keep the computational errors within the required limits. However, ranges can be extended with loss in accuracy. For example

$$h_4 = 10.5271 - [-11.1656 - 5.5124y + 127.4973(y)^{1/2}]^{1/2} \times 10^4 \text{ ft} \tag{5}$$

in place of Eq. (1), would cover a range from −1000 ft to +39,000 ft with an error less than ±0.05% of true value.

It can be observed that with these equations no particular distinction is made in computations at tropopause transition level. The equations are simply fitted to U.S. Standard Atmosphere values and the measured values of pressure require usual corrections. It is not claimed that the implementation of these equations would offer many advantages especially with advanced digital computation techniques now available, but these equations go well with the equations for Mach number and airspeeds in the forms proposed by Bogel.

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Aerodynamic Heat Transfer to a Hypersonic Research Aircraft Model (X-24C) at Mach 6

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Introduction

A joint USAF/NASA study has developed a conceptual design for a new high-speed research airplane (X-24C).[1] The vehicle, a rocket-boosted, delta planform aircraft, is launched from a B-52 and is capable of 40 sec of rocket cruise

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