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RESEARCH MEMORANDUM

EFFECT OF AIRFOIL SECTION AND TIP TANKS ON THE AERODYNAMIC CHARACTERISTICS AT HIGH SUBSONIC SPEEDS OF AN UNSWEPT WING OF ASPECT RATIO 5.16 AND TAPER RATIO 0.61

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

An investigation of the effect of two wing sections and a tip tank on the aerodynamic characteristics of a rigid unswept wing was made in the Langley high-speed 7- by 10-foot tunnel over a Mach number range extending from 0.60 to 0.90.

Analysis of the results indicates that airfoil section had an appreciable effect on the aerodynamic-center location of the wing, that the trailing-edge angle of the airfoil section was a principal factor in controlling this effect at high subsonic Mach numbers, that the tip tank produced less than 1.5-percent change in the aerodynamic-center location of the wing regardless of airfoil section, that the effective aspect ratio change produced by the end-plate effect of the tip tank was appreciably larger when the gap between the tank and wing was sealed, and that the unstable pitching moment of the tank about a point located at 40 percent of the wing-tip chord was neutralized by a horizontal tank fin which was 23 percent of the projected area of the tank.

INTRODUCTION

The behavior of auxiliary fuel tanks mounted at the tips of straight wings is well established (reference 1) in the region of speeds where compressibility and aeroelastic effects are of secondary importance. As the speeds of aircraft increase, however, compressibility and aeroelasticity become of major importance even on a wing without a tip tank so that the necessity for obtaining information on the effect of tip-mounted tanks at high speeds is apparent.

The results presented in this paper were obtained in the Langley high-speed 7- by 10-foot tunnel and include data obtained on two identical wing plan forms having different airfoil sections, with and without a tip tank, over a Mach number range from 0.60 to 0.90. Also shown are the effects of two modifications to the trailing portion of one of the airfoil sections. Modifications to the basic profile were accomplished by extending the wing trailing edge. The lift and pitching-moment coefficients of the tank alone in the presence of the rigid-unswept-wing model are included in the results presented. Pitching moments of the tank alone are presented about the 40-percent-tip-chord point which is considered representative of the elastic-axis location of a flexible wing. The effect of horizontal tank stabilizing fins on the properties of the tank alone in presence of the wing are shown.

COEFFICIENTS AND SYMBOLS

The coefficients and symbols referred to in this paper are defined as follows:

C_L lift coefficient (Twice panel lift/qS)

C_m pitching-moment coefficient, referred to the $0.25\bar{c}$ (original plan form) (Twice panel pitching moment/$qS\bar{c}$)

C_D drag coefficient (Twice panel drag/qS)

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

M Mach number (V/a)

R Reynolds number ($\rho V\bar{c}/\mu$)

q dynamic pressure, pounds per square foot $\left(\frac{1}{2}\rho V^2\right)$

ρ mass density of air, slugs per cubic foot

V velocity of air, feet per second

μ absolute viscosity, pound-seconds per square foot

a velocity of sound, feet per second

S twice panel area of semispan model (see table I)

$\bar{c}$ mean aerodynamic chord (see table I)

c chord, inches

A aspect ratio, b^2/S

b twice panel span of semispan model (16.44 in.)

α angle of attack of the wing chord line

ϕ trailing-edge angle, degrees (included angle between upper and lower surfaces at last 5 percent of chord)

A_f/A_t ratio of area of fin to projected area of tank

Subscripts:

f fin

t tank

APPARATUS AND MODELS

Force and moment measurements were made with a strain-gage balance mounted on a wall of the Langley high-speed 7- by 10-foot tunnel and sealed to prevent leakage of air into the flow field of the model. A drawing of the test setup with the models of the wing with the tip tank in place is presented in figure 1. Surveys have indicated that wall boundary-layer effects may be eliminated by locating the test model approximately 3 inches from the tunnel wall. At this location a boundary-layer plate was installed by a sealed fairing through which extended the strain-gage-balance model support bracket. A small end plate was added to the wing root at a distance of 1/32 of an inch from the boundary-layer plate to cover the unported area of the boundary-layer plate around the model support bracket and to minimize the interference effects of the small boundary layer built up over the boundary-layer plate. Leakage around the root chord of the wing was minimized by sealing the balance and the support fairing and maintaining the smallest practical clearance between boundary-layer plate and the wing end plate.

Two small aluminum semispan wings of identical plan form but of differing airfoil section (referred to herein as section A which was an NACA 65-210 profile and section B which was similar to an NACA 66_1-212 profile) were used in this investigation. The aspect ratio of the original plan form was 5.16 and the taper ratio, 0.61. The ordinates of each of the two airfoil sections, along with a sketch of the profile shapes, are presented in table II. Modifications made to section B are shown in figure 2. Modification 1 was made by extending the trailing edge of the chord 3 percent and accentuating the trailing-edge cusp aft of the 80-percent-chord point. Modification 2 was a flat-sided addition to the wing aft of the 75-percent-chord point that was 4-percent-chord thick at the trailing edge of the basic wing section and represented a 3.4-percent-chord extension at the root and a 5.3-percent-chord extension at the tip. The trailing edge of modification 2 was a semicircular form. The trailing-edge angles of section B with modifications 1 and 2 were designed to approach the trailing-edge angle of section A ($\phi = 7.00^{\circ}$). Presented in table I are pertinent geometric characteristics of the wing with modifications to section B.

A drawing of the tank tested at the tip of the wing with sections A and B, along with the ordinates defining the tank shape, is presented in figure 3. Also shown are the small $\left(\frac{A_f}{A_t} = 0.0675\right)$ and the large $\left(\frac{A_f}{A_t} = 0.232\right)$ tank stabilizing fins. Photographs of the tip tank on the wing are presented in figure 4. The lift and pitching moment of the tank in the presence of the tip of the wing with section A were measured at the 40-percent-tip-chord point which was considered representative of the elastic axis of a flexible wing by a two-element strain-gage beam (see fig. 4(a)) that was the supporting link between the wing tip and the tank.

TESTS

Tests were made in the Langley high-speed 7- by 10-foot tunnel over a Mach number range that generally extended from 0.60 to 0.90 at angles of attack from -2° to 8°. Wing section B with modification 1 was tested over an extended Mach range (from 0.20 to 0.90). Lift, drag, and pitching-moment coefficients were measured for the wing with sections A and B without tank and with the tank (gap open and sealed) at the wing tip; for section B with two modifications to the wing section including roughness over the wing leading edge extending aft 10 percent chord; and for two sizes of horizontal tank stabilizing fins on the wing with section A. Lift and pitching moment of the tip tank in the presence of the wing with section A were obtained for the tank alone and for the tank with fins.

Test Mach numbers were obtained from a calibration of the air velocity on the boundary-layer plate without a model in place. A survey in the plane of the model span showed that the spanwise Mach number gradient was negligibly small.

The test results were not corrected for jet-boundary effects because the tunnel test section was very large compared to the size of the test models. For this reason blockage effects of the models on the dynamic pressure were also negligible. The effect of the support fairing and boundary-layer plate on blocking was accounted for in the calibration of air velocity. The choking Mach number of the test section in this investigation was considerably higher than the highest test Mach number.

The test Reynolds number over a Mach number range from 0.20 to 0.90 is presented in figure 5. The solid curve represents the mean Reynolds number with the range of departures from the mean, occasioned by atmospheric conditions, represented by the cross-hatched region.

RESULTS

The results of the investigation are presented in the following figures:

	Figure
Basic force data:	
Wing with section A	6
Effect of tip tank	6(a)
Effect of tip-tank fins	6(b)
Wing with section B	7
Effect of tip tank	7(a)
Effect of modifications to section B	7(b)
Forces on tip tank in the presence of wing with section A . .	8
Lift-drag ratios:	
Effect of tip tank, section A	9
Effect of tip tank and modifications, section B	10
Effect of Mach number on the aerodynamic characteristics:	
Wing with section A and tip tank	11
Wing with section B, tip tank and trailing-edge modifications	12

The coefficients of force and moment presented in this paper are based on the area of the basic wing plan form except for the results of section B with modifications where the coefficients are based on the modified wing area. (See table I.) The projected area of the tip tank or the tank fins was not included in the area of the model for tank-on tests. Pitching-moment coefficients presented herein are presented about the quarter-chord point of the mean aerodynamic chord of the unmodified wing plan form.

The lift and pitching-moment coefficients of the tank in the presence of the wing with section A are based on the area of the original wing plan form of the wing with section A with the moments being presented about the 40-percent-chord point of the tip chord.

The slope of pitching-moment coefficient as a function of lift coefficient at constant Mach number $\left(\partial C_m/\partial C_L\right)_M$ and lift coefficient as a function of angle of attack at constant Mach number $\left(\partial C_L/\partial \alpha\right)_M$ were generally measured through $C_L = 0$. Where nonlinearities of the curves occurred at zero lift, average slopes were taken at $C_L = 0.1$ over a range that generally extended from $C_L \approx 0$ to $C_L \approx 0.2$.

The drag coefficients presented herein include the drag of the wing end plate.

DISCUSSION

Effect of Original Airfoil Sections

The parameter $\left(\partial C_m/\partial C_L\right)_M$ is a measure of the aerodynamic-center location relative to the quarter-chord point of the mean aerodynamic chord. At the lowest Mach number tested, M = 0.60, the aerodynamic center of the wing with section B (section similar to NACA 66_1-212) is approximately 7.5 percent forward of the aerodynamic center of the wing with section A (NACA 65-210) (figs. 11 and 12). As the Mach number increases, the aerodynamic center of section B moves farther forward while the aerodynamic center of section A remains relatively constant to M = 0.85, whereupon it moves sharply aft. At M = 0.85 the aerodynamic center of section B is about 16.5 percent ahead of the aerodynamic center of section A or about 14 percent ahead of the quarter-chord point.

A preliminary examination of the pitching-moment characteristics of a number of airfoil sections made in reference 2 revealed that airfoil sections with large trailing-edge angles had aerodynamic-center locations considerably forward of those with small trailing-edge angles. It is to be noted that section B, which has an aerodynamic center forward of that of section A, has a trailing-edge angle approximately 2.5 times greater than section A.

The drag coefficient of section B is about 0.0020 higher over the Mach number range than that of section A at zero lift, and the drag break Mach number is slightly lower (figs. 11 and 12). As the lift coefficient is increased, the drag of section B increases more rapidly than does that of section A. (See figs. 6 and 7.)

The poorer drag characteristics of the wing with section B are reflected directly in the lift-drag ratios. It is seen that section B has an $(L/D)_{max}$ approximately 10 percent lower than that of section A (figs. 11 and 12).

The lift-curve slope of section B is lower than that of section A with the reduction generally increasing as the Mach number is increased until $\left(\partial C_L/\partial \alpha\right)_M$ of section B is only about 65 percent of $\left(\partial C_L/\partial \alpha\right)_M$ of section A at the highest Mach number investigated ($M = 0.90$). (See figs. 11 and 12.)

It is cautioned, however, that a quantitative application of these data to similar profiles at larger scale is attended by some risk because of the susceptibility of the separation phenomenon involved in Reynolds number effects.

Effect of Modifications to Section B

In an effort to move the aerodynamic center of section B as far aft as possible and still maintain a practical airfoil section, two modifications designed to decrease the trailing-edge angle were made to the aft part of the original airfoil section. The largest rearward movement produced by either of the modifications was of the order of 2 percent mean aerodynamic chord at Mach numbers below force break. Both modifications were effective, however, in producing a normal rearward movement of the aerodynamic center with Mach number above force break (fig. 12(a)).

Modifying the trailing edge of section B resulted in notable increases in drag coefficient, particularly at the high lift coefficients (fig. 7(b)).

The effect of extending the Mach number range to $M = 0.20$ and thus lowering the test Reynolds number and, in addition, adding leading-edge roughness to section B with modification 1 (accentuated cusp trailing edge) is included in these data (fig. 12(a)). Reduction of the test Reynolds number results in a rearward movement of the aerodynamic center of 2 percent, but leading-edge roughness has a small effect on the aerodynamic-center location. Leading-edge roughness does, however, produce a large increase in drag coefficient.

Effect of Tip Tank

The maximum change in the aerodynamic-center location of sections A and B caused by adding a tip tank with tank gap open or sealed is a forward movement of about 1.5 percent mean aerodynamic chord below force break (figs. 11(a) and 12(b)).

The drag characteristics of the wing-tank combination with tank gap open and either airfoil section at zero lift coefficient as a function of Mach number show that the tank lowers the force-break Mach number about 0.02, and, at the force break M of the wing-tank combination ($M = 0.77$), the drag contribution of the tip tank is about 48 percent of the drag of the wing with section A and about 38 percent of the drag of the wing with section B. At the lowest test Mach number ($M = 0.60$) the drag increment of the tip tank is, in coefficient form, about 0.0030. Below force-break Mach number sealing the tank gap does not have any appreciable effect on the drag characteristics of the installation at zero lift of the model. The difference in force-break characteristics shown for the tank on the tip of the wing with sections A and B may be attributed to juncture effects.

The increase in the effective aspect ratio of the wing produced by the end-plate effect of the tip tank (see reference 1) results in reduced drag coefficients at the higher lift coefficients. The reduction is such that the drag added by the tip tank is largely negated at lift coefficients of about 0.4 to 0.5 at the lower Mach numbers (figs. 6(a) and 7(a)) with the most effective end-plate action and hence the lowest drag coefficients being produced with the tank gap sealed.

The importance of sealing the tank gap is illustrated in figures 9, 10, 11(a), and 12(b) by the large increases in $(L/D)_{max}$ that are obtained, particularly on section A.

Because of the increased effective aspect ratio, the lift-curve slope of the wing with both sections and the tip tank was on an average of 12 percent higher than the lift-curve slope of the wing alone. Sealing the tank gap increased $\left(\partial C_L/\partial\alpha\right)_M$ at the lower Mach numbers, but this increase is less than the contributions of the basic tip tank.

The results of tests of two sizes of horizontal stabilizing fins on the tank on the wing with section A show that the small fin $\left(\frac{A_f}{A_t} = 0.0675\right)$ moves the aerodynamic center of the wing-tank combination aft about 2.5 percent, while the large fin $\left(\frac{A_f}{A_t} = 0.232\right)$, which has approximately 2.5 times more area than the small fin, moves the aerodynamic center aft about 5 percent (fig. 11(b)). Thus, per unit area the greatest stabilizing influence is exerted by a horizontal fin whose chord is large compared to the span. However, the use of horizontal fins of this type on tip tanks may prove costly to the performance of the airplane because of the flow separation over the fin in the rotational field at the wing tip, and, consequently higher drag. It is cautioned, however, that results involving the phenomenon of flow separation, particularly flow separation from a low-aspect-ratio flat plate such as the horizontal fins, are susceptible to Reynolds number effects. Hence, similarly large drag increases may not be observed at larger scales.

Characteristics of the Tip Tank in the Presence of the Wing

The tip tank without horizontal fins is unstable about the 40-percent-tip-chord point of the wing with section A (fig. 8). The 40-percent-tip-chord point is considered representative of the location of the elastic axis of a flexible wing. To stabilize the tank, a horizontal fin of about 23 percent of the projected area of the tank is required. The nonlinearity of the tank pitching moment with the large fin is in substantial agreement with hypothesis of flow separation over the horizontal fins. Because of the magnitude of the coefficients involved, a more exact definition of the lift-coefficient range over which the horizontal fins are subject to flow separation may be obtained from the tank pitching-moment coefficients. Below a wing lift coefficient of about 0.10 the stabilizing influence of the large horizontal fin is largely negated by separation. Flow separation from the small horizontal fin is less severe, and seems to occur at a somewhat higher lift coefficient.

CONCLUSIONS

Analysis of the results of an experimental investigation of the effect of two wing sections and a tip tank on the aerodynamic characteristics of a semispan unswept wing of aspect ratio 5.16 and taper ratio 0.61 at high subsonic speeds indicates that:

1. Below force-break Mach number the wing with a section similar to NACA 66_1-212 gave a 10 percent lower maximum lift-drag ratio, an appreciable lower lift-curve slope, and an aerodynamic-center location 7.5 percent farther forward than the wing with an NACA 65-210 section. Above force break the aerodynamic characteristics of the wing with a section similar to an NACA 66_1-212 section compared even less favorably with those of the wing with NACA 65-210 section.

2. Two trailing-edge modifications, designed to reduce the trailing-edge angle of the section which was similar to the NACA 66_1-212 section, moved the aerodynamic center of the wing appreciably rearward particularly above force break.

3. Locating a tank at the tip of the wing resulted in a forward movement of the aerodynamic center of the wing of less than 1.5 percent, reduced the Mach number for force break slightly, and at zero lift resulted in a 48-percent increase in drag coefficient of the wing alone with the NACA 65-210 section and a 38-percent increase in the drag coefficient of the wing with a section similar to the NACA 66_1-212 section.

4. The increase in effective aspect ratio produced by the end-plate effect of the tip tank was appreciably larger when the gap between the tank and wing was sealed.

5. A horizontal tank fin which was 23 percent of the projected area of the tank neutralized the unstable pitching moment of the tank about the 40-percent-tip-chord point of the wing with the NACA 65-210 section.

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1. Hanson, Fredrick H., Jr.: The Effect of a Wing-Tip-Mounted Fuel Tank on the Aerodynamic Characteristics of a High-Speed Bomber Wing. NACA ACR 5H06, 1945.

2. Polhamus, Edward C.: Preliminary Correlation of the Effect of Compressibility on the Location of the Section Aerodynamic Center at Subcritical Speeds. NACA RM L8D14, 1948.

TABLE I

PERTINENT GEOMETRIC CHARACTERISTICS OF THE WING PLAN FORMS INVESTIGATED

[All dimensions are in feet and square feet]

Geometry	Original plan form	Modified plan form	
		Modification 1	Modification 2
$\bar{c}$	0.270	0.279	0.282
c_{tip}	0.200	0.210	0.211
c_{root}	0.330	0.338	0.342
S/2	0.182	0.187	0.19
A	5.16	5.01	4.88
λ	0.61	0.62	0.62
ϕ(deg)	17.37	8.32	9.08

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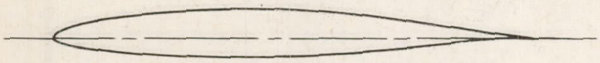
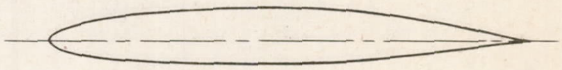
Table II

Ordinates of the Original Airfoil Sections





Wing section A (NACA 65-210) Wing section B

(Stations and ordinates in percent of wing chord)

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.435	.819	.565	-.719
.678	.999	.822	-.859
1.169	1.273	1.331	-1.059
2.408	1.757	2.592	-1.385
4.898	2.491	5.102	-1.859
7.394	3.069	7.606	-2.221
9.894	3.555	10.106	-2.521
14.899	4.338	15.101	-2.992
19.909	4.938	20.091	-3.346
24.921	5.397	25.079	-3.607
29.936	5.732	30.064	-3.788
34.951	5.954	35.049	-3.894
39.968	6.067	40.032	-3.925
44.984	6.058	45.016	-3.868
50.000	5.915	50.000	-3.709
55.014	5.625	54.986	-3.435
60.027	5.217	59.973	-3.075
65.036	4.712	64.964	-2.652
70.043	4.128	69.957	-2.184
75.045	3.479	74.955	-1.689
80.044	2.783	79.956	-1.191
85.038	2.057	84.962	-.711
90.028	1.327	89.972	-.293
95.014	.622	94.986	.010
100.000	0	100.000	0
L.E. radius: 0.687. Slope of radius thru L.E.: 0.084.			
$\phi = 7.00°$			

Station	Ordinate	
	Upper	Lower
0	0	0
.75	1.193	-1.005
1.25	1.538	-1.283
2.5	2.161	-1.793
5.0	3.023	-2.488
7.5	3.678	-2.962
10.0	4.212	-3.337
15.0	5.022	-3.923
20.0	5.625	-4.345
25.0	6.108	-4.630
30.0	6.465	-4.845
35.0	6.712	-4.983
40.0	6.855	-5.063
45.0	6.918	-5.078
50.0	6.884	-5.020
55.0	6.738	-4.875
60.0	6.463	-4.633
65.0	6.037	-4.272
70.0	5.492	-3.785
75.0	4.794	-3.203
80.0	3.970	-2.543
85.0	3.028	-1.847
90.0	2.027	-1.152
95.0	.990	-.540
100.0	0	0
L.E. radius: 0.800. Slope of radius thru L.E.: 0.055		
$\phi = 17.37°$		

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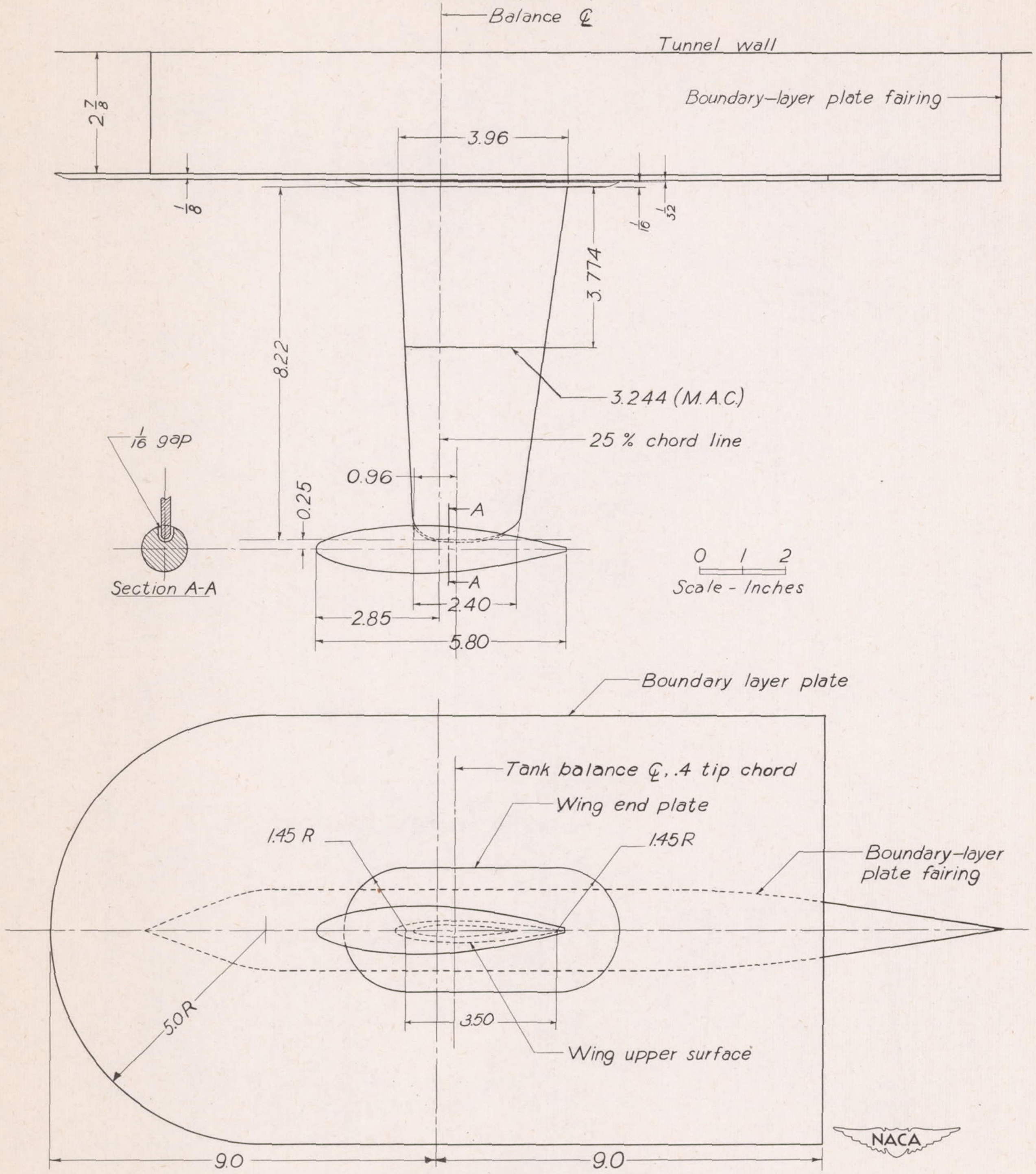

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Figure 1 - Drawing of test models with tip tank attached.

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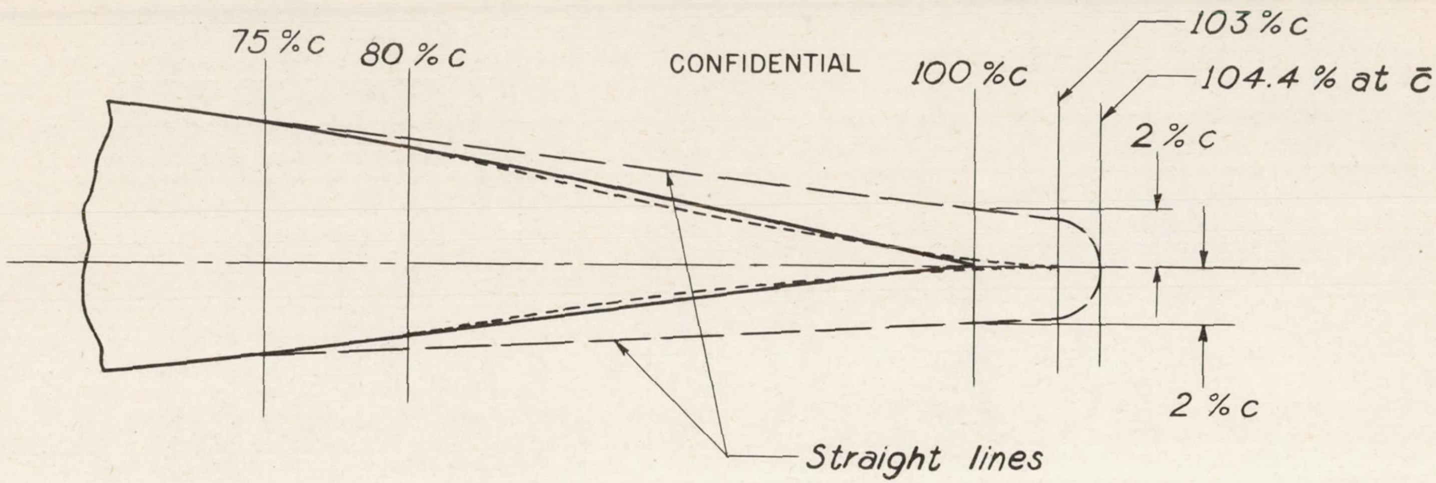

Ordinates for Mod. 1		
Station, % c	Ordinate, % c	
	Upper	Lower
75	4.794	-3.203
80	3.97	-2.543
82.5	3.48	-2.17
85	2.97	-1.78
87.5	2.43	-1.42
90	1.90	-1.06
92.5	1.39	-0.75
95	0.95	-0.50
97.5	0.54	-0.26
100	0.23	-0.12
103	0	0

	ϕ
———— Original section B	17.37°
------- Mod. 1	8.32°
—— —— Mod. 2	9.08°

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Figure 2.- Trailing-edge modifications tested on the wing with section B.

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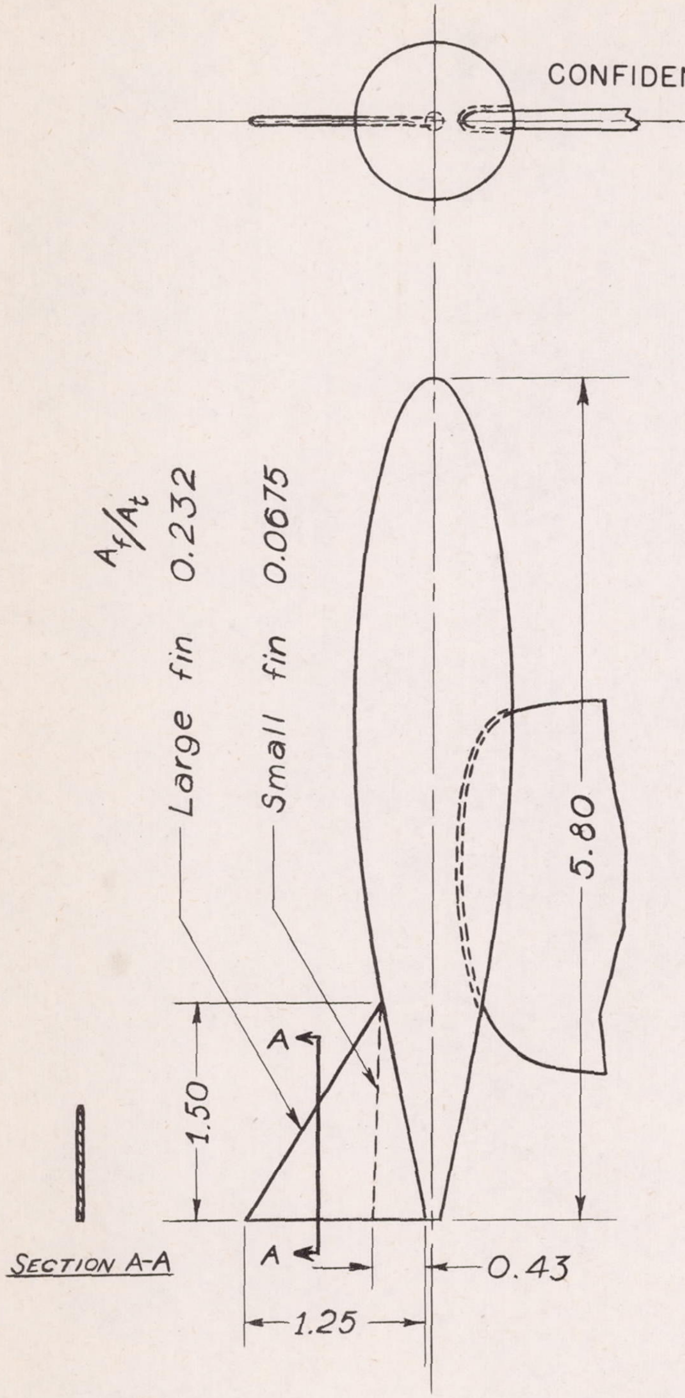

Tank ordinates (Percent tank length)	
Station	Radius
0	0
1.25	1.97
2.50	3.00
5.0	4.28
10.0	5.86
15.0	7.03
20.0	7.83
25.0	8.52
30.0	8.96
40.0	9.43
45.0	9.50
50.0	9.34
60.0	8.43
70.0	6.91
80.0	5.02
90.0	2.98
100.0	0.74
L.E. radius:	2.17

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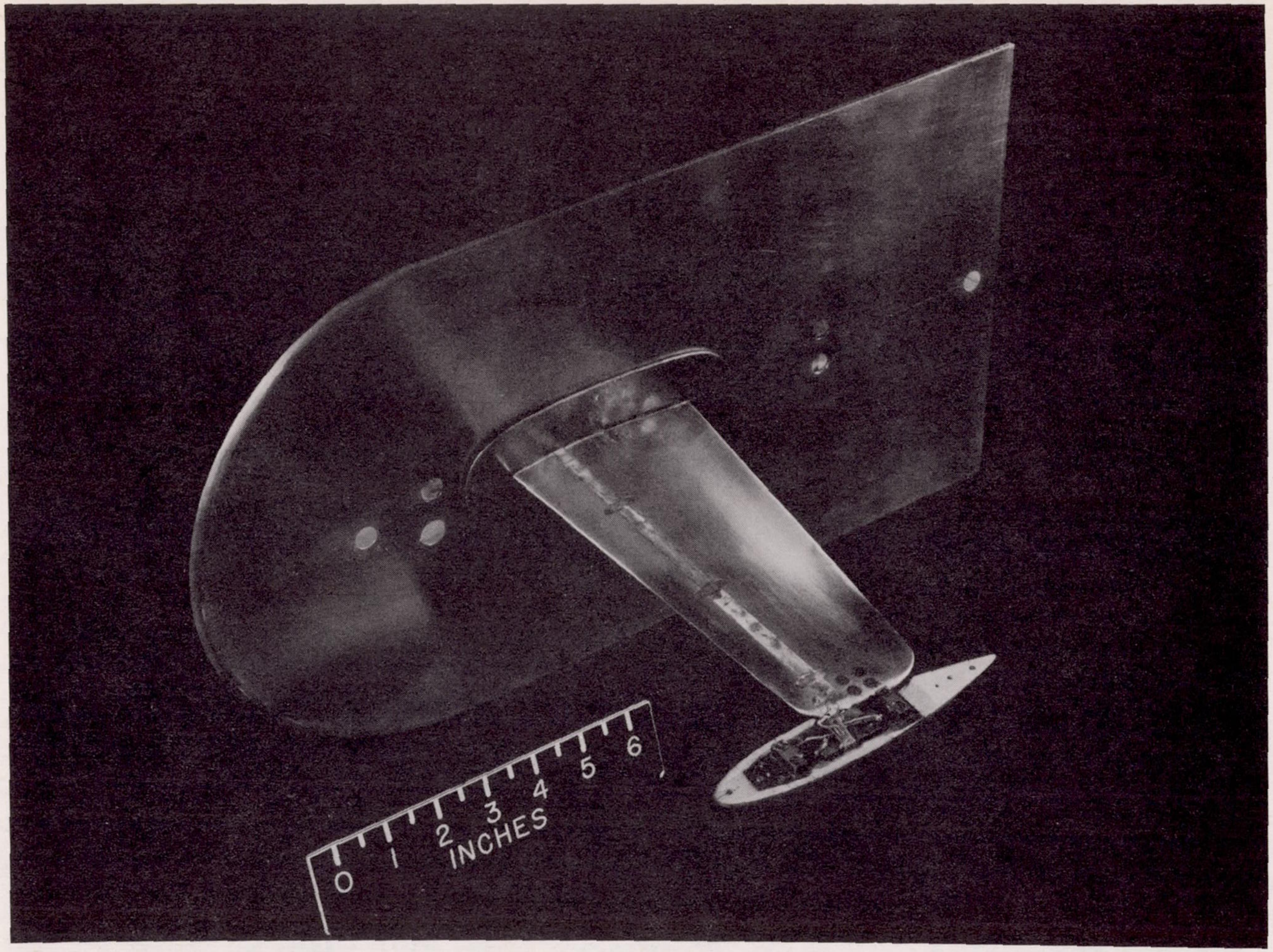
Figure 3.- Fuel tank and fins tested.



(a) Wing with section A.

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Figure 4.- Photograph of the test model with the tip-tank mounted on the boundary-layer plate.

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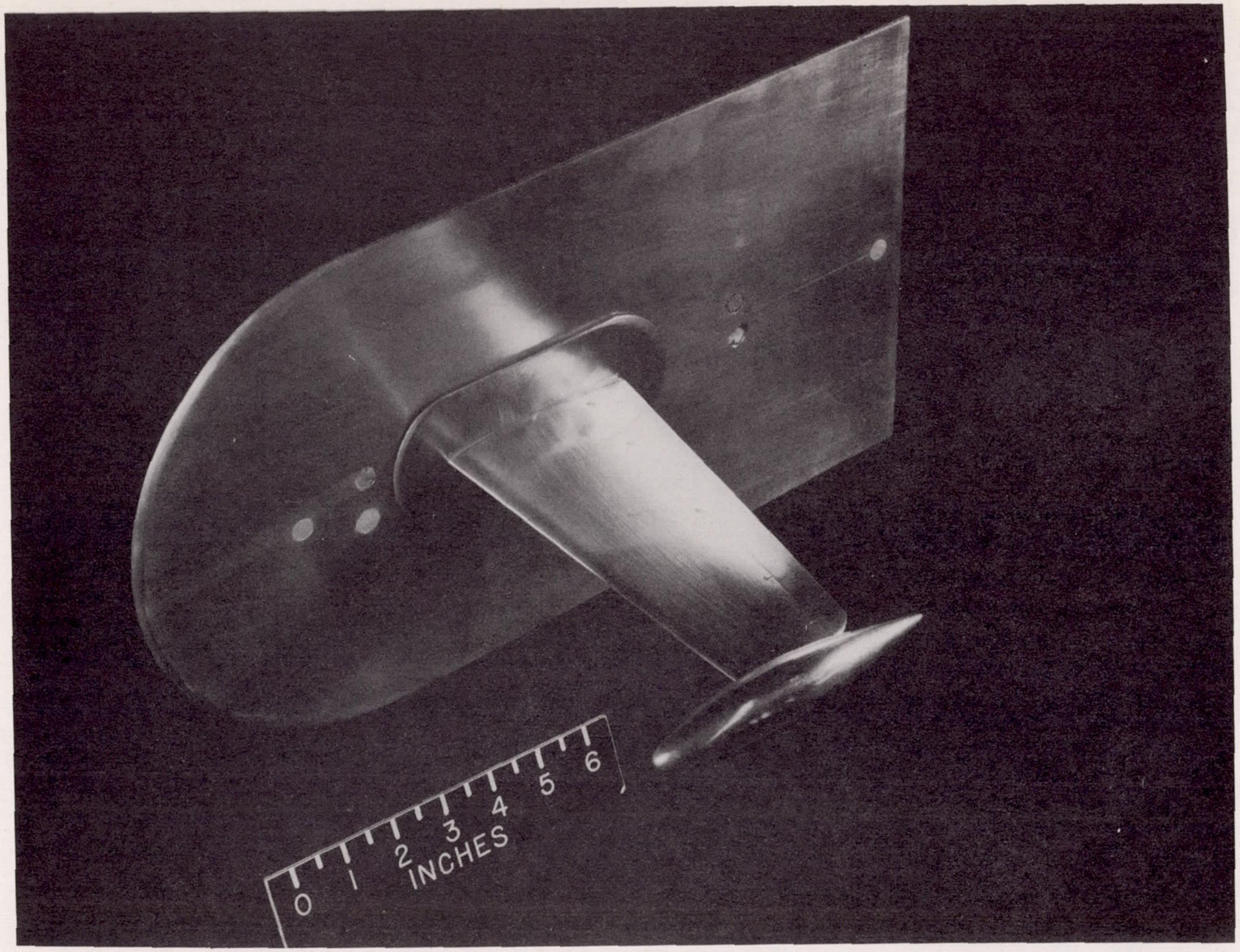

(b) Wing with section B.

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Figure 4.- Concluded.

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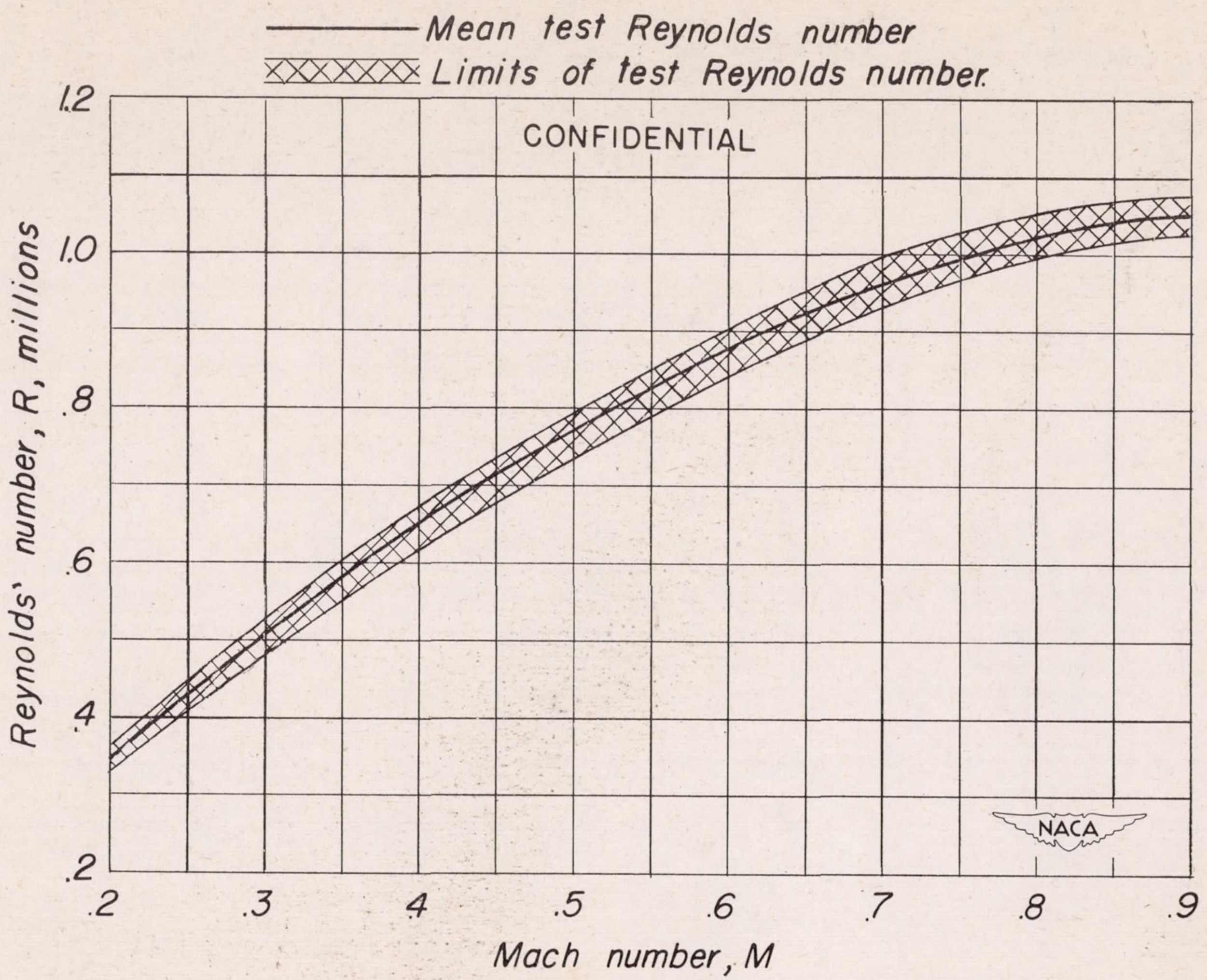

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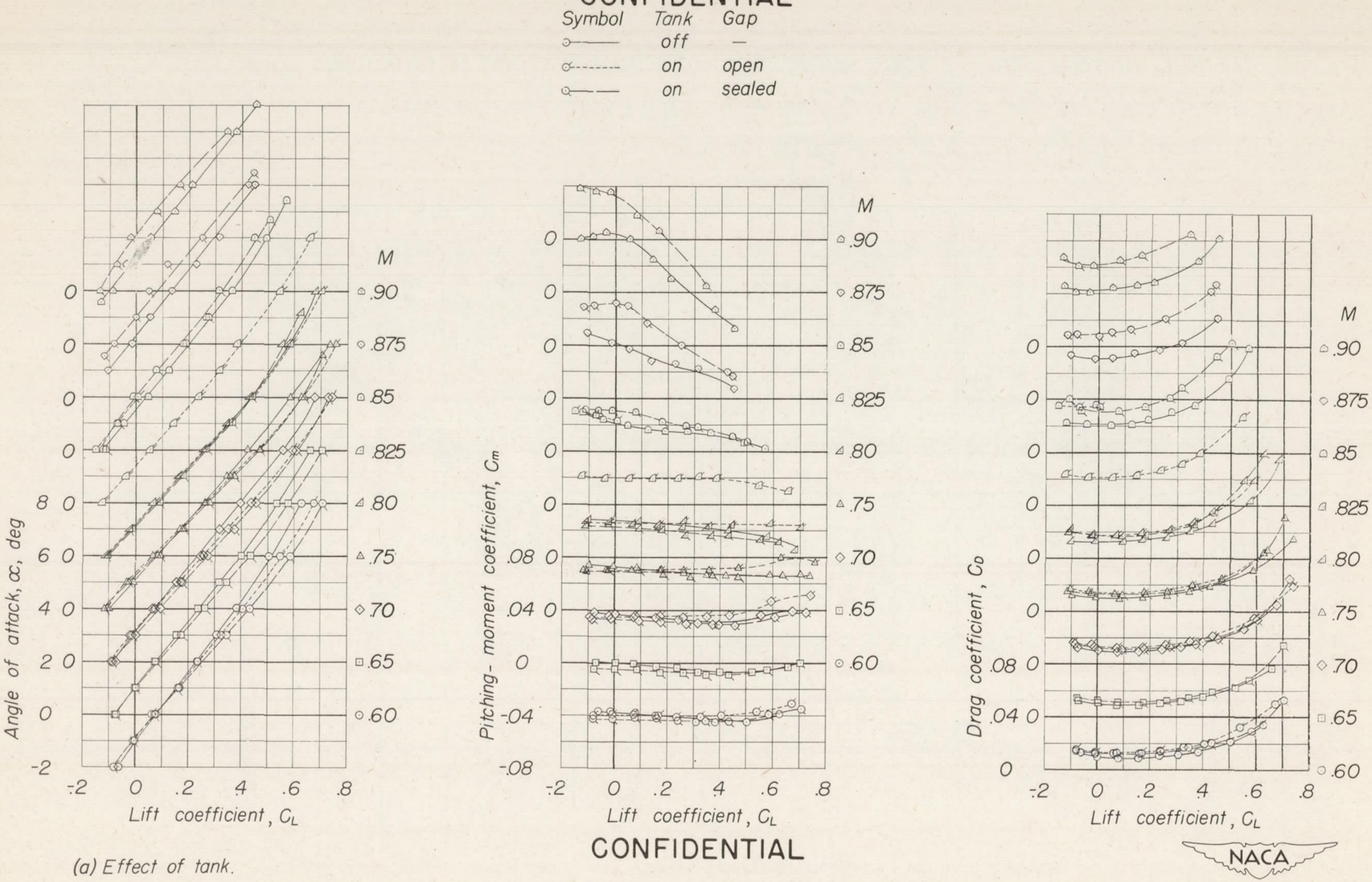
Figure 5.— Variation of Reynolds number with Mach number for wing with sections A and B in the Langley high-speed 7- by 10-foot tunnel.



(a) Effect of tank.

Figure 6.— Effect of a tip-mounted auxiliary fuel tank installation on the aerodynamic characteristics of the wing with section A.

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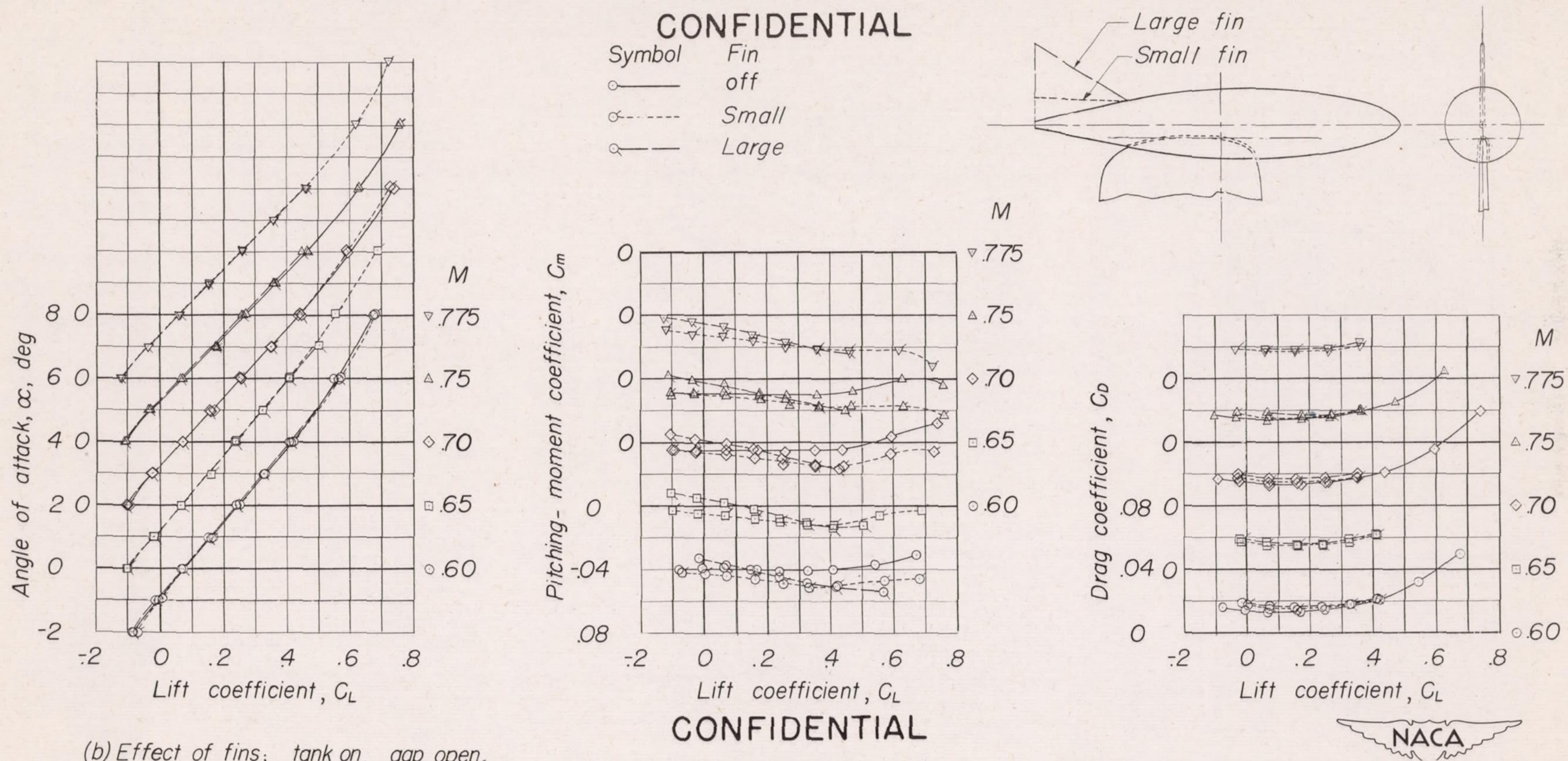

(b) Effect of fins; tank on, gap open.

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Figure 6 — Concluded.

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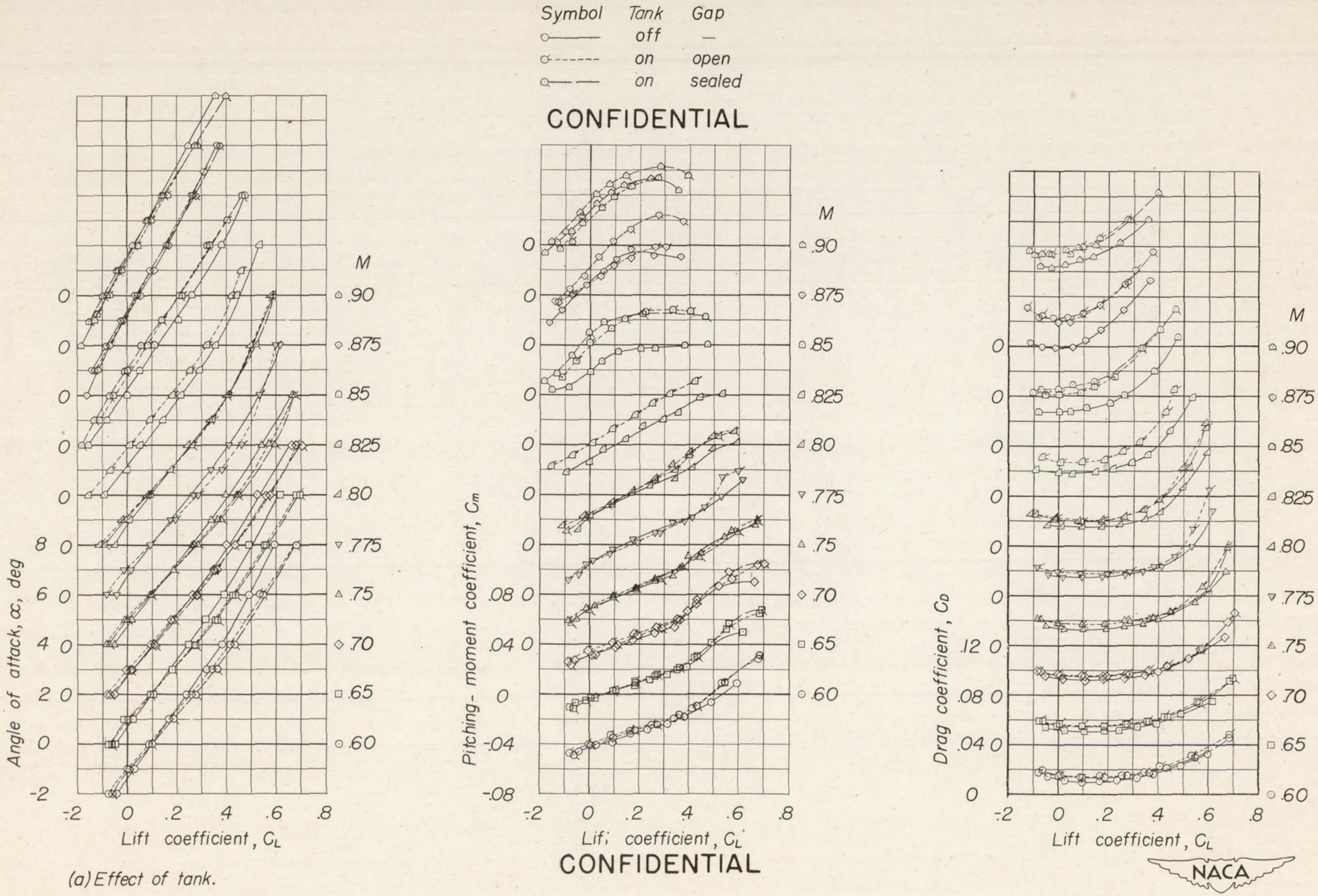

(a) Effect of tank.

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Figure 7.— Effect of a tip-mounted auxiliary fuel tank and trailing edge modifications on the aerodynamic characteristics of the wing with section B.

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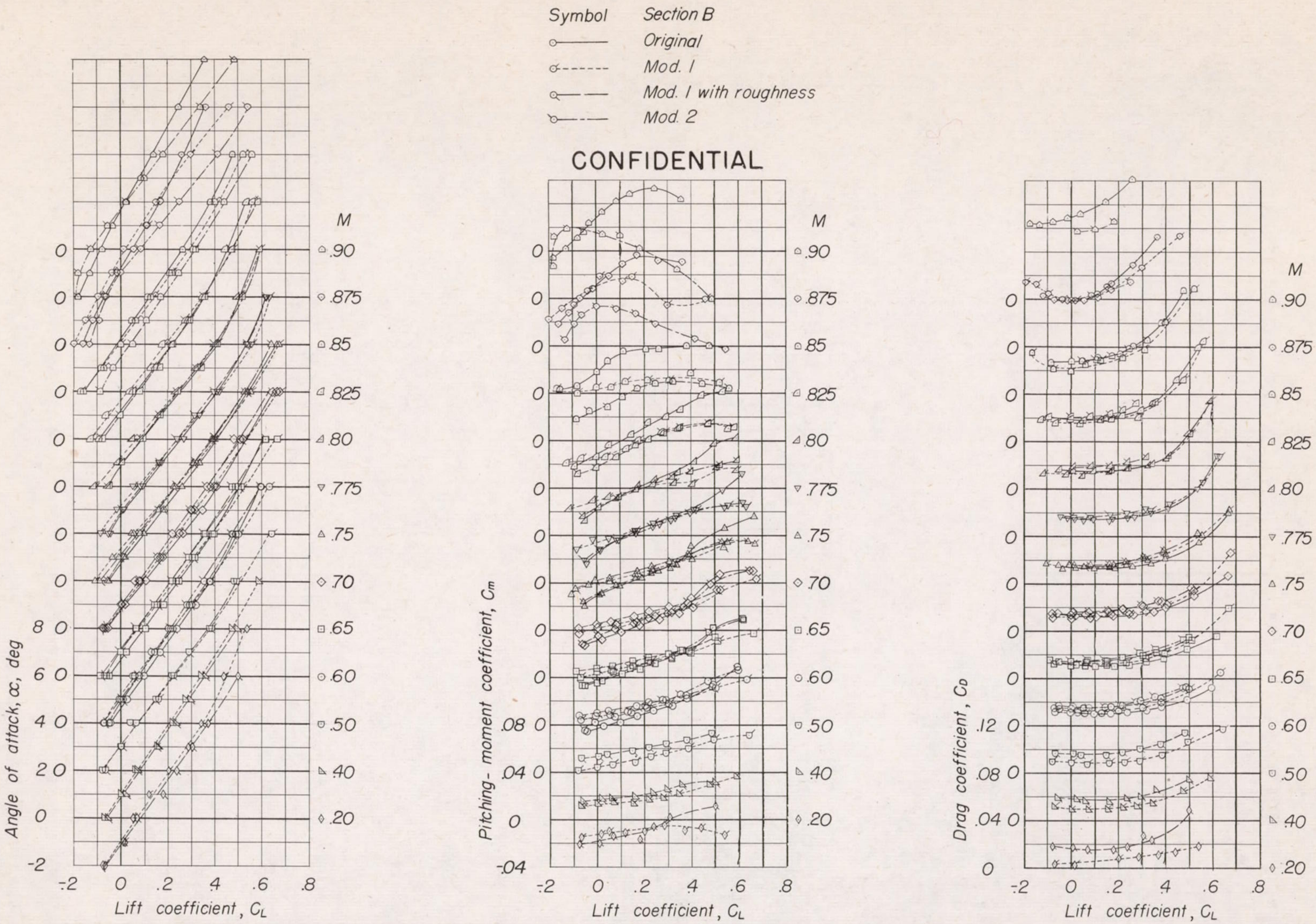

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(b) Effect of trailing edge modifications.

Figure 7. - Concluded

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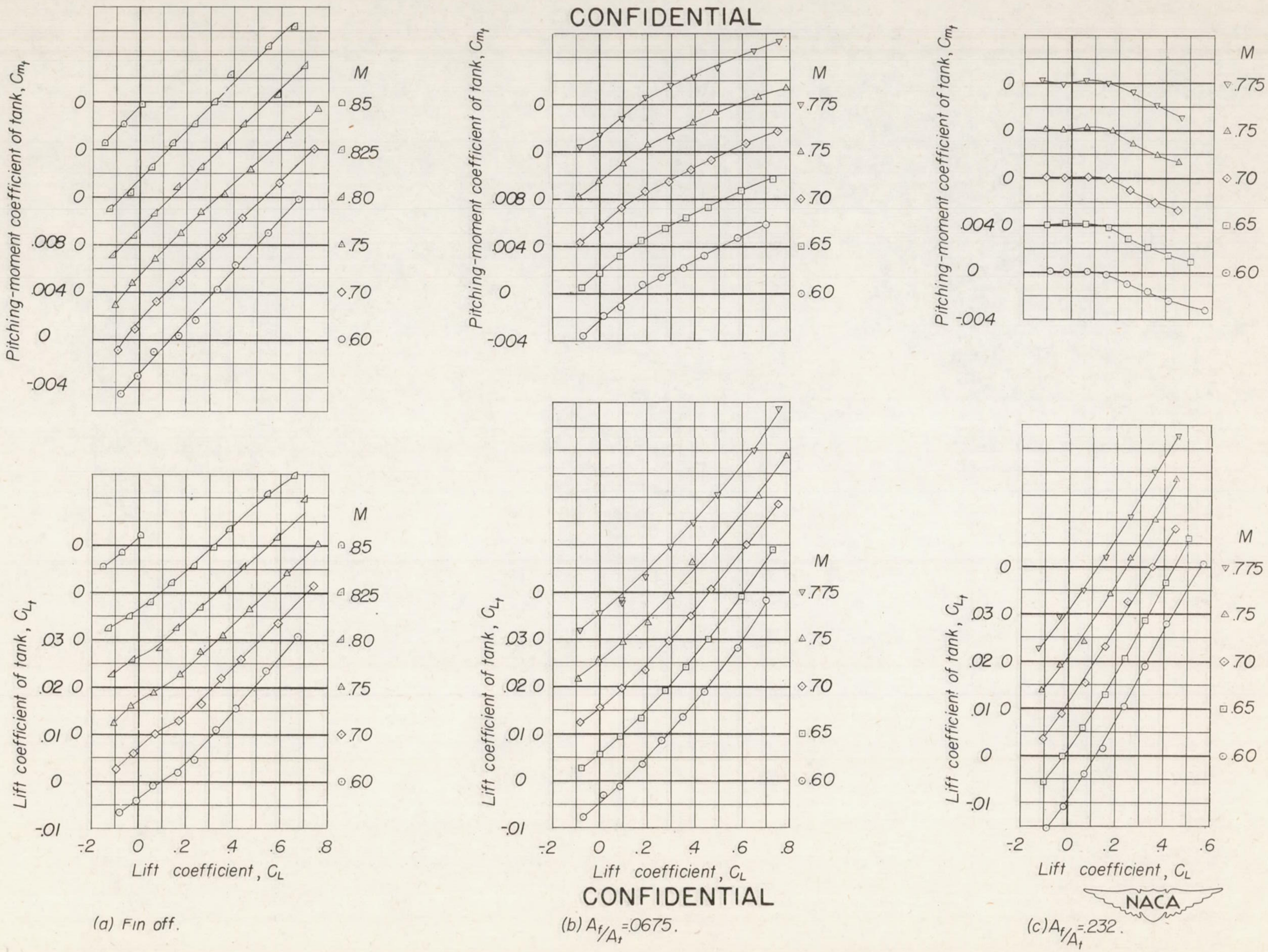

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(a) Fin off. (b) A_f/A_t=.0675. (c) A_f/A_t=.232.

Figure 8. — Aerodynamic characteristics of a model of an auxiliary fuel tank mounted at the tip of the wing with section A, gap open.

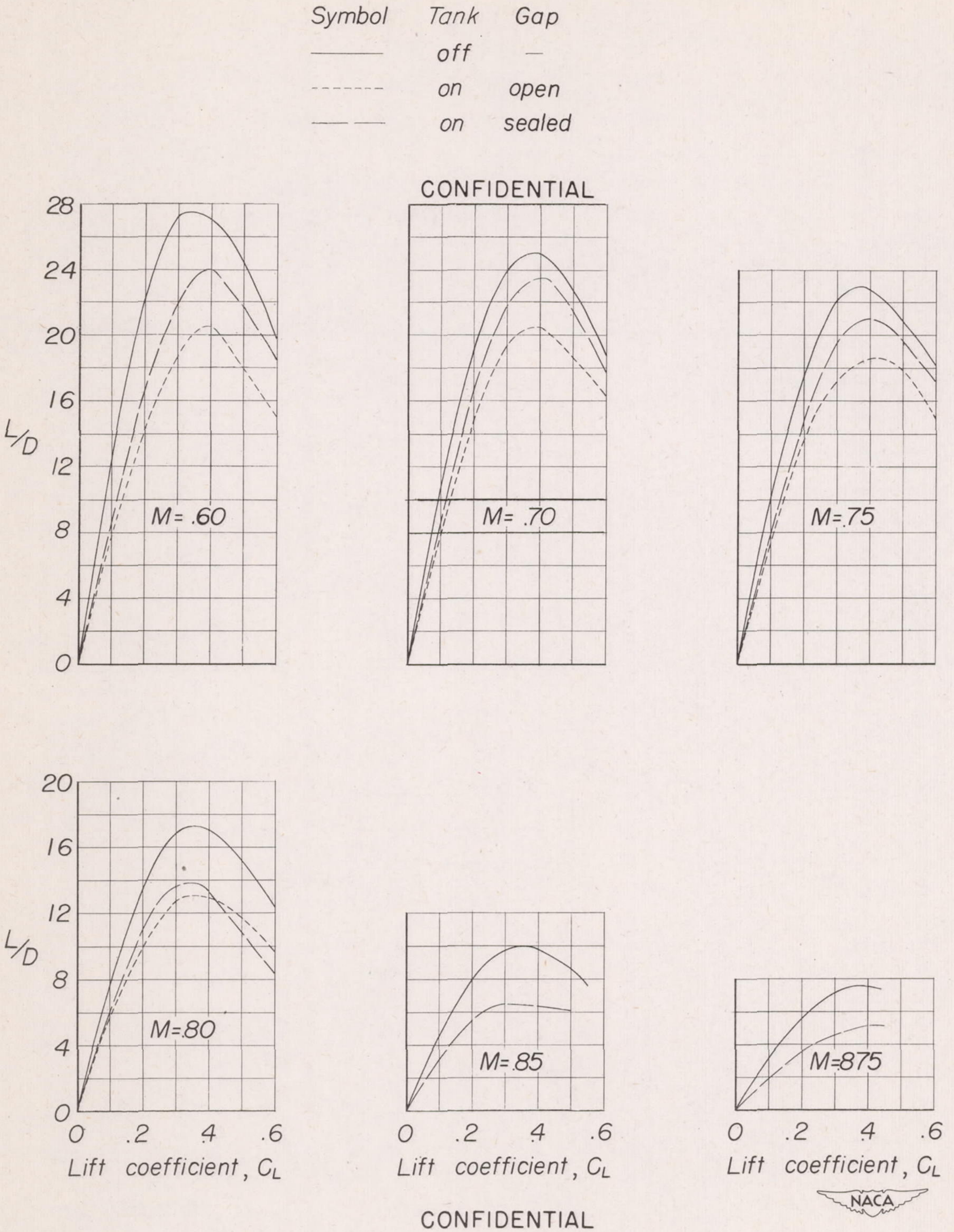

Figure 9.—Effect of a tip-mounted auxiliary fuel tank on the lift-drag ratio of the wing with section A.

Symbol	Section B	Tank	Gap
———	Original	off	—
--------	Original	on	open
— —	Original	on	sealed
— - —	Mod. I	off	—

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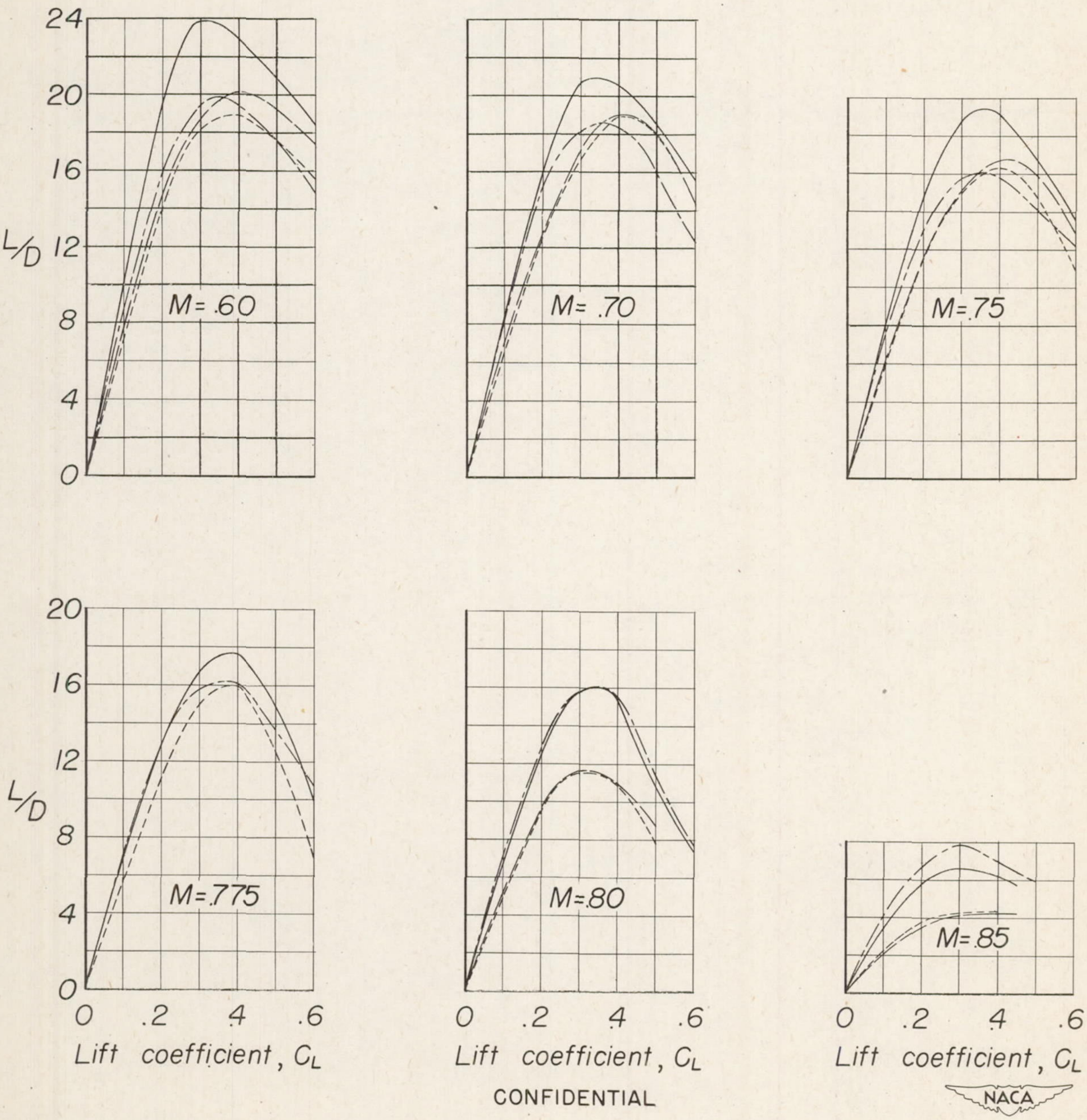

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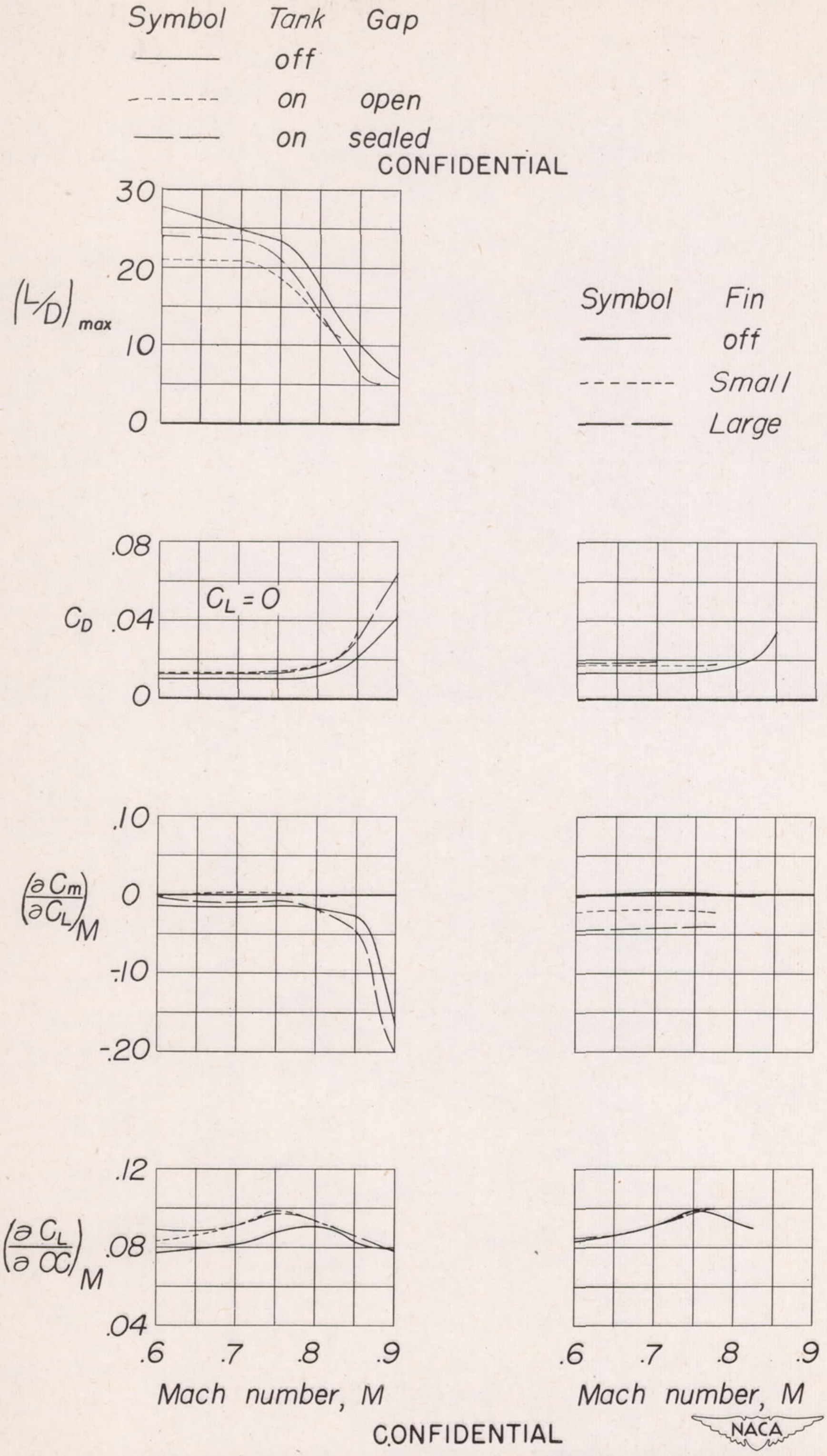
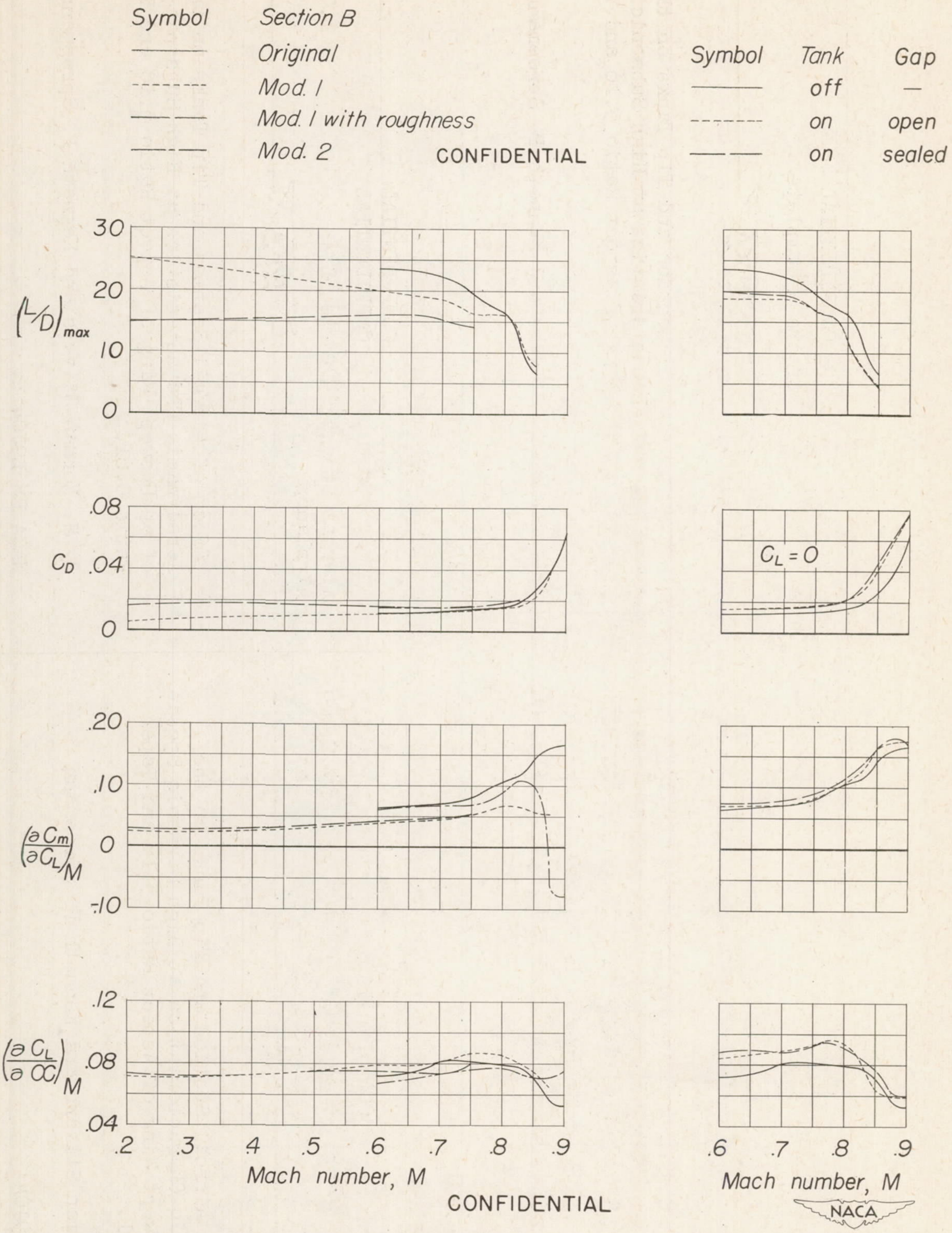
Figure 10.—Effect of a tip-mounted auxiliary fuel tank and a trailing-edge modification on the lift-drag ratio of the wing with section B.



(a) Effect of tank.

(b) Effect of fins; tank on, gap open.

Figure 11.– Effect of Mach number on the aerodynamic characteristics of the wing with section A and a tip-mounted auxiliary fuel tank installation.



(a) Effect of trailing edge modifications.

(b) Effect of tank.

Figure 12.—Effect of Mach number on the aerodynamic characteristics of the wing with section B and a tip-mounted auxiliary fuel tank and trailing edge modifications.