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RESEARCH MEMORANDUM

for the

U. S. Air Force

AERODYNAMIC CHARACTERISTICS OF A 0.04956-SCALE MODEL OF THE CONVAIR TF-102A AIRPLANE AT TRANSONIC SPEEDS

COORD. NO. AF-120

By Robert S. Osborne

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Langley Field, Va.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

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SUMMARY

The basic aerodynamic characteristics of a 0.04956-scale model of the Convair TF-102A airplane with controls undeflected have been determined at Mach numbers from 0.60 to 1.135 for angles of attack up to approximately 22° in the Langley 8-foot transonic tunnel. In addition, comparisons have been made with data obtained from a previous investigation of a 0.04956-scale model of the Convair F-102A airplane.

The results indicated the TF-102A airplane was longitudinally stable for all conditions tested. An increase in lift-curve slope from 0.045 to 0.059 and an 11-percent rearward shift in aerodynamic-center location occurred with increases in Mach number from 0.60 to approximately 1.05. The zero-lift drag coefficient for the TF-102A airplane increased 145 percent between the Mach numbers of 0.85 and 1.075; the maximum lift-drag ratio decreased from 9.5 at a Mach number of 0.60 to 5.0 at Mach numbers above 1.025. There was little difference in the lift and pitching-moment characteristics and drag due to lift between the TF-102A and F-102A configurations. However, as compared with the F-102A airplane, the zero-lift drag-rise Mach number for the TF-102A was reduced by at least 0.06, the zero-lift peak wave drag was increased 50 percent, and the maximum lift-drag ratio was reduced as much as 20 percent.

INTRODUCTION

At the request of the U. S. Air Force, an investigation of the aerodynamic characteristics of a 0.04956-scale model of the Convair TF-102A airplane has been conducted at transonic speeds in the Langley 8-foot transonic tunnel.

The TF-102A airplane is a trainer version of the Convair F-102A high-altitude supersonic interceptor configuration. It employs the same delta wing and vertical tail as the F-102A airplane and the same fuselage rearward of the canopy. In order to allow side-by-side seating of two crew members, the canopy and forward portion of the trainer fuselage have been enlarged and the side air inlets lowered as compared with the single-seat interceptor.

Tests of 0.04956-scale models of the F-102A airplane have been conducted previously in the Langley 8-foot transonic tunnel and the results reported in references 1 to 3. In order to evaluate the effects on static longitudinal stability and drag of the previously described major modifications required to convert the F-102A airplane to a training configuration, force tests of a 0.04956-scale model of the TF-102A airplane were performed with controls undeflected in the Langley 8-foot transonic tunnel at Mach numbers from 0.60 to 1.135 for angles of attack up to approximately 22°. The results are presented herein.

SYMBOLS

b	wing span, in.
$\bar{c}$	wing mean aerodynamic chord, in.
C_D	drag coefficient, D/qS
C_L	lift coefficient, L/qS
$C_{L,(L/D)_{max}}$	lift coefficient for maximum lift-drag ratio
C_{L_α}	lift-curve slope per degree, averaged from $\alpha = 0^\circ$ over linear portion of curve
C_m	pitching-moment coefficient, $M_{cg}/qS\bar{c}$
C_{m_o}	pitching-moment coefficient at zero lift
$\partial C_m/\partial C_L$	static longitudinal stability parameter, averaged from $C_L = 0$ over linear portion of curve
$C_{p,b}$	base pressure coefficient, $\dfrac{p_b - p_\infty}{q}$

C_T thrust coefficient, T/qS

D drag adjusted to free-stream static pressure at model base, lb

L lift, lb

$(L/D)_{max}$ maximum lift-drag ratio

M free-stream Mach number

M_{cg} pitching moment about center-of-gravity location, in-lb

p_b static pressure at model base, lb/sq ft

p_∞ free-stream static pressure, lb/sq ft

q free-stream dynamic pressure, lb/sq ft

R Reynolds number based on wing mean aerodynamic chord

S total wing area, sq ft

T engine thrust, lb

α angle of attack of wing-chord plane assuming no leading-edge camber, deg

α_o angle of attack at zero lift, deg

APPARATUS AND METHODS

Tunnel and Model Support System

The tests were conducted in the Langley 8-foot transonic tunnel which is a dodecagonal, slotted-throat, single-return wind tunnel designed to obtain aerodynamic data through the speed of sound while minimizing the usual effects of blockage. The tunnel operates at approximately atmospheric stagnation pressures. Details of test-section design and flow uniformity are available in reference 4.

The model was attached to a sting support by the use of an electrical strain-gage balance located inside the fuselage. The sting support was cylindrical for 2.8 base diameters downstream of the model base and was fixed on the tunnel axis by two sets of struts projecting from the tunnel walls. Angled couplings in the sting were employed to maintain the model in a position near the center of the tunnel through the angle-of-attack range.

Model

The 0.04956-scale model of the Convair TF-102A training airplane used in this investigation was supplied by the contractor. Dimensional details of the model are presented in figure 1 and table I. The nose and canopy shapes of the TF-102A model and the F-102A model of reference 3 are compared in figure 2, and the total cross-sectional area distributions of the two models are presented in figure 3.

The wing of the TF-102A airplane is identical to the basic wing of the F-102A of reference 3 and was derived from a plane 60° delta wing with modified NACA 0004-65 streamwise airfoil sections (ref. 5) by extending the leading edge approximately 4.1 percent of the mean aerodynamic chord (this extension increased the leading-edge sweep angle to 60.14°) and by conically cambering the outboard 6.37 percent of the local semispan for a design lift coefficient of 0.15 at a Mach number of approximately 1.0 (ref. 6). The trailing edges of the wing tips outboard of the 82-percent semispan were deflected upward 10° about the elevon hinge line extended. The wing was constructed with a steel core covered by a tin-bismuth surface and had aluminum-alloy leading edges and steel tips.

Installed on the wing were two sets of chordwise fences. Upper-surface fences extending from 1.8 to 33 percent of the local chord were located at the 35-percent-semispan station, and wraparound fences extending from 22.7 percent of the local chord on the lower surface around the leading edge to 67 percent of the chord on the upper surface were located at the 66-percent-semispan station. These fences were identical to those used on the F-102A model and are discussed in detail in reference 2.

The fuselage was equipped with ram air inlets which were closed for these tests by means of faired plugs (fig. 1). In order to provide for side-by-side seating in the TF-102A airplane, the canopy was made higher and wider and extended farther forward on the nose portion of the fuselage as compared with the single-seat F-102A (fig. 2). Also, the air inlets were lowered on the sides of the fuselage, and the fuselage nose droop was decreased approximately 1°. The result of these extensive modifications to the forward portion of the fuselage just described is reflected in the cross-sectional-area distribution (fig. 3) as a substantial increase in the initial slope of the distribution and the creation of a severe area peak forward of the normal wing-fuselage area peak for the TF-102A model as compared with the F-102A model. The portion of the fuselage rearward of the canopy is identical to that of the F-102A airplane. The F-102A fuselage was designed according to the supersonic area-rule concept and was indented for the wing and tail in order to give a favorable total area distribution at a design Mach number of 1.2.

The vertical tail was the same as that of the F-102A model. It had a 60° sweptback leading edge, a 5° sweptforward trailing edge, and used modified NACA 0004-65 streamwise airfoil sections. A flat-plate antenna was located just above the rudder. The configuration had no horizontal tail, longitudinal control being obtained from wing elevons.

Measurements and Accuracy

Normal force, axial force, and pitching moment were measured with an internal strain-gage balance and converted to lift, drag, and pitching-moment coefficients. The pitching-moment coefficients are presented for a center-of-gravity location of 29.6 percent of the mean aerodynamic chord and 4.5 percent of the mean aerodynamic chord above the wing-chord plane. Accuracies of the coefficients are estimated to be within the following limits:

C_L . . . . ± 0.005 throughout C_L range
C_D . . . . ± 0.001 up to $C_L \approx 0.4$
C_m . . . . ± 0.001 throughout C_L range

The angle of attack was determined within ±0.15° by a pendulum-type inclinometer located in the sting support and by a calibration of sting and balance deflection due to model loads.

The Mach number was determined within ±0.003 from a calibration with respect to the pressure in the chamber surrounding the slotted test section. Base pressure coefficients were obtained from an orifice located inside the model and 2 inches forward of the plane of the base. The accuracy of the base pressure coefficients is estimated to be within ±0.005.

Tests

The complete model was tested with controls undeflected at Mach numbers from 0.60 to 1.135. The angle-of-attack range extended from an angle of attack of approximately 0° to angles varying from about 22° at a Mach number of 0.60 to 14° at Mach numbers above 1.00. The decrease in maximum attainable angle of attack with increasing Mach number was the result of tunnel power and balance limitations. The tests were made with the inlets faired closed.

The test Reynolds number based on the wing mean aerodynamic chord varied from 4.2×10^6 to 4.9×10^6 through the Mach number range (fig. 4).

Corrections

Subsonic boundary interference is minimized by the slotted test section, and no corrections for this interference have been applied. The effects of supersonic boundary-reflected disturbances were reduced by testing the model a few inches off the tunnel center line. However, it is possible that these disturbances caused small errors in the drag and pitching-moment measurements at Mach numbers of 1.075 and 1.135. It is believed, however, that these possible errors would have little effect on the trends indicated by, or the conclusions drawn from, the faired data plotted against Mach number in the summary and analysis plots.

The data have been adjusted to an assumed condition of free-stream static pressure acting over the model base by the base pressure coefficients presented in figure 5. No sting-interference corrections have been applied.

RESULTS

The tests were made with the air inlets faired closed and the data have been adjusted to represent free-stream static pressure at the model base using the base-pressure coefficients shown in figure 5.

The lift, drag, and pitching-moment data for the configuration are presented as a function of angle of attack or lift coefficient at constant Mach number in figure 6.

A summary and brief analysis of the aerodynamic characteristics are presented as a function of Mach number in figures 7 to 11. These figures also include a comparison with the aerodynamic characteristics obtained from tests of a 0.04956-scale model of the Convair F-102A airplane as reported in reference 3. The model of reference 3 also had the air inlets faired closed, and the data have been adjusted to simulate free-stream conditions at the model base and are computed for the same center-of-gravity location as the present model.

DISCUSSION

Lift and Pitching-Moment Characteristics

Lift characteristics.- The lift curves for the TF-102A airplane were generally linear over the Mach number and angle-of-attack range tested. (See fig. 6(a).) The angle of attack for zero lift was approximately 1.5° over the Mach number range as compared with 1.4° for the F-102A model of reference 3. (See fig. 7.) The lift-curve slope of the TF-102A airplane varied from 0.045 at a Mach number of 0.60 to about 0.059 at a Mach number of 1.05 (fig. 7); this variation represents a decrease of approximately 2 percent with respect to the lift-curve slope of the F-102A.

Pitching-moment characteristics.- The pitching-moment curves for the TF-102A airplane (fig. 6(b)) were nearly linear and indicated static longitudinal stability over the lift and Mach number range investigated. However, as was shown for the F-102A, neutral stability was approached at a Mach number of 0.60 for a small lift-coefficient range beginning at about 0.60. This destabilizing change in the slope of the pitch curve suggests the possibility of a mild pitch-up tendency in this region (ref. 7). The pitching-moment coefficient at zero lift for the TF-102A airplane was of the order of 0.01 over the Mach number range and agrees closely with the value for the F-102A (fig. 8). The value of the static-longitudinal-stability parameter $\partial C_m/\partial C_L$ for the TF-102A airplane decreased from -0.07 at a Mach number of 0.60 to approximately -0.18 at Mach numbers above 1.025 (fig. 8); this decrease indicates a rearward shift in aerodynamic-center location of 11 percent of the wing mean aerodynamic chord. The only significant difference between the static margins of the TF-102A and F-102A configurations occurred at Mach numbers above 1.025 where the value for the TF-102A was the smaller by approximately 2 percent of the mean aerodynamic chord.

In general, it was apparent that conversion of the F-102A interceptor into the TF-102A training airplane had little effect on the lift and pitching-moment (static longitudinal stability) characteristics. This was not unexpected since both configurations retained the same lifting surfaces in identical positions on fuselages which were the same length and differed in size and shape only from the nose to approximately the leading edge of the wing-fuselage juncture (fig. 2). Effects of the larger forward portion of the fuselage for the TF-102A airplane were probably small and generally confined to a slight forward shift in fuselage center-of-pressure location. Such phenomena as separation of the flow over the rearward portion of the canopy or behind the side air inlets would affect only a very small portion of the total lifting surface.

Drag Characteristics

Drag at zero lift.- The subsonic (0.6 Mach number) zero-lift drag coefficient for the TF-102A airplane was 0.002, or about 17 percent, higher than the value for the F-102A (fig. 9). This was probably caused primarily by flow separation associated with the contours of the rearward portion of the canopy and the region rearward of the air inlets which resulted from the increased size of the forward portion of the fuselage of the trainer configuration. In addition, the larger fuselage frontal area (fig. 2) represents an increase in total airplane frontal area of approximately 14 percent as compared with the F-102A airplane.

Although there was a significant steady increase in drag coefficient with increases in Mach number above 0.60, the zero-lift transonic drag rise for the TF-102A airplane could probably be considered to begin at a Mach number of approximately 0.85 as compared with 0.91 for the F-102A (fig. 9). This decrease in drag-rise Mach number can be associated with the decrease in equivalent forebody fineness ratio from 3.1 for the F-102A airplane to approximately 2 for the TF-102A. The peak zero-lift drag coefficients occurred near a Mach number of 1.075 for both configurations.

The zero-lift peak-wave-drag coefficient, taken as the difference in drag coefficient between the Mach numbers of 0.85 and 1.075, was 0.024 for the TF-102A airplane as compared with 0.016 for the F-102A, or an increase of 50 percent. The wave drag of a wing-body combination near the speed of sound depends upon the axial distribution of total cross-sectional area, and, in order to keep the wave drag to a minimum, the area distribution for a given equivalent body fineness ratio must be kept as smooth as possible. On this basis, it can readily be seen from figure 3 that the increased wave drag for the TF-102A airplane is associated with the enlarged canopy and attendant modifications used on the trainer airplane which have resulted in an extremely unfavorable area distribution characterized by an increased initial slope, a severe forebody peak, and a sizable dip between the forebody and the usual wing-fuselage peaks. The forebody peak, in particular, indicates the presence of severe velocity gradients which usually result in large shock and separation losses. The higher subsonic drag level combined with the large increase in transonic wave drag for the TF-102A airplane as compared with the F-102A (fig. 9) resulted in total zero-lift drag-coefficient increases which were as high as 0.012, or approximately 42 percent, at a Mach number of 1.075.

Drag at lifting conditions.- The differences in drag coefficient between the TF-102A and the F-102A airplanes were approximately the same at lift coefficients of 0.2 and 0.4 as they were at a lift coefficient of zero (fig. 9); this indicates that enlargement of the forward portion of the fuselage had little effect on the drag due to lift. This result is not surprising since the wings and the fuselage in the region of the wings were identical for both configurations.

The maximum lift-drag ratio for the TF-102A airplane decreased from 9.5 at a Mach number of 0.60 to approximately 5 at Mach numbers above 1.025 (fig. 10); thus, a loss varying from 5 percent at a Mach number of 0.60 to 20 percent at Mach numbers above 0.90 is indicated with respect to the F-102A. These losses were caused primarily by the increase in zero-lift drag previously discussed. The lift coefficient for maximum lift-drag ratio was somewhat higher for the TF-102A airplane than for the F-102A at the higher Mach numbers and varied from a lift coefficient of 0.22 at subsonic speeds to 0.37 at a Mach number of 1.075 (fig. 10).

Performance comparison.- In order to obtain some indication of the effect on performance of converting the F-102A interceptor into the TF-102A training airplane, drag coefficients for trimmed level flight for the two configurations are compared with a typical-engine thrust coefficient curve at an altitude of 35,000 feet in figure 11. A wing loading of 36 pounds per square foot was assumed for each airplane, and the resulting lift coefficients required varied from 0.286 at a Mach number of 0.60 to 0.080 at a Mach number of 1.135. The trimmed drag coefficients for the F-102A airplane were obtained from the data of reference 3. The trimmed drag for the TF-102A airplane was estimated by assuming that the increment in drag due to trimming the airplane from the condition with controls undeflected was the same as that for the F-102A. This assumption was considered reasonable because of the previously described static-longitudinal-stability agreement and the similarity in longitudinal control configuration for the two models. In addition, by using the comparison of Convair F-102 full-scale flight data and model data presented in reference 8 in an effort to present a more realistic performance comparison, the trimmed drag coefficients for the TF-102A and F-102A models tested at Reynolds numbers of approximately 4.8×10^6 have been reduced by 0.0025 to simulate full-scale aircraft flying at Reynolds numbers of the order of 50×10^6. The available-thrust curve represents a turbojet engine having a static sea-level thrust rating of 16,000 pounds with afterburner.

The comparison presented in figure 11 indicates that at an altitude of 35,000 feet conversion of the F-102A supersonic interceptor into the TF-102A training airplane has reduced the maximum level flight Mach number from at least 1.15 to slightly less than 1.0. Since for a given jet-engine—airplane combination the maximum Mach numbers attained are usually less at other altitudes than at approximately 35,000 feet, it was apparent that with the assumed engine the TF-102A airplane would be incapable of supersonic speeds in level flight. In addition, climb performance, maximum altitude, and range would be significantly reduced for the TF-102A airplane as compared with the F-102A at the higher subsonic Mach numbers.

CONCLUSIONS

An investigation of the basic aerodynamic characteristics of a 0.04956-scale model of the Convair TF-102A airplane at transonic speeds and a comparison with data obtained from previous tests of a 0.04956-scale model of the Convair F-102A airplane indicated the following conclusions:

1. The TF-102A airplane was longitudinally stable over the Mach number and angle-of-attack range tested. With increases in Mach number from 0.60 to approximately 1.05, the lift-curve slope increased from 0.045 to 0.059 and the aerodynamic center shifted rearward 11 percent of the wing mean aerodynamic chord.

2. The zero-lift drag coefficient for the TF-102A airplane increased by 0.024, or 145 percent, between the Mach numbers of 0.85 and 1.075. The maximum lift-drag ratio decreased from 9.5 at a Mach number of 0.60 to 5.0 at Mach numbers above 1.025.

3. Conversion of the F-102A supersonic interceptor into the TF-102A trainer had little effect on the lift and pitching-moment characteristics and drag due to lift. However, as compared with the F-102A, the zero-lift drag-rise Mach number for the TF-102A airplane was reduced by at least 0.06, the zero-lift peak wave drag was increased 50 percent, and the maximum lift-drag ratio was reduced up to 20 percent.

4. It was estimated that with identical engines maximum Mach number, maximum altitude, climb performance, and range would be significantly reduced for the TF-102A airplane as compared with the F-102A.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., May 1, 1957.

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Eugene C. Draley
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Chief of Full-Scale Research Division

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TABLE I.- DIMENSIONS OF THE 0.04956-SCALE MODEL

OF THE CONVAIR TF-102A AIRPLANE

Wing:	
Airfoil section	Modified NACA 0004-65 with leading-edge camber and wing tips outboard of 0.82b/2 deflected upward 10° about elevator hinge line extended
Total area, sq ft	1.709
Aspect ratio	2.1
Taper ratio	0
Incidence, deg	0
Dihedral, deg	0
Elevator area rearward of hinge line, sq ft	0.166
Vertical tail:	
Airfoil section	Modified NACA 0004-65
Exposed area, sq ft	0.1704
Aspect ratio	1.1
Taper ratio	0
Fuselage:	
Length, in.	34.161
Frontal area (including canopy), sq ft	0.1014
Fineness ratio (including canopy)	7.92
Total base area, sq ft	0.0236

[redacted]

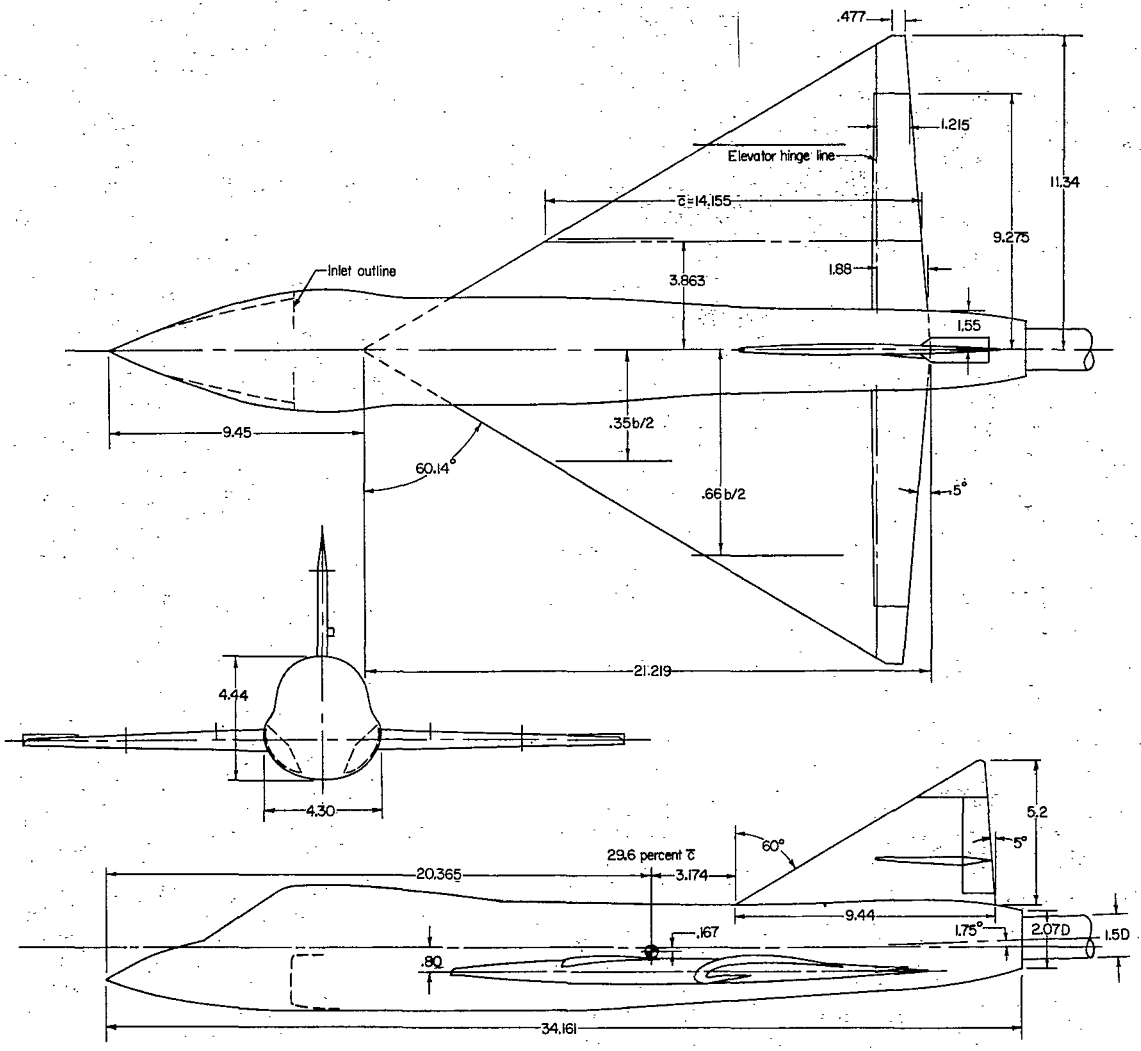

Figure 1.- Drawing of a 0.04956-scale model of the Convair TF-102A airplane with air inlets faired closed. All dimensions are in inches unless otherwise noted.

[redacted]

[redacted]

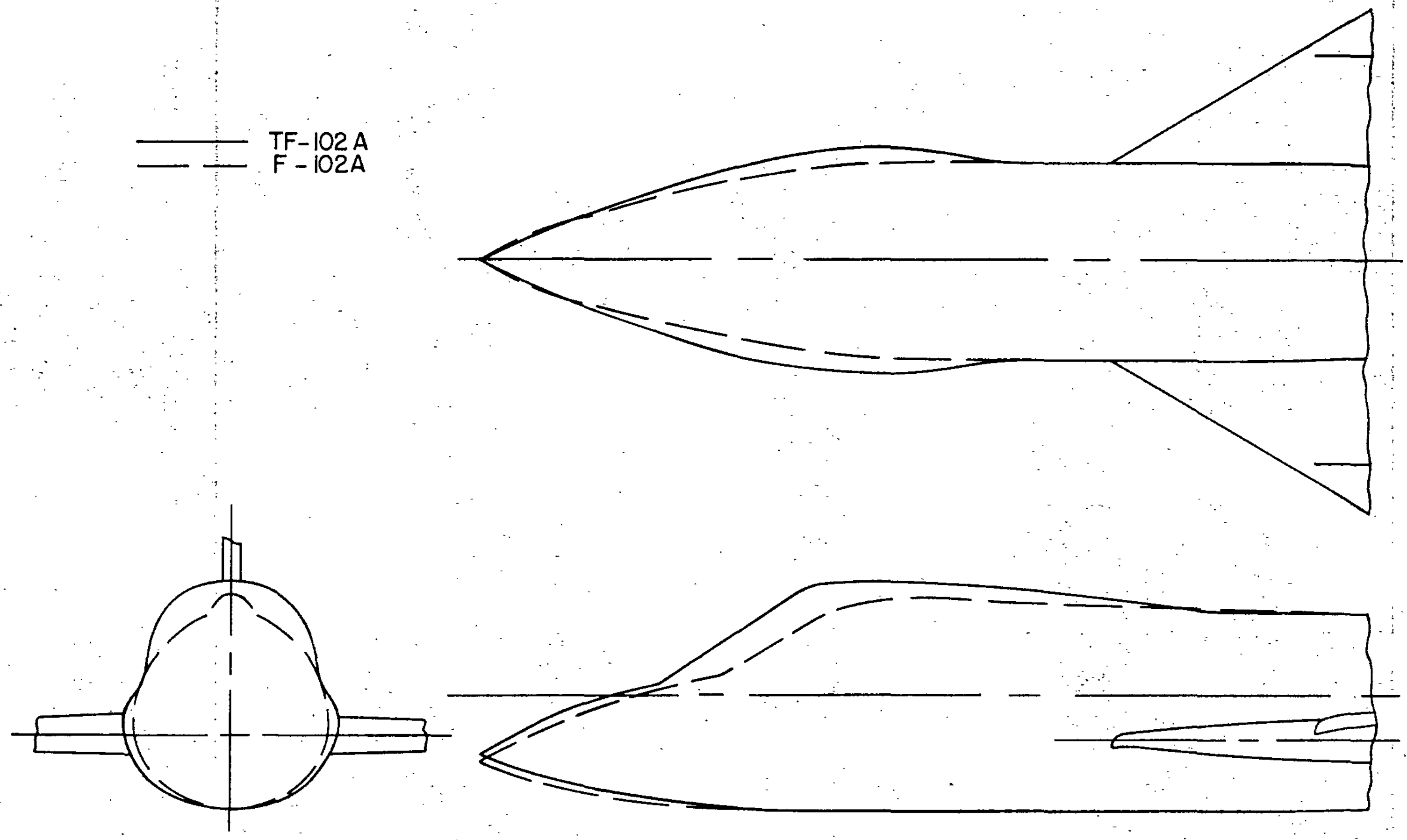

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Figure 2.- Comparison of 0.04956-scale models of the Convair TF-102A and F-102A airplanes with air inlets faired closed.

[redacted]

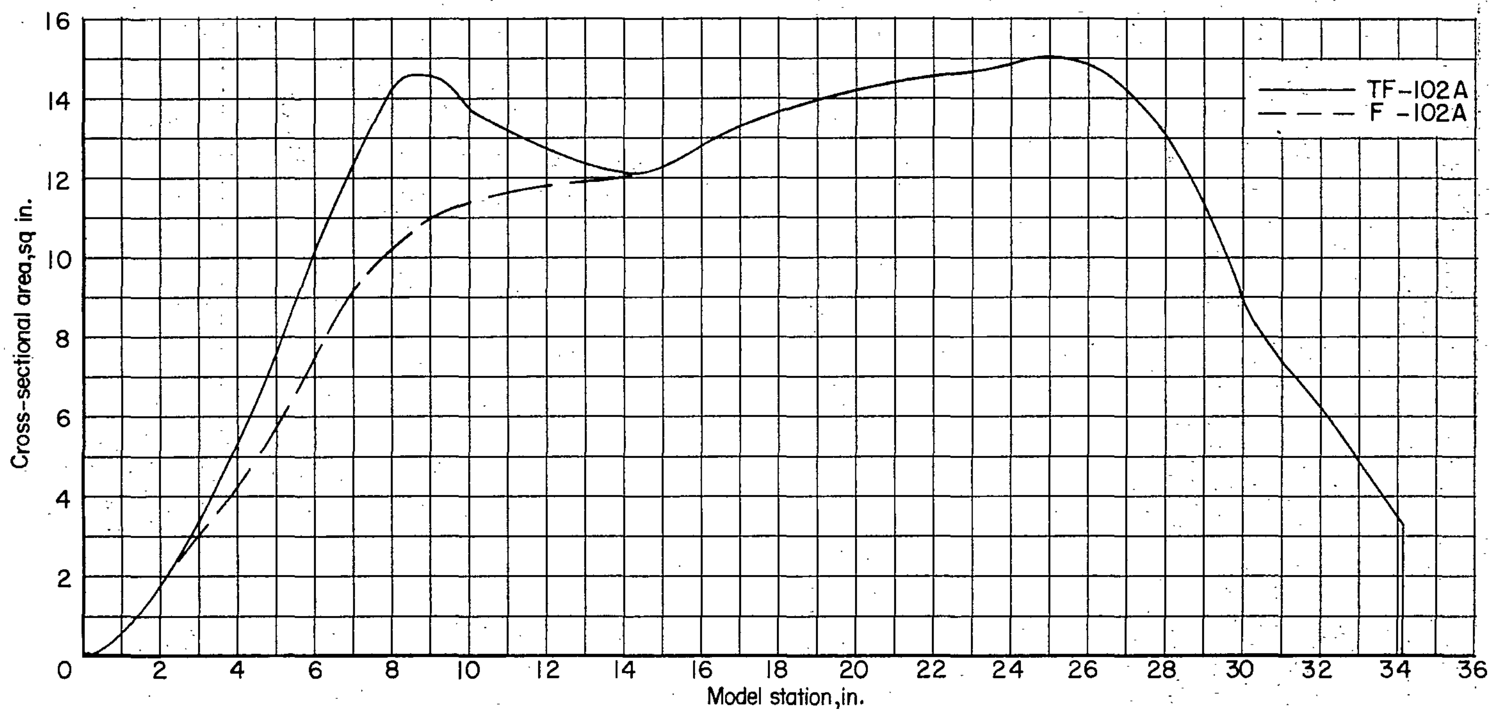

Figure 3.- Cross-sectional area distributions of 0.04956-scale models of the Convair TF-102A and F-102A airplanes with air inlets faired closed.

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[redacted]

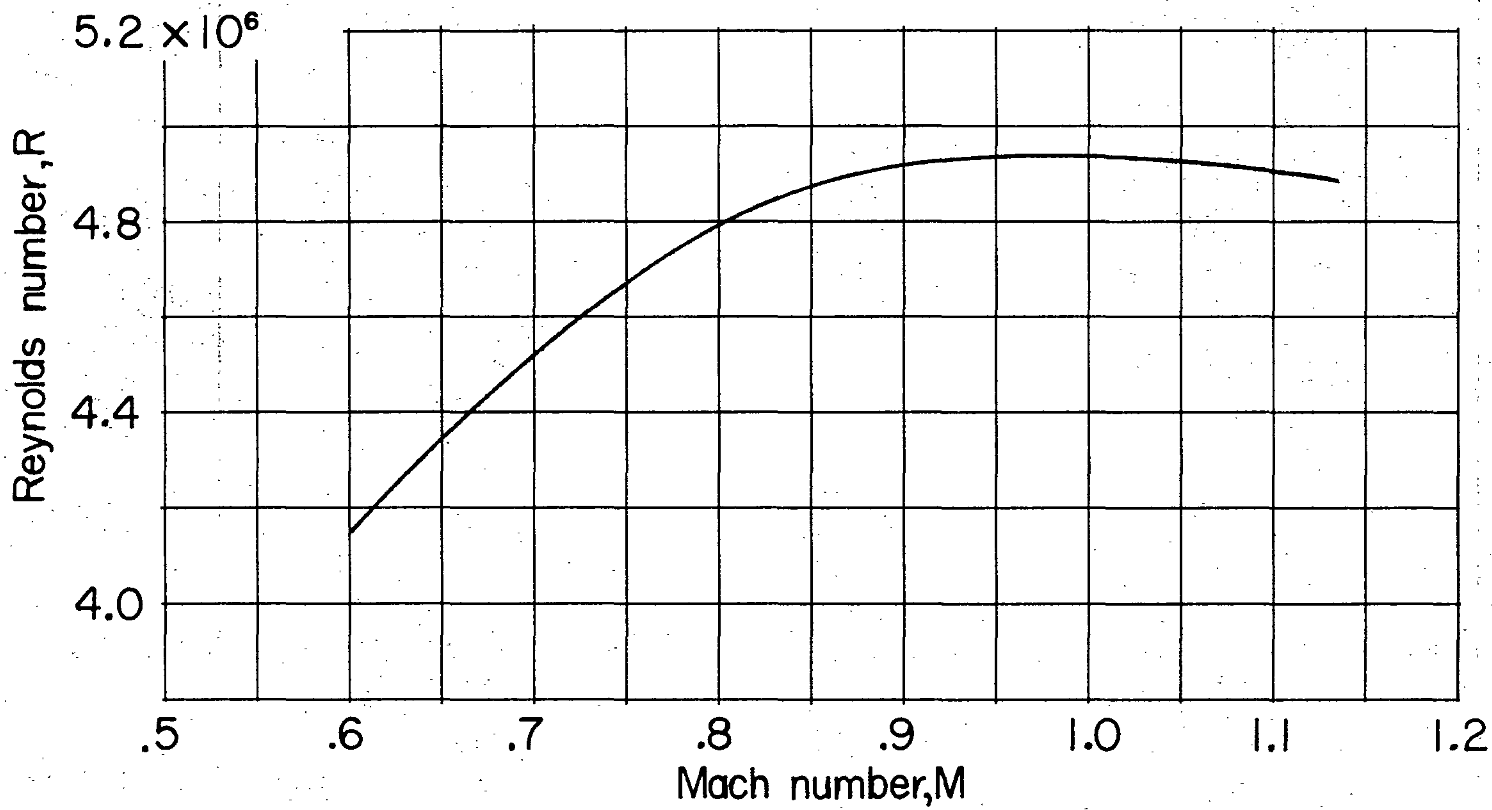

Figure 4.- Variation with Mach number of average test Reynolds number based on the wing mean aerodynamic chord.

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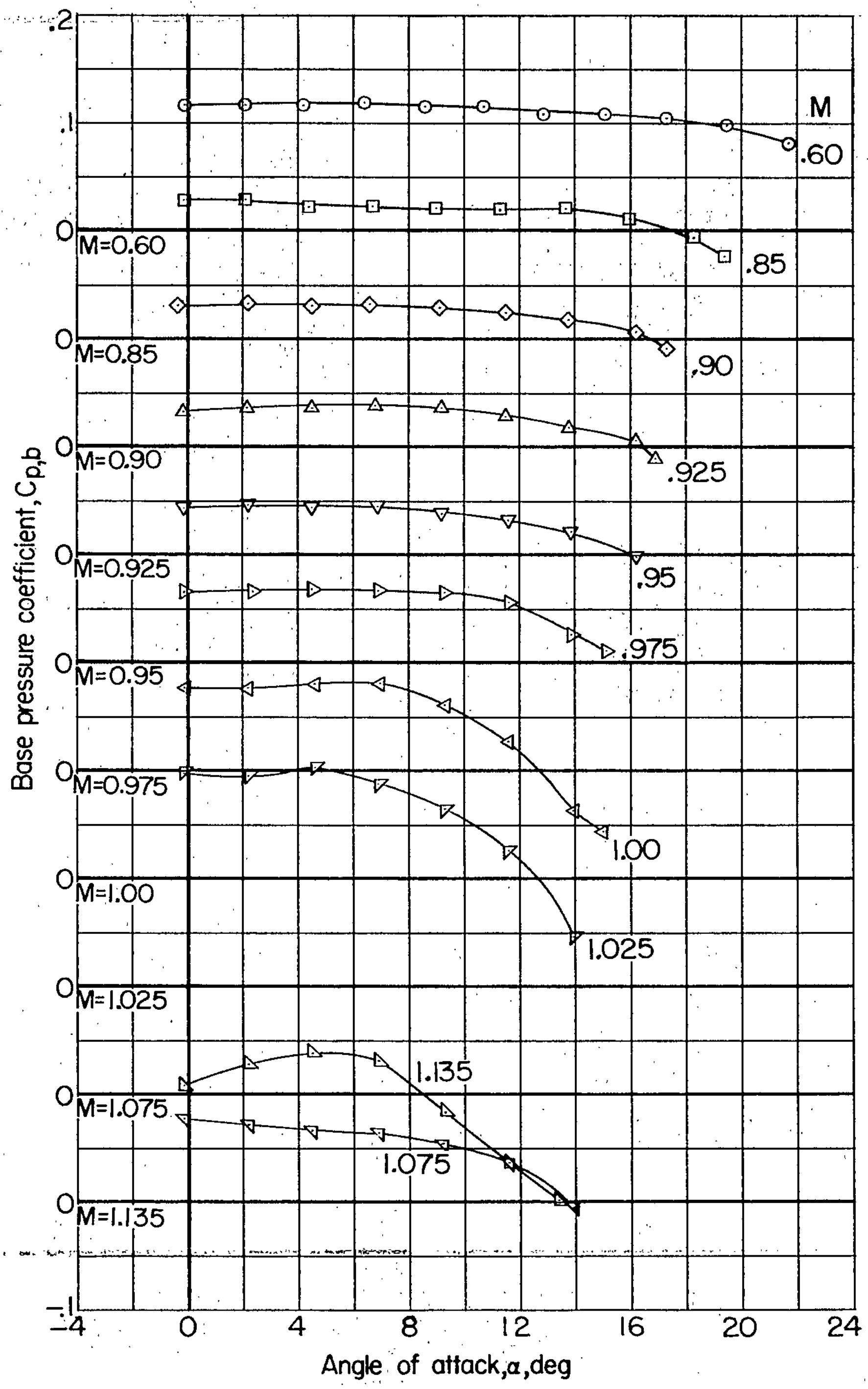
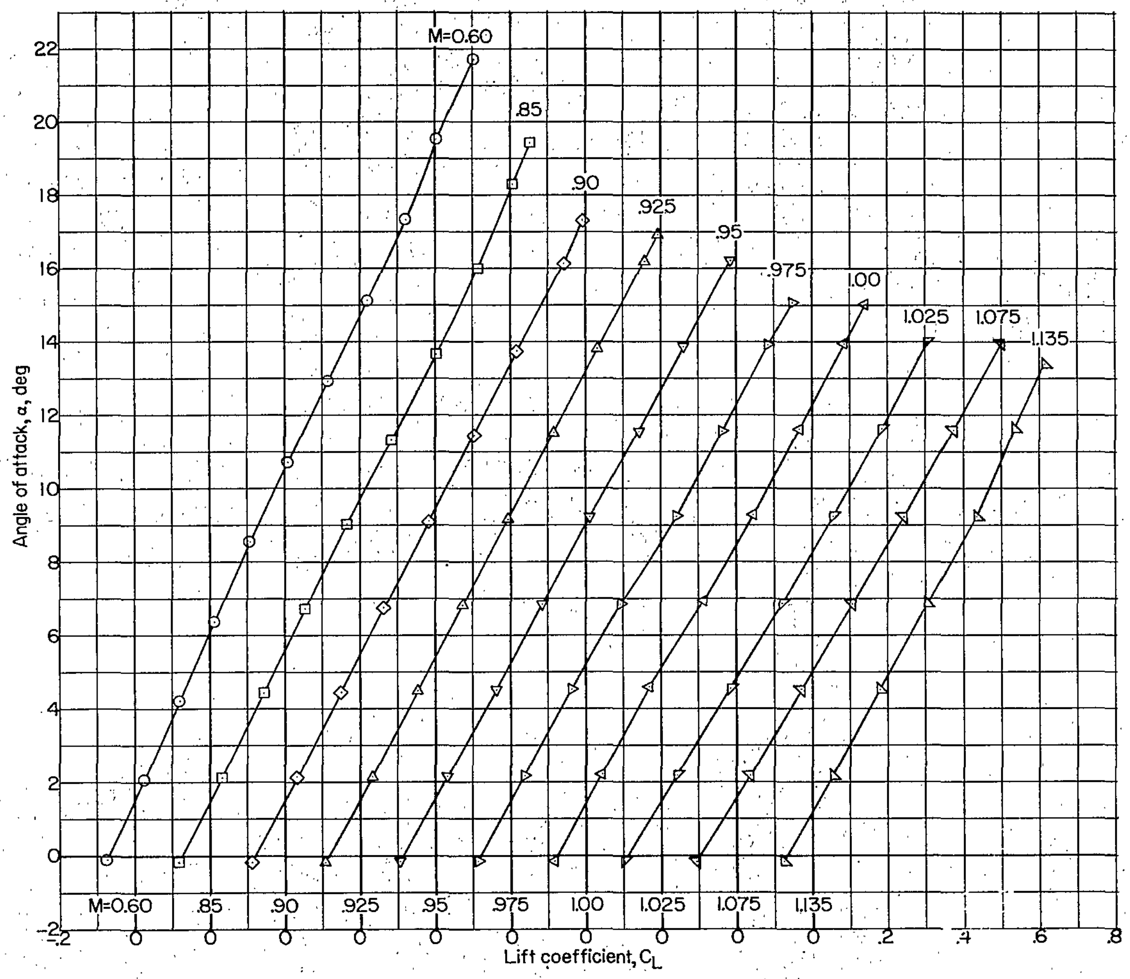
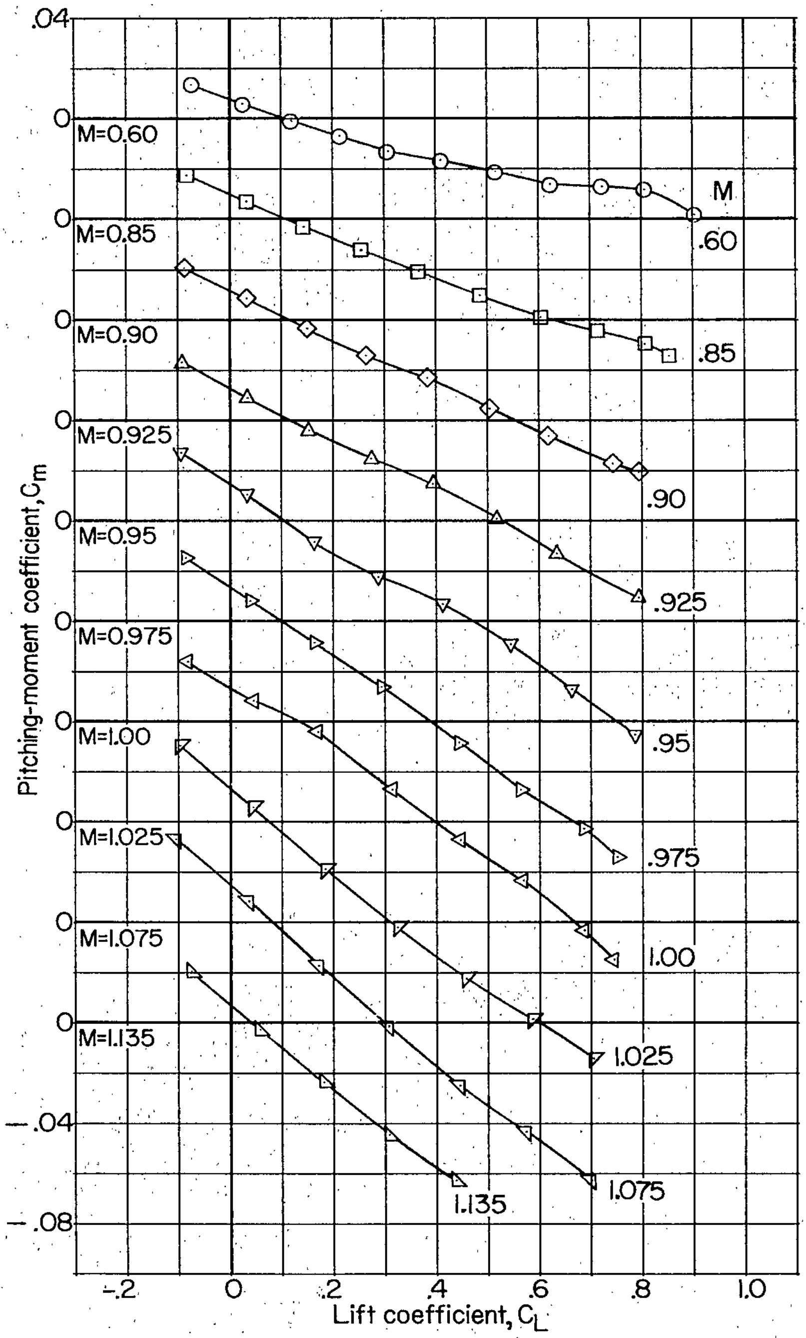


Figure 5.- Base pressure coefficients for the TF-102A model.



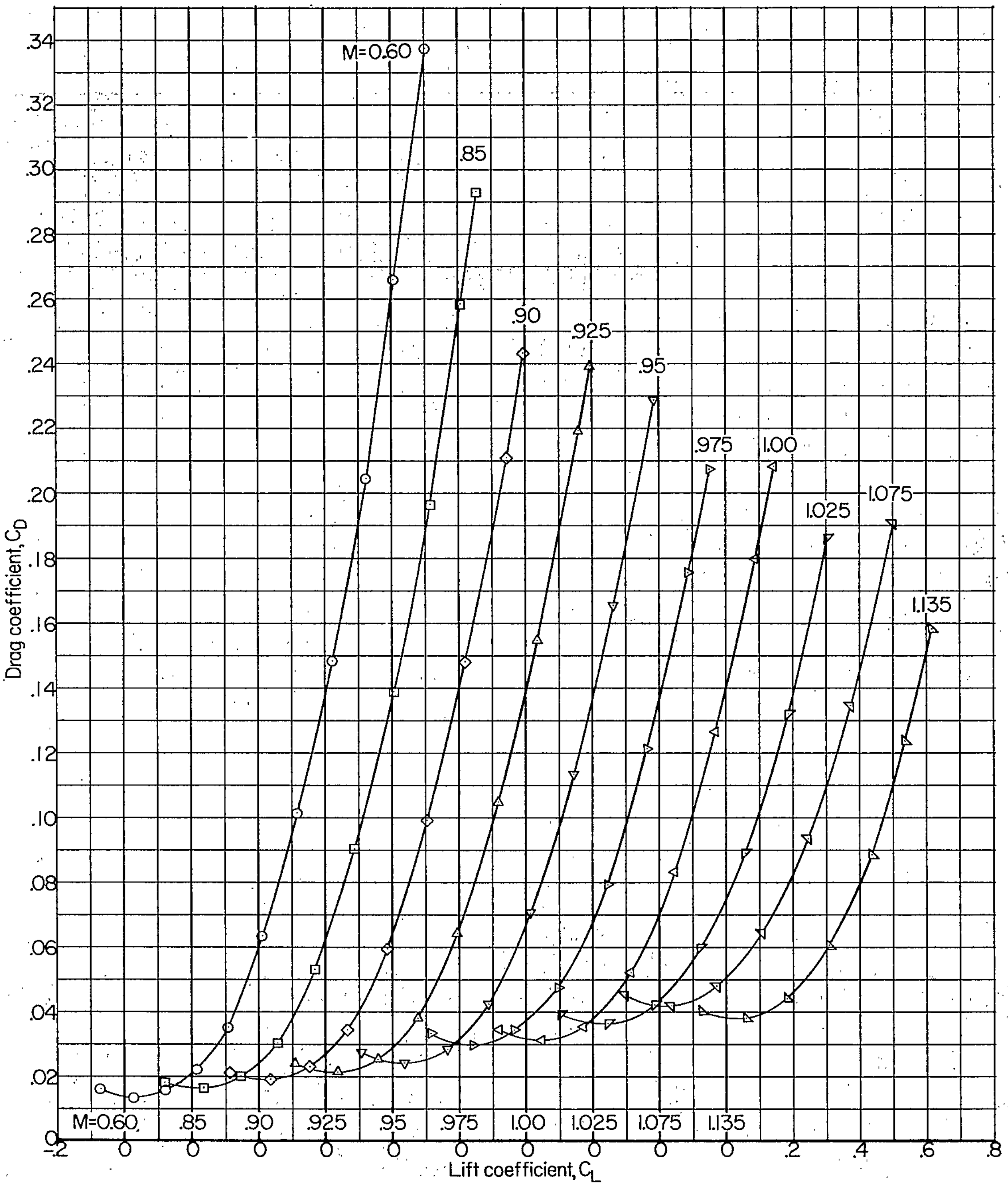
(a) Variation of α with C_L.

Figure 6.- Basic aerodynamic characteristics of the TF-102A model.



(b) Variation of C_m with C_L.

Figure 6.- Continued.



(c) Variation of C_D with C_L.

Figure 6.- Concluded.

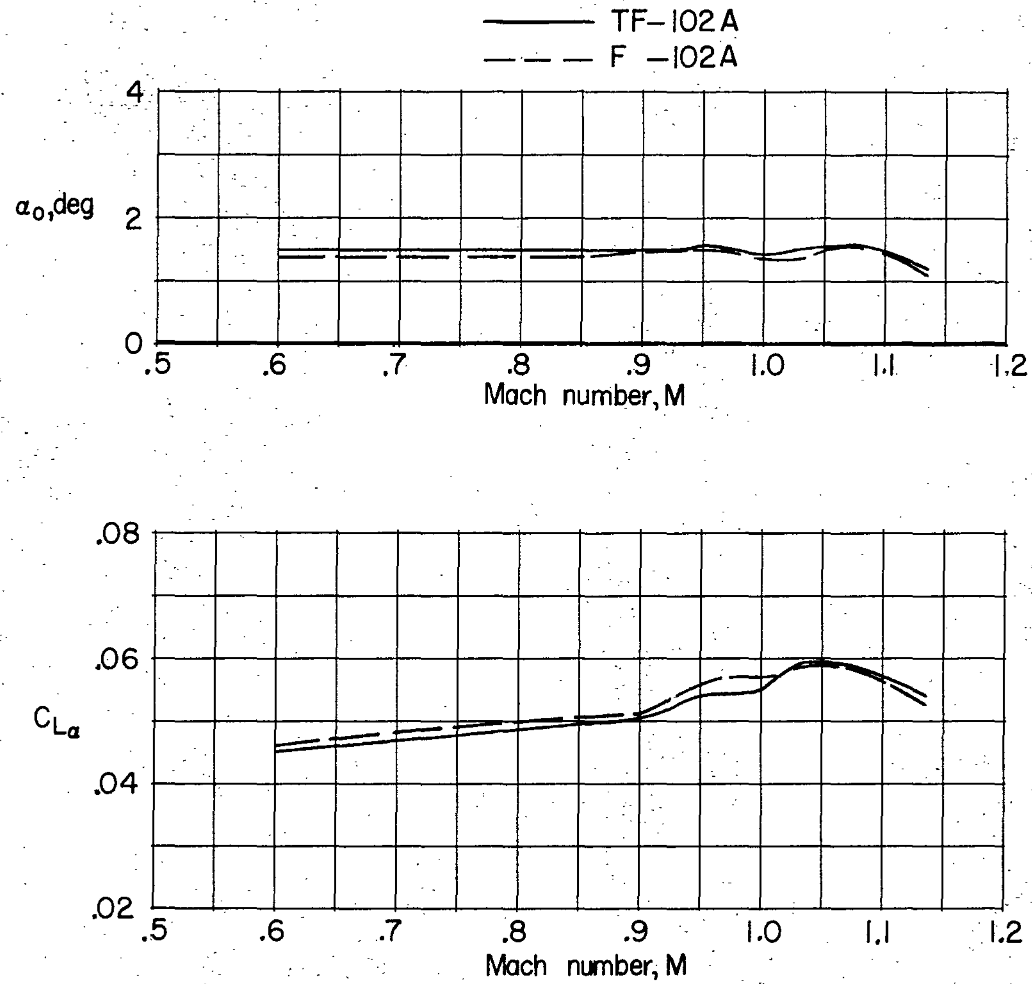


Figure 7.- Variation with Mach number of angle of attack at zero lift and lift-curve slope for the TF-102A and F-102A models.

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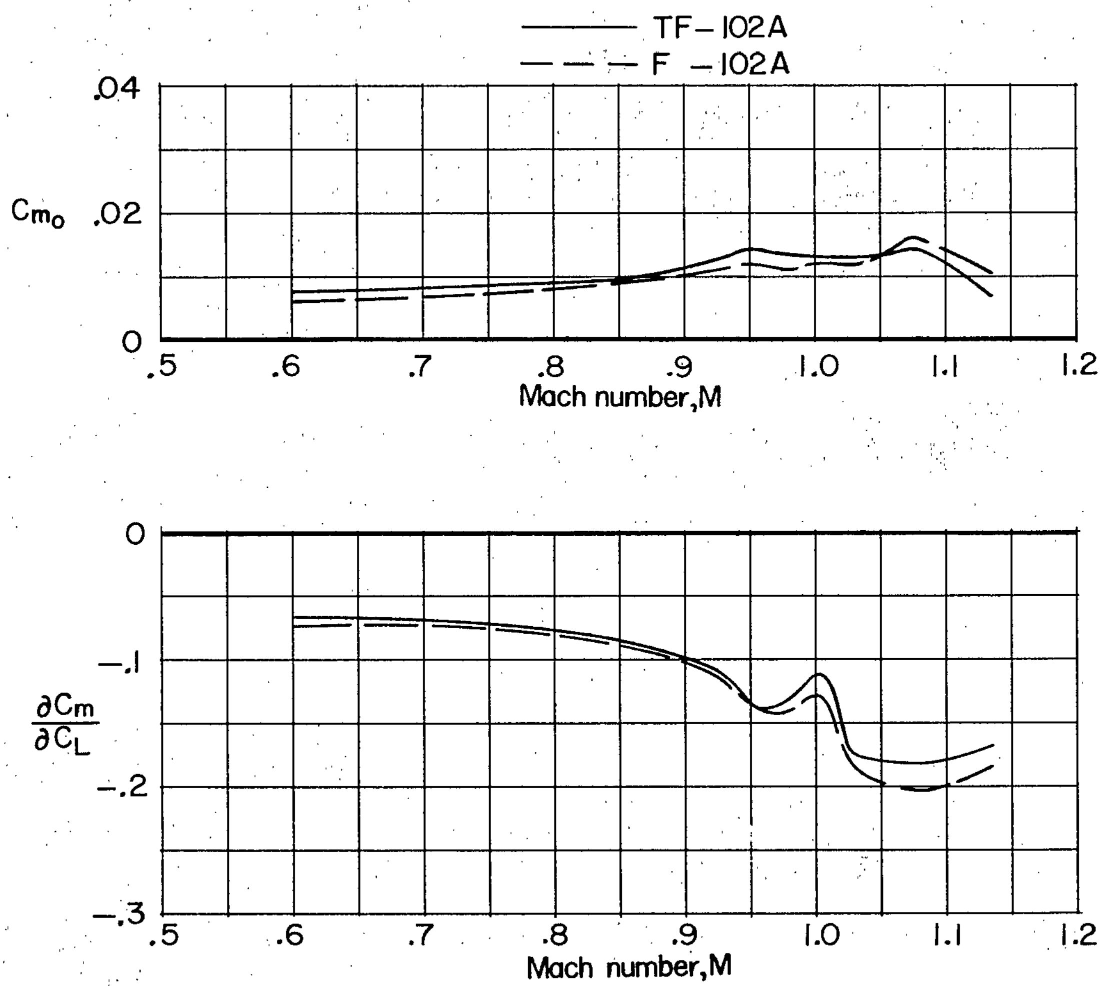

Figure 8.- Variation with Mach number of pitching-moment coefficient at zero lift and static longitudinal stability parameter for the TF-102A and F-102A models.

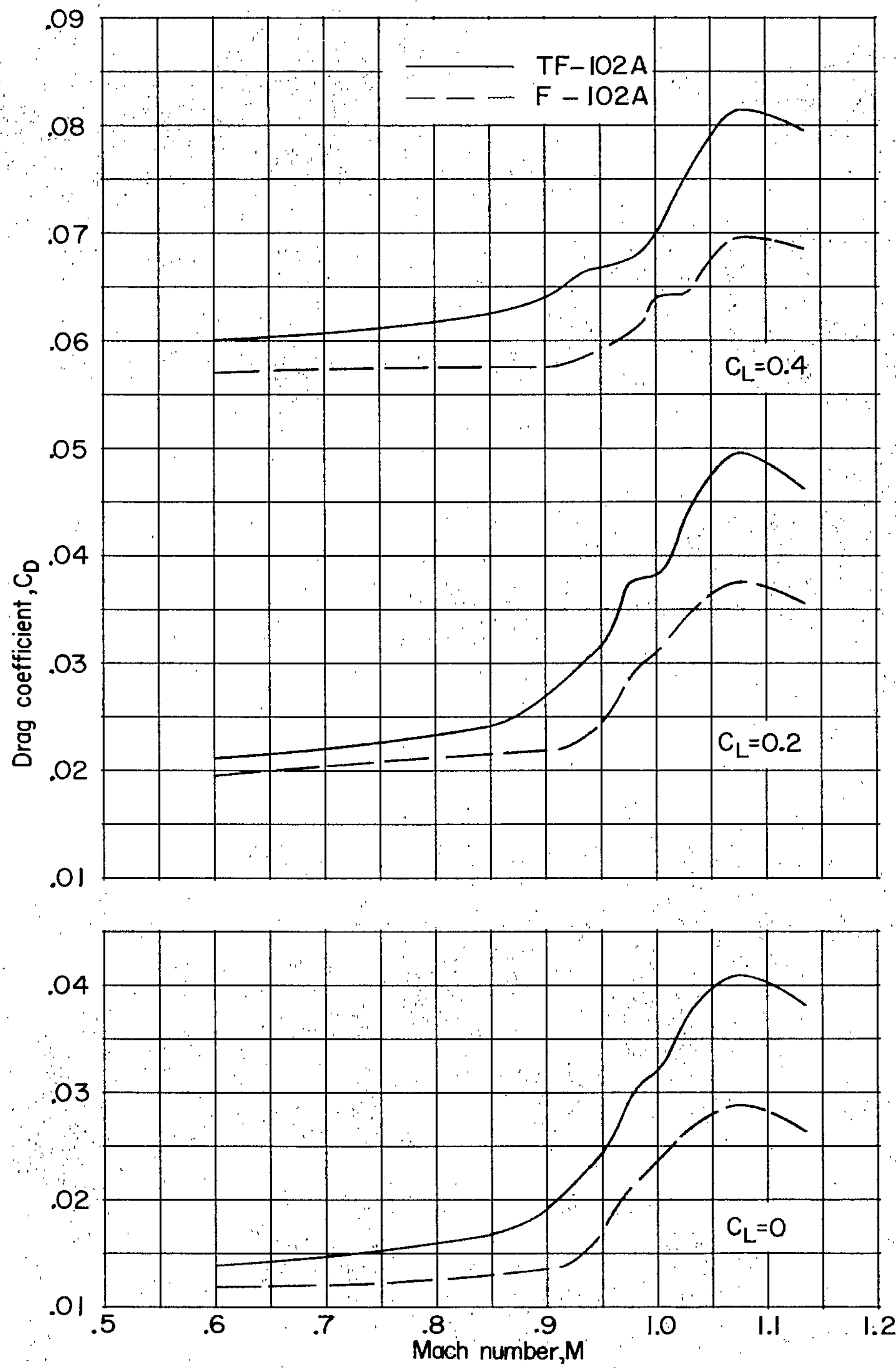

Figure 9.- Variation of drag coefficient with Mach number for several lift coefficients for the TF-102A and F-102A models.

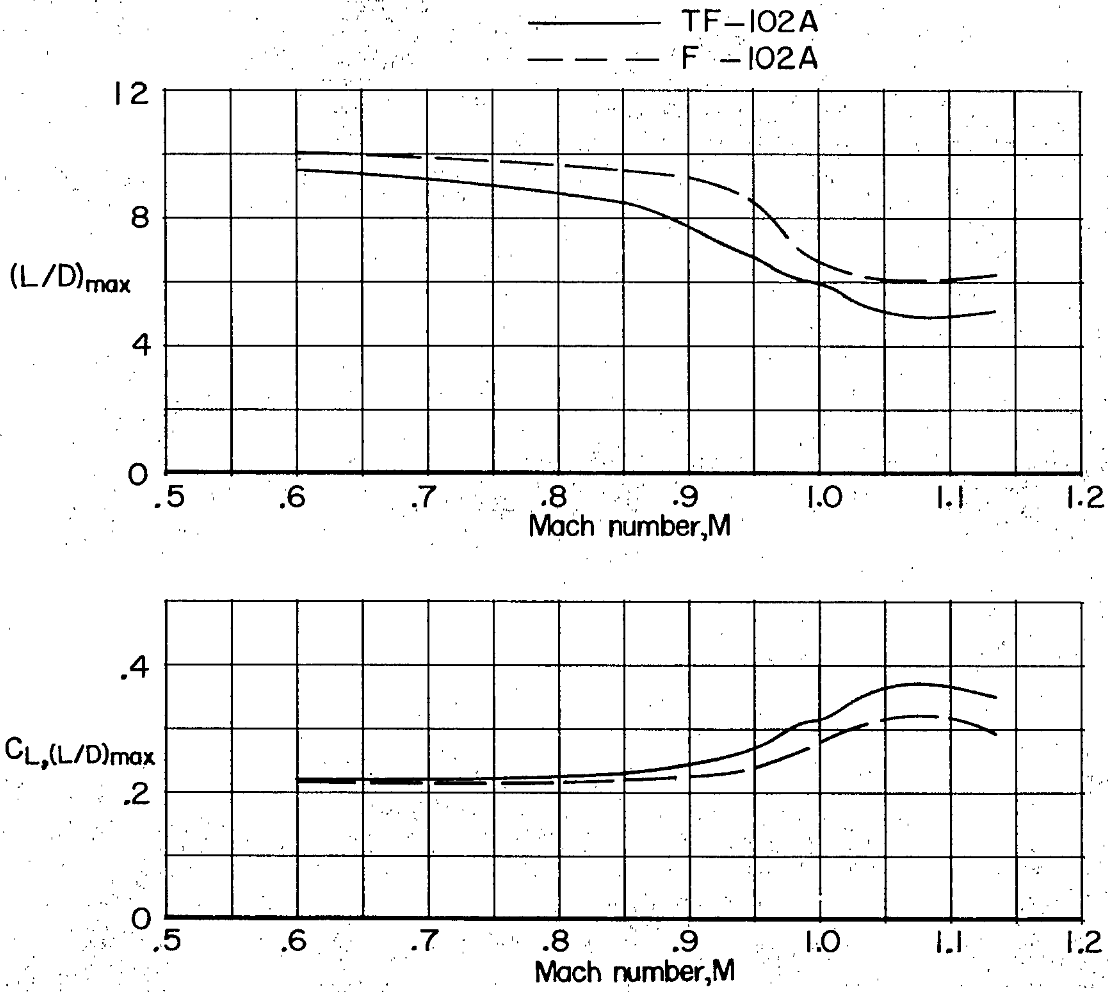

Figure 10.- Variation with Mach number of maximum lift-drag ratio and lift coefficient for maximum lift-drag ratio for the TF-102A and F-102A models.

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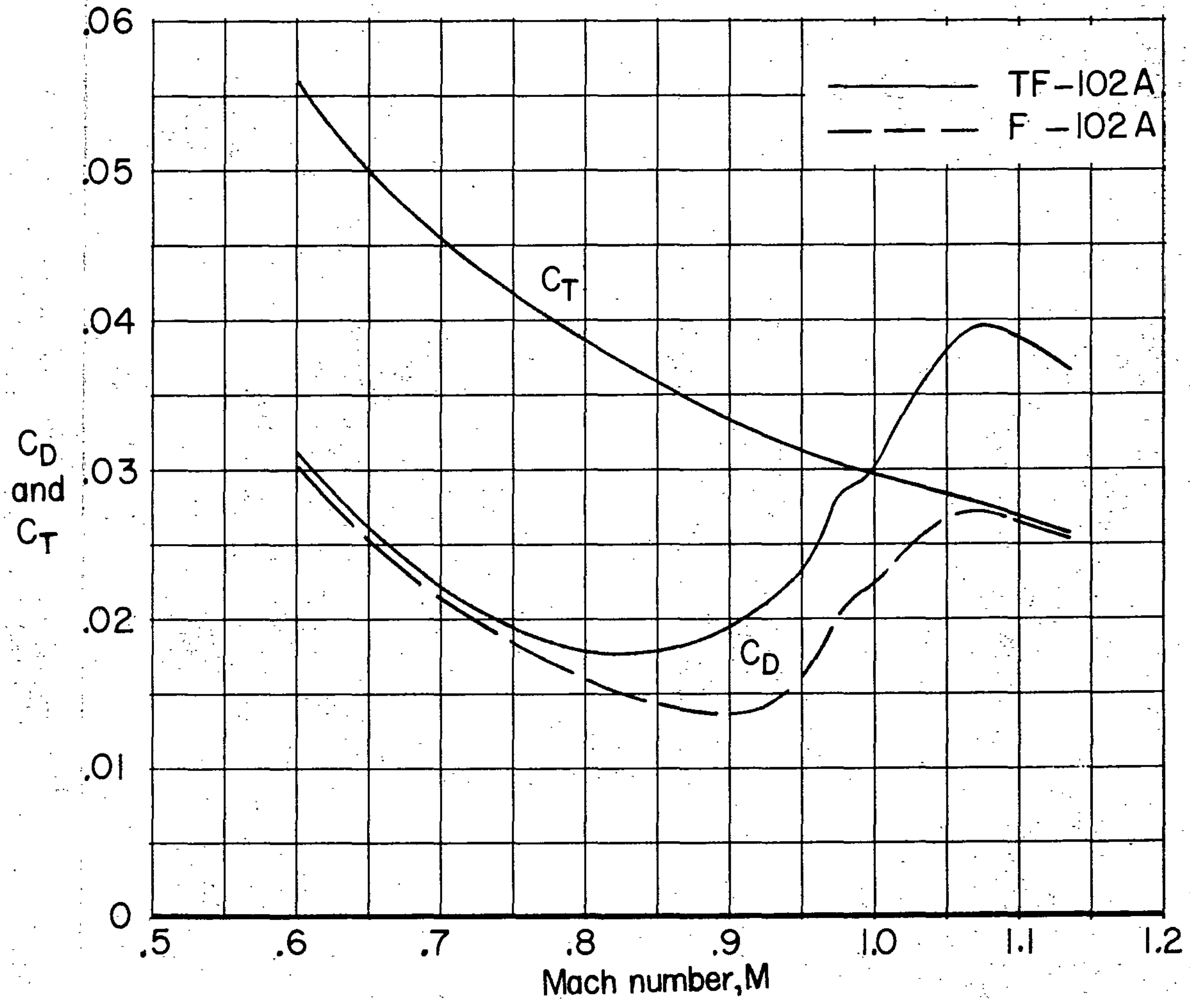

Figure 11.- Variation with Mach number of engine thrust coefficient and airplane drag coefficient for trimmed level flight of the TF-102A and F-102A airplanes at an altitude of 35,000 feet.

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ABSTRACT

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