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SURFACE PRESSURE DISTRIBUTION AT HYPERSONIC SPEEDS FOR BLUNT DELTA WINGS AT ANGLE OF ATTACK

By Marcus O. Creager

Ames Research Center
Moffett Field, Calif.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON
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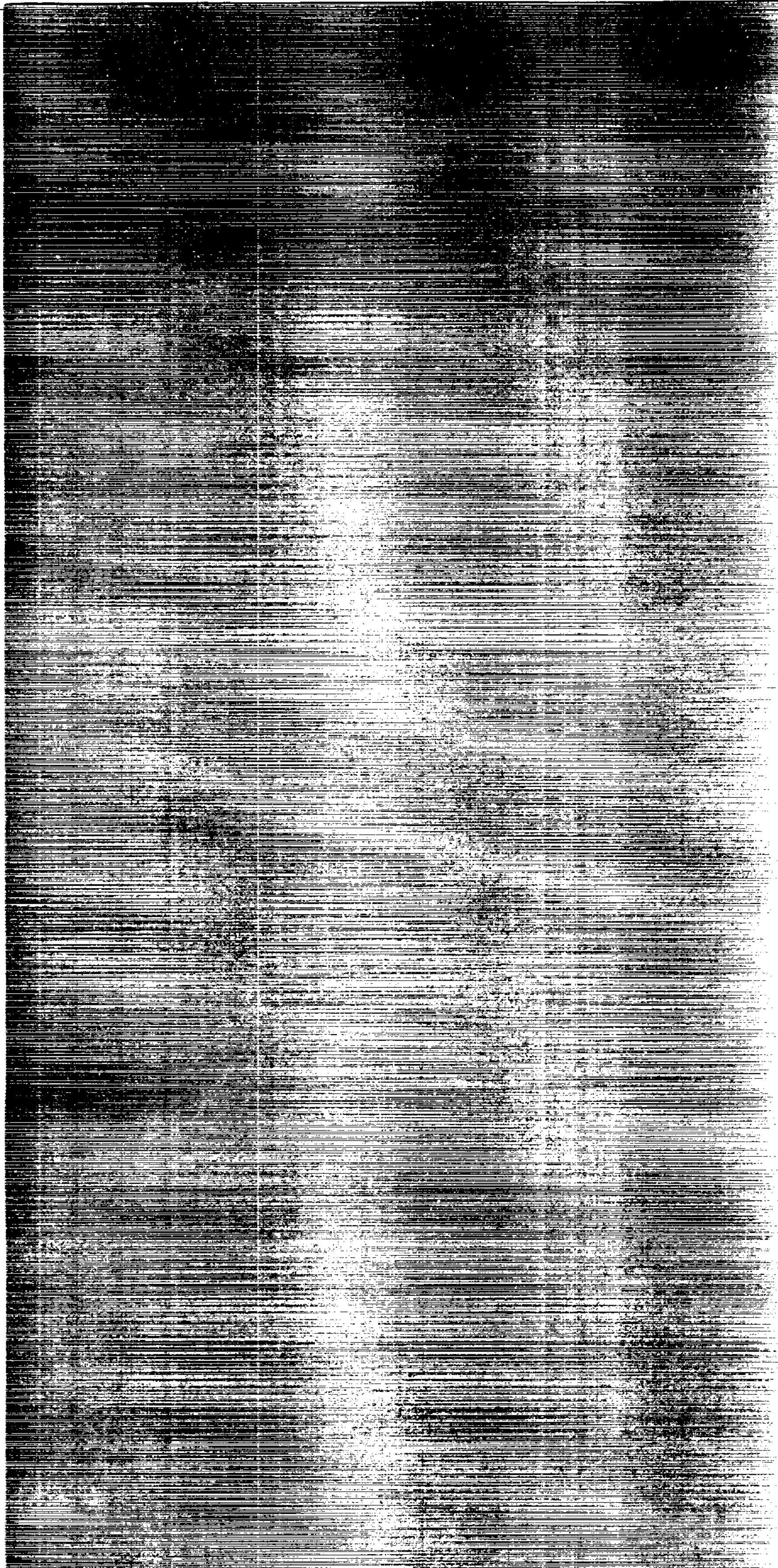

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SURFACE PRESSURE DISTRIBUTION AT HYPERSONIC SPEEDS FOR BLUNT DELTA WINGS AT ANGLE OF ATTACK

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SUMMARY

Surface pressures were measured over a blunt 60° delta wing with extended trailing edge at a Mach number of 5.7, a free-stream Reynolds number of 20,000 per inch, and angles of attack from -10° to $+10^\circ$. Aft of four leading-edge thicknesses the pressure distributions evidenced no appreciable three-dimensional effects and were predicted qualitatively by a method described herein for calculation of pressure distribution in two-dimensional flow.

Results of tests performed elsewhere on blunt triangular wings were found to substantiate the near two-dimensionality of the flow and were used to extend the range of applicability of the method of surface pressure predictions to Mach numbers of 11.5 in air and 13.3 in helium.

INTRODUCTION

A simple type of hypersonic glider is a blunted slab wing of triangular plan form. The aerodynamic and heat-transfer characteristics of such a glider depend on the surface pressures. Methods have been proposed (refs. 1, 2, and 3) which utilize a combination of boundary-layer and blast-wave theory to predict the pressures on a two-dimensional infinite wing section with a fair degree of accuracy. The conditions under which these methods may be applicable to delta wings have not been established.

The purposes of the research described in this paper were threefold: (1) to study in the Ames low density wind tunnel the surface pressures on a triangular-plan-form wing with extended trailing edge whose span is larger than the test stream; (2) to compare these pressures to the pressures calculated for a simple swept infinite wing; and (3) to compare the results to those reported in references 4, 5, and 6 for simple triangular-plan-form blunt wings.

SYMBOLS

C_D	pressure drag coefficient
C_w	constant in linear viscosity relation, $\frac{\mu_w}{\mu} = C_w \frac{T_w}{T}$
C_γ	constant equal to 0.112 for air and 0.169 for helium
d	leading-edge diameter or thickness of plate
M	Mach number
p	pressure
$Re_{\infty d}$	Reynolds number, $\frac{\rho_\infty U_\infty d}{\mu_\infty}$
$Re_{\alpha s}$	Reynolds number, $\frac{\rho_\alpha U_\alpha s}{\mu_\alpha}$
s	streamwise distance from leading edge or nose (see fig. 1(b))
T	temperature
T_t	total temperature
T_w	wall or plate temperature
U	velocity
x	distance normal from leading edge (see sketch (a))
y	spanwise distance from body center line (see fig. 1(b))
z	distance along leading edge (see sketch (a))
α	angle of attack, deg
γ	ratio of specific heats
ρ	density
Λ	leading-edge sweep angle, deg
μ	viscosity

Subscripts

α property calculated from sharp-wedge oblique shock wave or Prandtl-Meyer expansion theory

∞ property calculated from free-stream conditions

EQUIPMENT AND TEST METHOD

Wind Tunnel

The tests were conducted in the Ames 8-inch low density wind tunnel which is an open-jet nonreturn type and is described in references 1 and 2. The axisymmetric nozzle produces a usable stream about 2 inches in diameter at a Mach number of 5.7 and a free-stream pressure of 250 microns of mercury absolute.

Model

The surface pressure distribution model was made from brass into a 1/4-inch-thick slab triangular in plan form with an extended trailing edge (see fig. 1(a)). The leading edge was semicylindrical, and the nose was a spherical sector (see fig. 1(b)). The width of the model was 4 inches, which was sufficient to span the usable stream. The triangular plan-form portion had a 60° included angle and a 3-1/2-inch maximum chord. The extended trailing edge was 7-1/2 inches in length.

The surface pressure orifices were drilled into the upper surface, hereafter called the test surface. These orifices were located along chord lines as shown in figure 1(b) and were connected to a multiple-tube manometer for pressure indication.

Test Method

The model was placed in the stream with the nose on the stream center line at a distance of 3-1/2 inches from the nozzle exit plane. Both the orifice chord line position in the stream and angle of attack of the test surface were varied. The angle of attack was varied from $+10^{\circ}$ (compression) to -10° (expansion). For these angles of attack, pressures were measured along orifice chord lines A_1 and C_1 (see fig. 1(b)). For 0° angle of attack the pressure measurements were obtained for the four orifice chord lines, A_1, B_1, C_1, and D_1.

In addition, at 0° angle of attack the model was shifted so that each orifice chord line in turn was in the stream center and the chord line stagnation point was 3-1/2 inches downstream from the nozzle exit plane. This set of tests was performed so that possible jet nonuniformities and boundary effects might be evaluated.

RESULTS AND DISCUSSION

The surface pressure distributions obtained in the present tests will be scrutinized for possible three-dimensional effects and compared to the values calculated by a method representative of the blunt infinite aspect ratio wing. This comparison will then be extended to other test results for triangular wings (refs. 4, 5, and 6).

Present Test Results

In figure 2 and subsequent figures the measured surface pressures in ratio to free-stream static pressure are plotted versus the ratio of streamwise distance from the leading edge to the leading-edge diameter. The surface pressures measured for $+10^{\circ}$, 0°, and -10° angles of attack are shown in figure 2 for various span locations of the orifice chord line on the body. Near the leading edge, the pressures along the center-line chord aft of the blunt apex are noted to be higher in value than those at corresponding locations along the outboard chord lines. There exists insufficient evidence to assess this effect. However, the chord-wise variation of pressures does not change appreciably with span location aft of an s/d of about 4 for $\alpha = 0^{\circ}$ and 10°, and aft of an s/d of 6 for $\alpha = -10^{\circ}$. This result indicates that the flow is of quasi-two-dimensional character insofar as the surface pressures are concerned and that a two-dimensional method of calculation might be used for predicting these pressures.

In reference 1 a method of calculation of surface pressures was developed for two-dimensional flow over a swept infinite wing at angle of attack. The equation proposed in reference 1 for this case is reproduced below.

$$\frac{p}{p_{\infty}} = \frac{p_{\alpha}}{p_{\infty}} + \frac{p_{\alpha}}{p_{\infty}} Ab_{\alpha}\bar{x}_{\alpha} + BcI \cos^{2}\Lambda \qquad (1)$$

where

$$\bar{x}_\alpha = \frac{{M_\alpha}^3 \sqrt{C_W}}{\sqrt{Re_{\alpha s}}}$$

$$I = \frac{(M_\infty)^2}{\left(\frac{s}{d}\right)^{2/3}}$$

$$b_\alpha = \gamma \left[\frac{0.865}{{M_\alpha}^2}\left(\frac{T_W}{T_\alpha}\right) + 0.166(\gamma-1)\right]$$

$$c = C_\gamma (C_D)^{2/3}$$

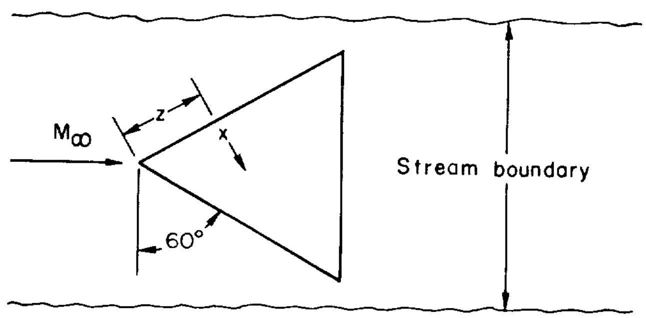
Equation (1) was shown in reference 1 to predict surface pressures within an average error ±8 percent over a wide range of test conditions when the empirical constants are taken as $A = 1/\sqrt{C_W}$ and $B = (1/2)^{2/3}$. The value of C_γ used was 0.112 for air and 0.169 for helium.

Equation (1) was used to calculate the solid curves in figure 2. For $\alpha = -10^\circ$, the data are in poor agreement with the solid line. However, for $\alpha = 0^\circ$ and 10° the measured pressures are within about 10 percent of the solid line back to an s/d of about 10 or 12 and depart markedly from the theory for larger distances aft of the leading edge. This drop-off has been noted in reference 1 for similar tests in this stream. From meager information available it seems probable that this effect is caused by the interactions of the bow shock wave with the stream boundary. From figure 2 it is also noted that the values measured for $+10^\circ$ (compression) angle of attack are lower than the theory, and the values measured for -10° (expansion) angle of attack are higher than the theory.

In an attempt to assess possible effects of stream variation in the data of figure 2, pressures were measured for $\alpha = 0^\circ$ as explained in the section on test methods (i.e., the stagnation point of each orifice line chord was placed in turn in the same stream position). These data are shown in figure 3 and may be noted to be similar to the values of pressure for $\alpha = 0^\circ$ shown in figure 2. Hence, no large stream effects could be identified.

Application to Other Test Results

Triangular wing.- Surface pressures measured on a delta plan-form wing with cylindrical leading edge at a Mach number of 13.3 in helium were reported by Bogdonoff and Vas (refs. 4 and 5). The data were obtained as explained in reference 4 from a test body as pictured in plan view in the sketch.



Some of their data are shown here in figure 4(a) for various angles of attack. The pressures were obtained along chord lines normal to the leading edge (see sketch) and therefore represent a wide variation in span location. By this method of presentation, if the data are plotted versus s/d (i.e., in free-stream direction), it is again noted that the chordwise variation of pressures does not change significantly with span location. The data are noted to be comparable to the values calculated from equation (1) for 0° and +10° angle of attack. However, the comparison between theory and experiment is poor for $\alpha = -4.25^\circ$.

Surface pressures were reported in reference 5 for a test body similar to that in the sketch but with 75° leading-edge sweep (i.e., 30° included angle). The data for 0° and 10° angle of attack are shown here in figure 4(b). The distribution of pressures along a streamwise chord line is noted to be nearly independent of span location of the chord line. The values calculated from equation (1) compare about as well with these data for $\Lambda = 75^\circ$ (fig. 4(b)) as with those for $\Lambda = 60^\circ$ (fig. 4(a)).

The correlations obtained in references 1 and 2 were for plate temperatures near recovery temperatures and low stream stagnation temperatures. The applicability of equation (1) for high temperature conditions must be investigated. Results of surface pressure measurements on an unswept blunt plate and a 60° delta wing in air at a Mach number 11.5 have been reported by Lee, reference 6. The stagnation temperature of the test stream was about 2400° R. The plate temperature, however, was not reported. Values of the pressure calculated by equation (1) for the unswept plate are shown in figure 5 as solid lines for an assumed ratio of $T_W/T_t = 0.7$ ($T_W = 1600^\circ$ R) and for $T_W/T_t = 0.25$ ($T_W = 600^\circ$ R). The data obtained at 0° angle of attack (fig. 5) are noted to be bracketed by these two calculations. In lieu of an actual measured value of T_W, further calculations for the data of reference 5 have been made for these two conditions, $T_W/T_t = 0.25$ and 0.7. Note that the predictions of equation (1) indicate a somewhat larger change in pressures with α than is displayed by the data.

The results of measurements of surface pressure (ref. 6) over the 60° delta wing, similar to that shown in the sketch, for various angles of attack are shown in figure 6. Also, the values calculated from equation (1) for $T_W/T_t = 0.25$ and 0.7 are plotted in figure 6. These data are also seen to exhibit nearly two-dimensional characteristics. The agreement between these sets of data and the calculated curves is as good as it is in the case of the present data and those of references 4 and 5.

CONCLUSION

Surface pressures were measured on a blunted 60° delta plan-form wing with an extended trailing edge at a Mach number of 5.7, free-stream Reynolds number of 20,000 per inch and at various angles of attack. For these test conditions three-dimensional effects were noted in the pressures near the leading edge. However, these effects were not significant aft of about four leading-edge thicknesses, and the surface pressures were predicted to within ±15 percent by the methods developed for two-dimensional flow.

Results of tests performed elsewhere on similar delta plan-form bodies essentially substantiate and extend the range of applicability of these conclusions to Mach numbers of 11.5 in air and 13.3 in helium.

Ames Research Center
National Aeronautics and Space Administration
Moffett Field, Calif., Feb. 12, 1959

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2. Creager, Marcus O.: Effects of Leading-Edge Blunting on the Local Heat Transfer and Pressure Distribution Over Flat Plate in Supersonic Flow. NACA TN 4142, 1957.

3. Bertram, Michel H., and Henderson, Arthur, Jr.: Effects of Boundary-Layer Displacement and Leading-Edge Bluntness on Pressure Distribution, Skin Friction, and Heat Transfer of Bodies at Hypersonic Speeds. NACA TN 4301, 1957.

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5. Bogdonoff, S. M., and Vas, I. E.: An Exploratory Study of a Delta Wing at Hypersonic Speeds. Part III: Pressure Distribution Studies on a 30° Included Angle Delta Wing at Angles of Attack. Bell Aircraft Rep. D143-978-009. Mar. 26, 1957.

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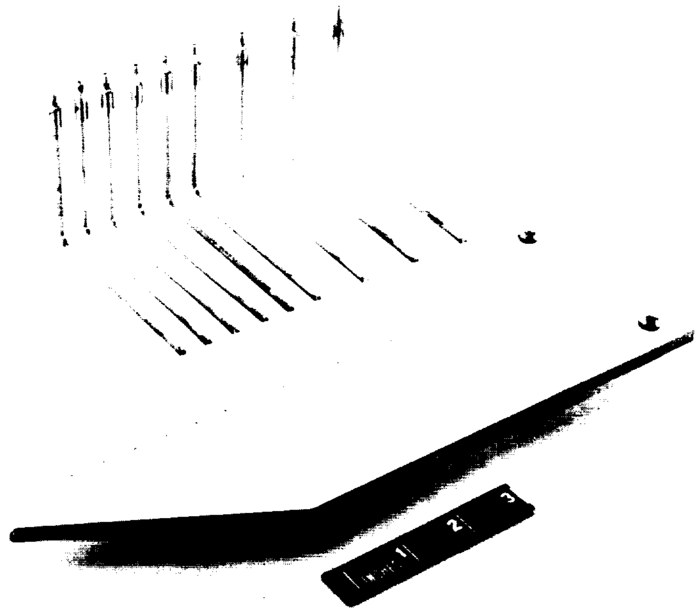

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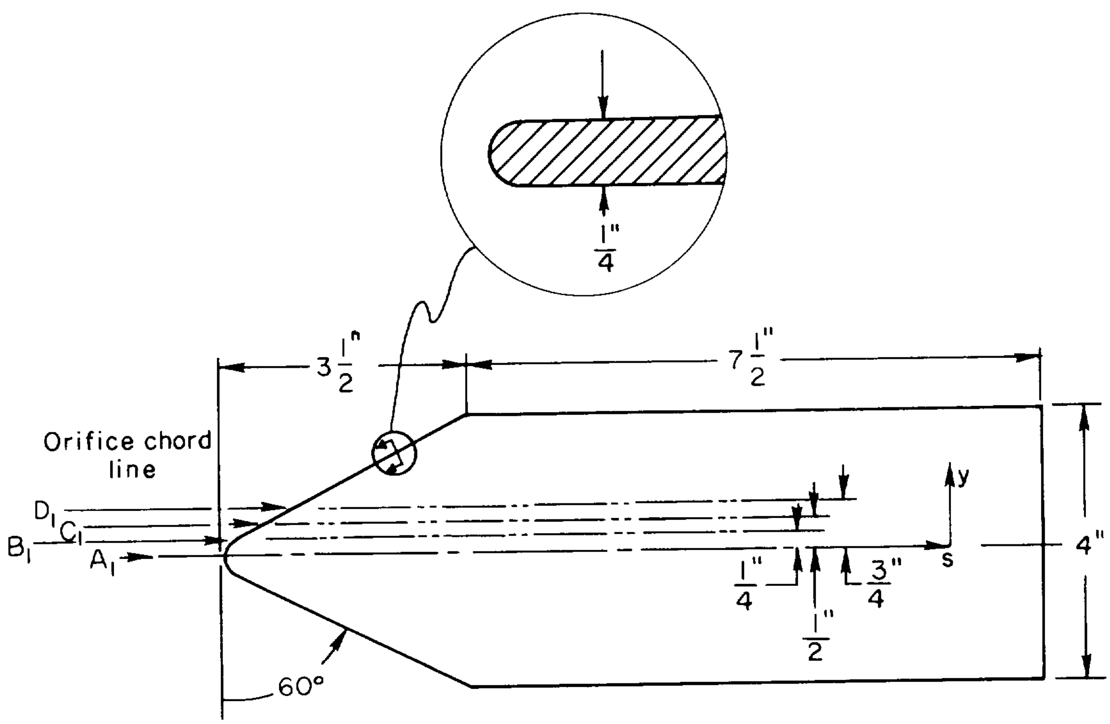
(a) Photograph of test body.



(b) Plan form of test body.

Figure 1.- Test body.

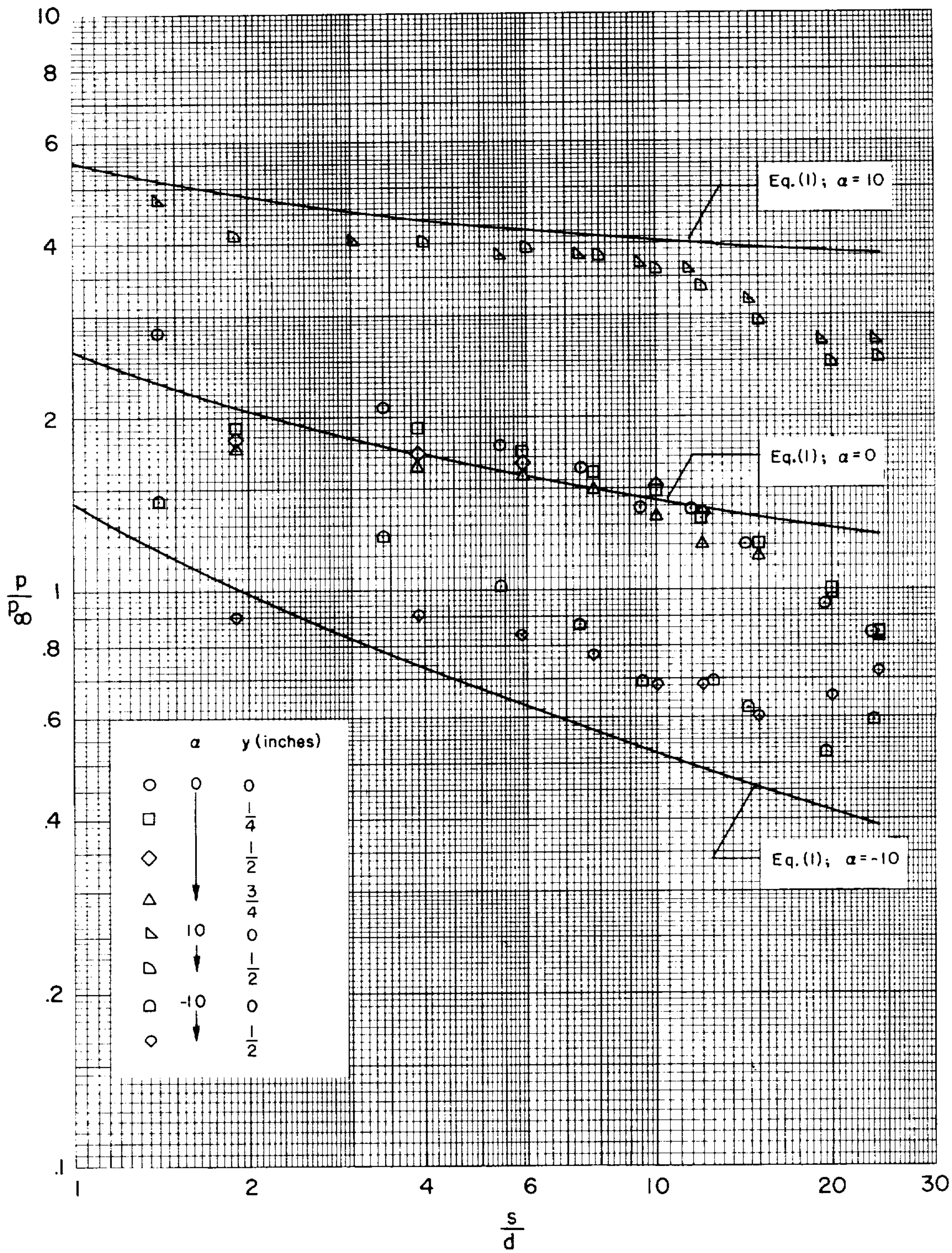


Figure 2.- Variation of surface pressures with streamwise distance from the blunt leading edge of a 60° triangular wing in air; $M_\infty = 5.7$, $Re_{\infty d} = 4860$.

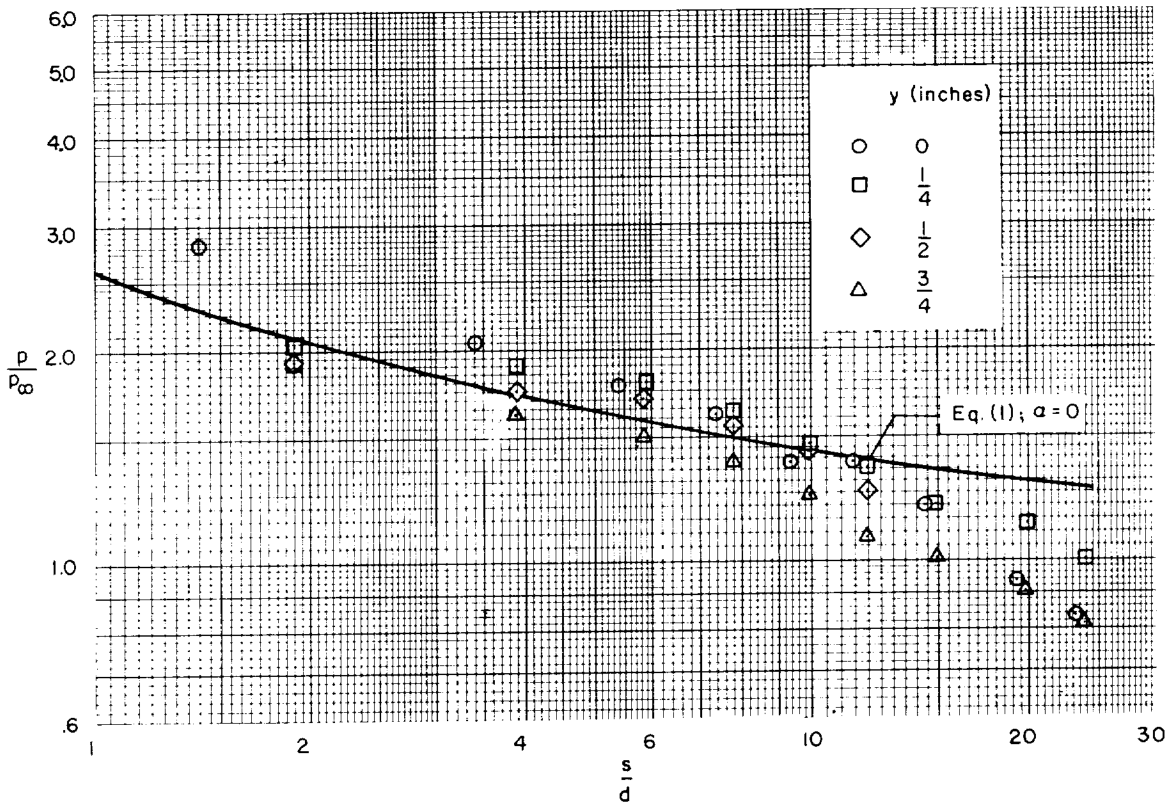
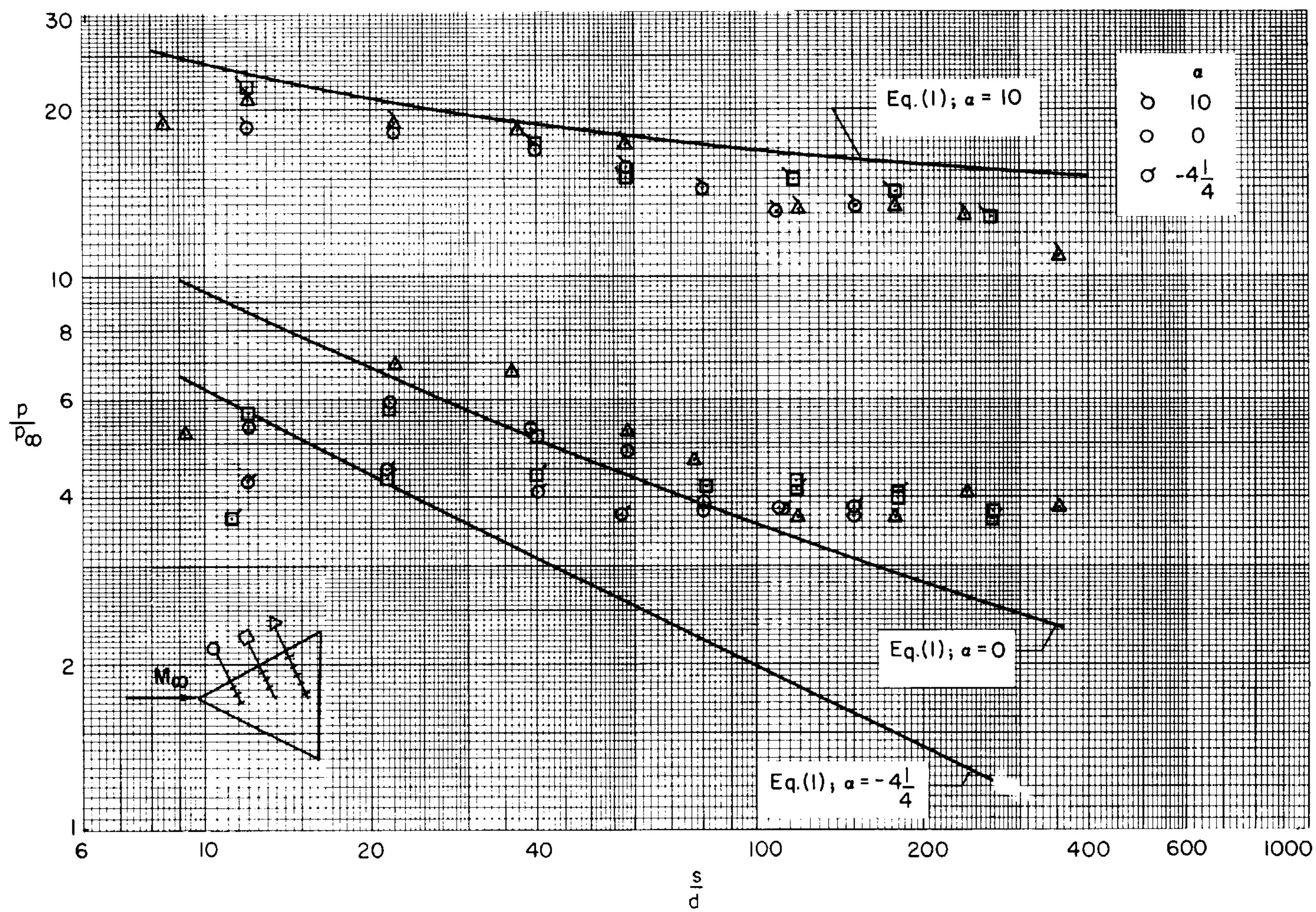
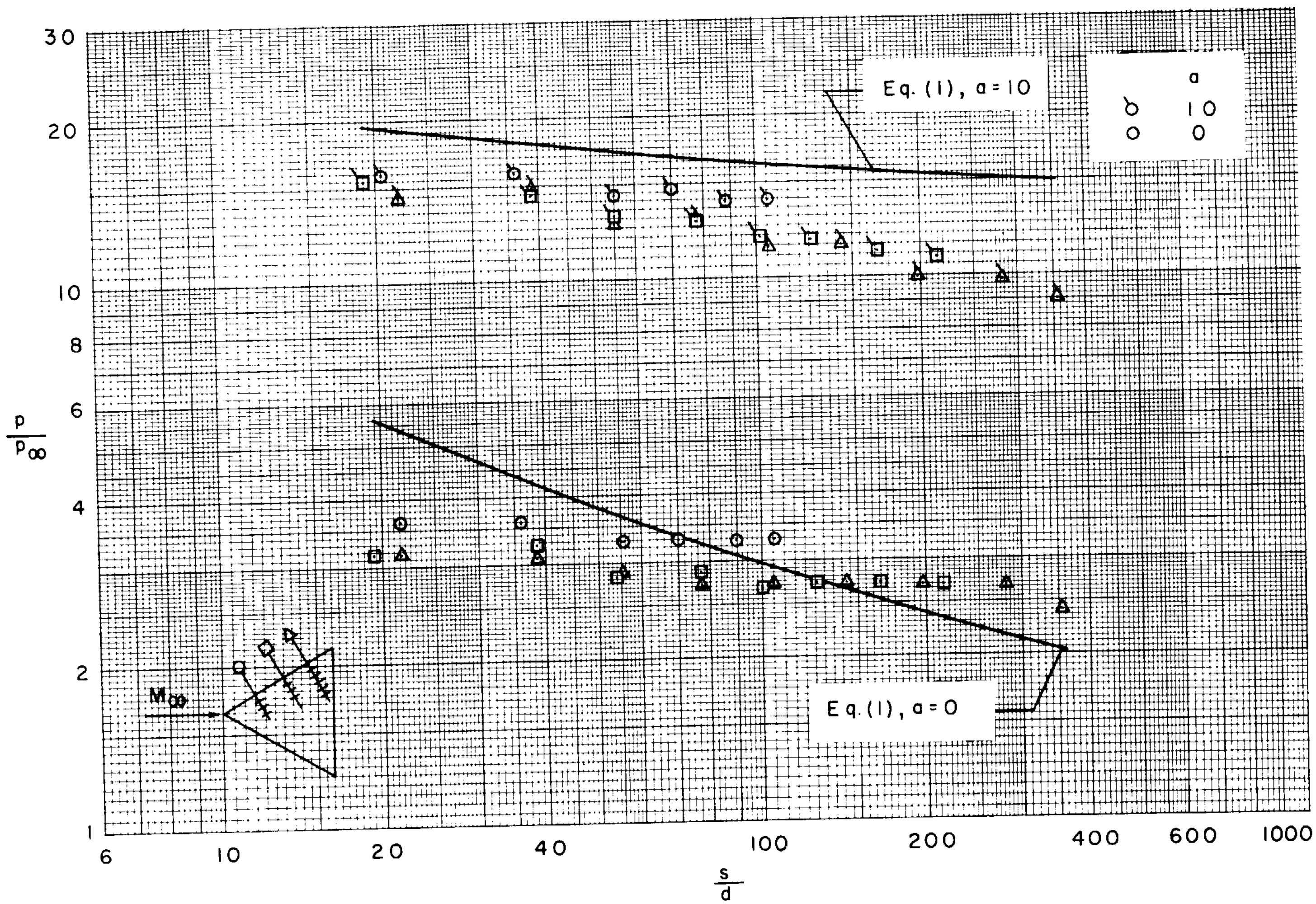


Figure 3.- Variation of surface pressures with streamwise distance from the blunt leading edge of a 60° included angle triangular wing for constant location of the orifice chord line in the test section; $M_\infty = 5.7$.



(a) $\Lambda = 60^\circ$ (ref. 4)

Figure 4.- Variation of surface pressures with streamwise distance from the blunt leading edge of triangular wings in helium; $M_\infty = 13.3$, $Re_{\infty d} = 4150$.



(b) $\Lambda = 75^{\circ}$ (ref. 5)

Figure 4.- Concluded.

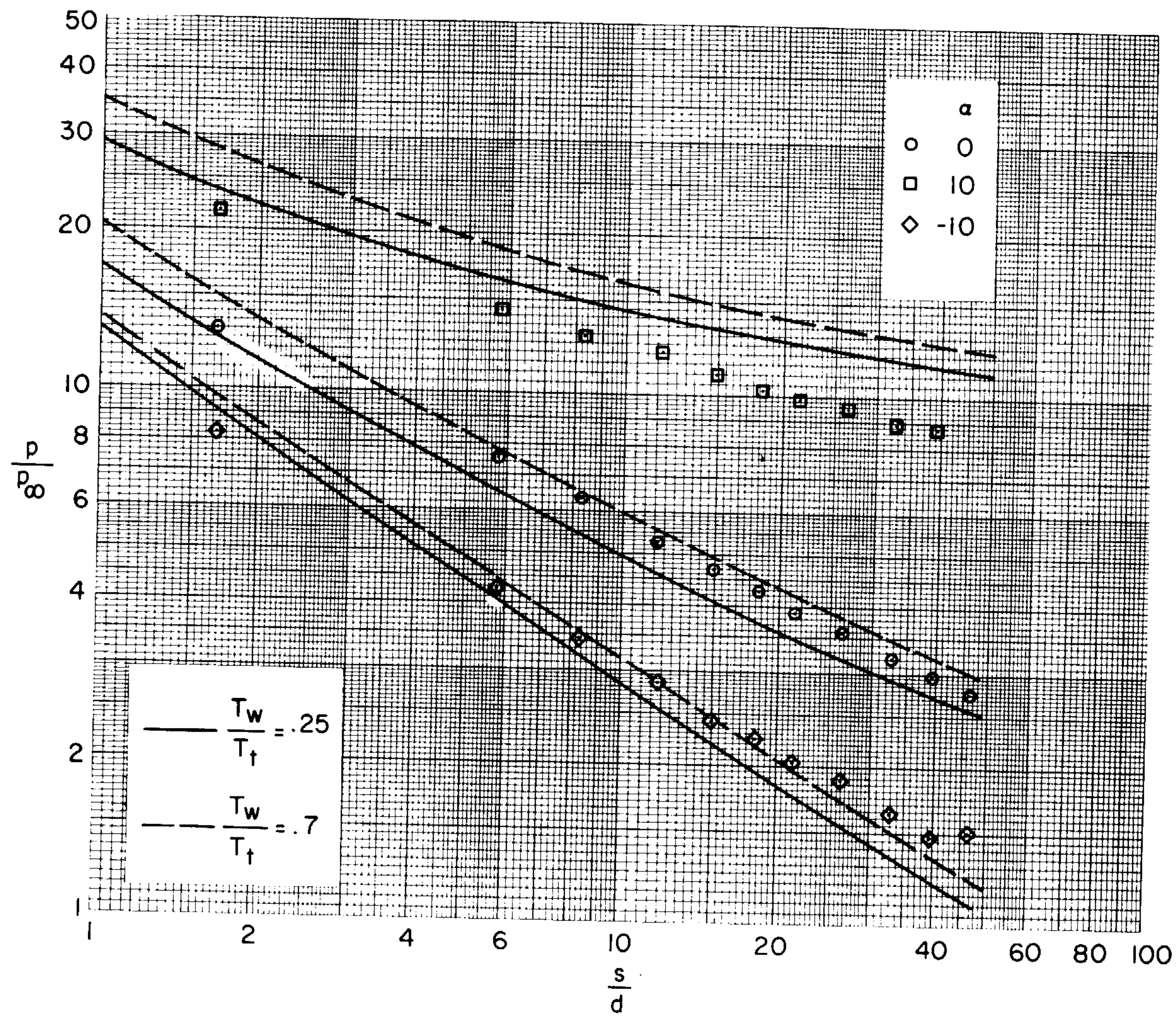


Figure 5.- Comparison of surface pressures measured over an unswept blunt flat plate in air at M_∞ = 11.5 and $Re_{\infty d}$ = 2838 (ref. 6) with the values calculated by equation (1) for various plate temperatures.

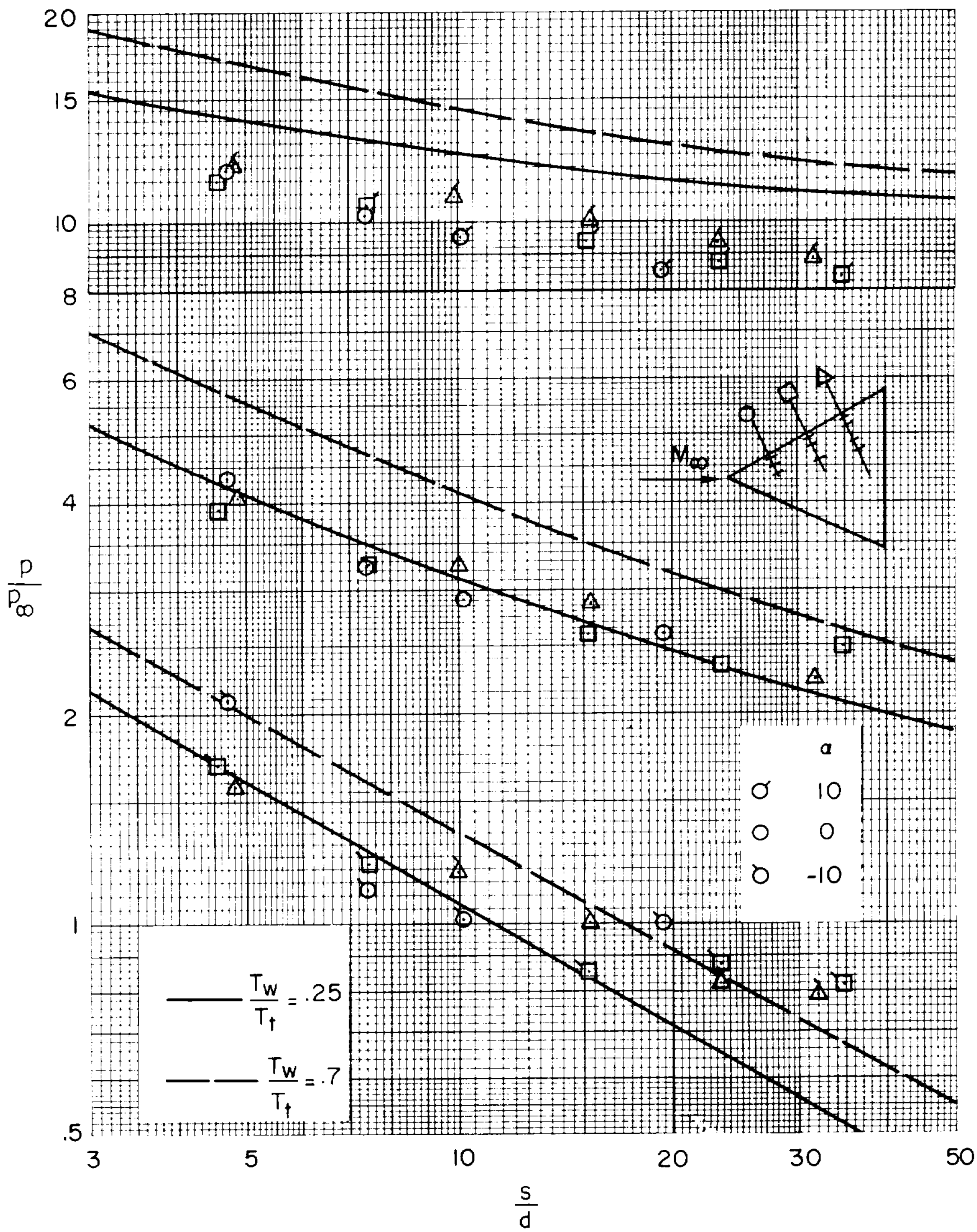


Figure 6.- Variation of surface pressures with streamwise distance from the blunt leading edge of a 60° included angle triangular wing in air; $M_\infty = 11.5$, $Re_{\infty d} = 2838$ (ref. 6). Curves calculated by equation (1).

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Surface pressures were measured on a blunt 60° delta wing at angles of attack from -10° to $+10^{\circ}$ for a Mach number of 5.7 and a Reynolds number per inch of 20,000. These surface pressures and pressures reported elsewhere for similar test bodies at Mach numbers of 11.5 in air and 13.3 in helium are compared with those predicted by a method developed previously for two-dimensional flow.

Copies obtainable from NASA, Washington

1. Flow, Supersonic (1.1.2.3)
2. Flow, Laminar (1.1.3.1)
3. Flow of Rarefied Gases (1.1.5)
4. Wings, Complete - Sweep (1.2.2.2.3)
5. Surface Conditions - Complete Wings (1.2.2.2.6)
6. Loads, Aerodynamic (4.1.1)

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