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NACA

RESEARCH MEMORANDUM

AERODYNAMIC CHARACTERISTICS AT SUPERSONIC SPEEDS OF A SERIES OF WING-BODY COMBINATIONS HAVING CAMBERED WINGS WITH AN ASPECT RATIO OF 3.5 AND A TAPER RATIO OF 0.2

EFFECTS OF SWEEP ANGLE AND THICKNESS RATIO ON THE AERODYNAMIC CHARACTERISTICS IN PITCH AT $M = 2.01$

By Ross B. Robinson

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

An investigation has been conducted in the Langley 4- by 4-foot supersonic pressure tunnel at a Mach number of 2.01 and a Reynolds number of 2.2×10^6, based on the wing mean aerodynamic chord, to determine the effects of sweep and thickness on the aerodynamic characteristics in pitch of a series of wing-body combinations having cambered wings with an aspect ratio of 3.5 and a taper ratio of 0.2. The wings, tested on a slender body of revolution, had quarter-chord sweep angles of 10.8°, 35°, and 47° for a thickness ratio of 4 percent, and thickness ratios of 4, 6, and 9 percent for a quarter-chord sweep angle of 47°. In addition, a 47° swept wing with a thickened root section (12 percent thick at the body center line tapering to 6 percent thick at the 40-percent semispan station) was investigated. A summary of the results of the investigation of this wing series at M = 1.60 is included for comparison with the results of the present tests at M = 2.01.

The results of this investigation indicate that, in general, for this range of thickness ratios and sweep angles, the effects of thickness are greater than the effects of sweep angle on the aerodynamic characteristics in pitch. A maximum lift-drag ratio of 6.40 was obtained for the 47° swept, 4-percent-thick wing. There appeared to be little change in minimum drag between M = 1.60 and M = 2.01. As would be expected from theory, the values of the lift-curve slopes of the wings at M = 2.01 are about 75 percent as large as the M = 1.60 values.

INTRODUCTION

A research program has been in progress at the Langley Aeronautical Laboratory to determine at subsonic, transonic, and supersonic speeds the effects of sweep and thickness on the aerodynamic characteristics of a series of wing-body combinations with cambered wings having an aspect ratio of 3.5 and a taper ratio of 0.2. The effects of thickness and of sweep on the aerodynamic characteristics in pitch of the wing series at subsonic and transonic speeds are presented in references 1 and 2, respectively, and at a Mach number of 1.60 in reference 3. The effects of sweep and thickness on the lateral stability characteristics of the wing series at a Mach number of 1.60 are presented in reference 4 and a Mach number of 2.01 in reference 5. The results of tests at Mach numbers of 1.60 and 2.01 of several nacelle configurations on the 6-percent-thick 47° swept wing are given in references 6 and 7, respectively.

The present paper presents the results of tests to determine the effects of sweep and thickness on the aerodynamic characteristics in pitch of this series of wings at a Mach number of 2.01 and a Reynolds number of 2.2×10^6 based on the mean aerodynamic chord. The wings had quarter-chord sweep angles of 10.8°, 35°, and 47° for a thickness ratio of 4 percent and thickness ratios of 4, 6, and 9 percent for a sweep angle of 47°. The effects of the addition of a horizontal canard surface to the 6-percent-thick 47° swept-wing configuration were investigated. A thickened-root wing of 47° sweep, having a thickness ratio of 12 percent at the root, tapering to 6 percent at the 40-percent semispan station, and remaining constant at 6 percent further outboard was also investigated. These results are presented without analyses to expedite publication.

SYMBOLS

C_L lift coefficient of wing-body combination, Lift/qS

C_D drag coefficient of wing-body combination, Drag/qS

C_m pitching-moment coefficient of wing-body combination about 0.25 mean aerodynamic chord, Pitching moment/$qS\bar{c}$

C_{L_f} lift coefficient of body, Lift/qA

C_{D_f} drag coefficient of body, Drag/qA

C_{m_f}	pitching-moment coefficient of body, Pitching moment/qAl
A	maximum cross-sectional area of body, 0.0276 sq ft
S	wing area including body intercept, 1.143 sq ft
$\bar{c}$	wing mean aerodynamic chord, ft
l	body length, ft
q	free-stream dynamic pressure, lb/sq ft
M	Mach number
t/c	streamwise wing-thickness ratio
L/D	lift-drag ratio
C_{L_α}	lift-curve slope
$C_{m_{C_L}}$	rate of change of pitching-moment coefficient with lift coefficient
$\Delta C_D/C_L^2$	drag rise factor
α	angle of attack of body center line, deg
Λ	sweep angle of wing quarter-chord line, deg

Subscripts:

max	maximum
min	minimum

APPARATUS AND MODELS

The tests were conducted in the Langley 4- by 4-foot supersonic pressure tunnel described in reference 3. The models used in these tests were composed of an ogive-cylinder body and various midwing configurations with a ratio of body diameter to wing span of about 0.094. The wings were positioned so that the quarter-chord point of the mean aerodynamic chord was always at the same body station. The wing airfoil sections had an NACA 65A-series thickness distribution and mean-line ordinates one-third of NACA 230 series plus an (a = 1) mean line for $C_L = 0.1$. The airfoil coordinates are given in table I. Details of the models are shown in figure 1.

The models were sting supported and had a six-component internal strain-gage balance in the body. The model and sting are shown in figure 2. Figure 3 is a photograph of the model in the tunnel. The models, balance, and indicating system were furnished by a U.S. Air Force contractor.

TESTS

Test Conditions and Procedure

The conditions for the tests of the wing-body configuration were:

Mach number	2.01
Reynolds number, based on wing mean aerodynamic chord . . .	2.2×10^6
Stagnation dew point, °F	<-30
Stagnation pressure, lb/sq in.	14
Stagnation temperature, °F	110

In order to establish an indication of the type of boundary layer existing over the basic body, the body alone was tested through a pressure range of about 4 pounds per square inch to 14 pounds per square inch corresponding to a Reynolds number range of 2.1 to 7.1×10^6 (based on body length). All the other test conditions remained unchanged.

Calibration of the nozzle prior to these tests has shown that the flow in the test section is reasonably uniform. The magnitudes of the variations in the flow parameters are summarized as follows:

Mach number	±0.015
Flow angle in horizontal plane, deg	±0.1
Flow angle in vertical plane, deg	±0.1

Tests of all of the configurations were made through an angle-of-attack range from -2° to 13°.

Corrections and Accuracy

The angle of attack of the model was corrected for deflection of the balance and support system due to lift and pitching moment. Angle corrections were obtained from in-place calibration of the balance for various lift loads and pitching moments. The validity of this method of correction was verified during the tests at $M = 1.60$ (ref. 3). The estimated error in angle of attack was $\pm 0.1^{\circ}$. During these tests, the model was yawed about -0.2° because of misalinement. No corrections were applied for this yaw angle or for the flow variations in the test section.

The estimated errors in the force data obtained by comparing the results of two tests of the same configuration are as follows:

C_L	±0.001
C_D	±0.001
C_m	±0.001

The base pressure was measured for all the configurations tested and the drag data were corrected to correspond to a base pressure equal to free-stream static pressure.

RESULTS

The results are presented with a minimum of analysis in order to expedite publication. In order to determine the type of boundary-layer flow over the model, the body alone was tested through a Reynolds number range of 2.1 to 7.1×10^6 (based on the body length) with and without a small transition strip near the nose of the body. The drag coefficients at zero lift for both configurations are presented in figure 4 as a function of Reynolds number. The results indicate that, at the highest Reynolds number obtainable, the boundary layer of the body without transition strip had not become completely turbulent. All further tests except one were made without the transition strip. The 6-percent-thick 47° swept-wing - body configuration was tested both with and without the transition strip on the body to investigate the effect of addition of the strip on the aerodynamic characteristics of the wing-body combination. All of the testing was done at a Reynolds number of 7.1×10^6 based on body length (2.2×10^6 based on the wing M.A.C.).

The experimental aerodynamic characteristics in pitch of the body alone, with and without the transition strip, and the theoretical values calculated by the method of reference 8 are presented in figure 5. Addition of the transition strip increased the drag about 30 percent and produced more lift at the higher angles of attack.

The aerodynamic characteristics in pitch of the 4-percent-thick wings in the sweep series are shown in figures 6(a) to 6(c) and of the 47° swept wings of the thickness series in figures 6(c) to 6(f). Tests of the 6-percent-thick 47° swept wing indicated that the addition of the transition strip to the body resulted in a slight increase in the drag of the wing-body combination because of the increased region of turbulent flow. The effects of the addition of a horizontal canard surface to the 6-percent-thick 47° swept-wing configuration are shown in figure 7.

The lift-drag ratios, as a function of lift coefficient for the wing series, are summarized in figures 8: the effect of the addition of the canard in figure 8(a), the effect of thickness in figure 8(b), and the effect of sweep in figure 8(c). The variation of the minimum drag coefficient with the square of the thickness ratio is presented in figure 9. Included for reference purposes on this figure is the drag coefficient of the body alone.

A summary of the variation of the aerodynamic characteristics in pitch with thickness ratio and sweep angle is presented in figure 10. Table II contains a summary of the longitudinal characteristics of this wing series at $M = 1.60$ (ref. 3) and $M = 2.01$. As would be expected, the values of $(L/D)_{max}$ and $C_{L_{\alpha}}$ decrease as the Mach number increases. The maximum value of L/D obtained at $M = 2.01$ was 6.40 for the 47° swept 4-percent-thick wing. The values of $C_{L_{\alpha}}$ at $M = 2.01$ were about 75 percent of the values obtained at $M = 1.60$. The variation of $C_{L_{\alpha}}$ with sweep angle (fig. 10) agrees closely with the variation predicted by theory (ref. 9). As the Mach number is increased, there is a slight decrease in $C_{m_{C_L}}$ and a corresponding forward movement of the aerodynamic center of the wing-body combination. Within the limitations of the accuracy of the measurements and the test techniques, there appeared to be little significant change in $C_{D_{min}}$ from $M = 1.60$ to $M = 2.01$.

In general, for this range of thickness ratios and sweep angles, the effects of thickness are larger than the effects of sweep angle on the aerodynamic characteristics in pitch of the wings.

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TABLE I

AIRFOIL COORDINATES FOR THE VARIOUS WINGS

[Thickness distribution: NACA 65A series; mean line ordinates: one-third of NACA 230 plus (a = 1) for $C_L = 0.1$]

$\frac{t}{c} = 0.04$

x/c	y/c	
	Upper surface	Lower surface
0	0	0
.5	.411	.245
.75	.499	.271
1.25	.665	.289
2.5	.962	.324
5.0	1.435	.367
7.5	1.776	.429
10	2.039	.472
15	2.423	.577
20	2.642	.682
25	2.800	.787
30	2.887	.892
35	2.983	.997
40	2.992	1.006
45	2.940	1.041
50	2.852	1.006
55	2.712	.945
60	2.511	.857
65	2.265	.761
70	1.986	.674
75	1.680	.577
80	1.356	.481
85	1.041	.385
90	.726	.289
95	.402	.201
100	.105	.105
Tangent point	80.00	60.00
L.E. radius = 0.0016c		

$\frac{t}{c} = 0.06$

x/c	y/c	
	Upper surface	Lower surface
0	0.061	0
.5	.577	.376
.75	.717	.446
1.25	.919	.534
2.5	1.304	.621
5.0	1.872	.761
7.5	2.318	.857
10	2.668	.980
15	3.150	1.269
20	3.482	1.496
25	3.701	1.697
30	3.858	1.846
35	3.946	1.960
40	3.981	2.021
45	3.937	2.030
50	3.823	1.977
55	3.613	1.872
60	3.342	1.697
65	3.018	1.487
70	2.651	1.277
75	2.231	1.059
80	1.785	.849
85	1.339	.639
90	.892	.420
95	.446	.210
100	0	0
L.E. radius = 0.0024c		

$\frac{t}{c} = 0.09$

x/c	y/c	
	Upper surface	Lower surface
0	0.156	0
.5	.846	.574
.75	1.021	.680
1.25	1.283	.846
2.5	1.789	1.069
5.0	2.537	1.400
7.5	3.111	1.662
10	3.577	1.896
15	4.244	2.352
20	4.705	2.751
25	5.045	3.052
30	5.288	3.276
35	5.415	3.441
40	5.473	3.529
45	5.424	3.519
50	5.249	3.422
55	4.967	3.208
60	4.579	2.916
65	4.102	2.566
70	3.568	2.197
75	2.975	1.837
80	2.382	1.468
85	1.789	1.098
90	1.186	.739
95	.593	.369
100	0	0
L.E. radius = 0.0056c		

Thickened root

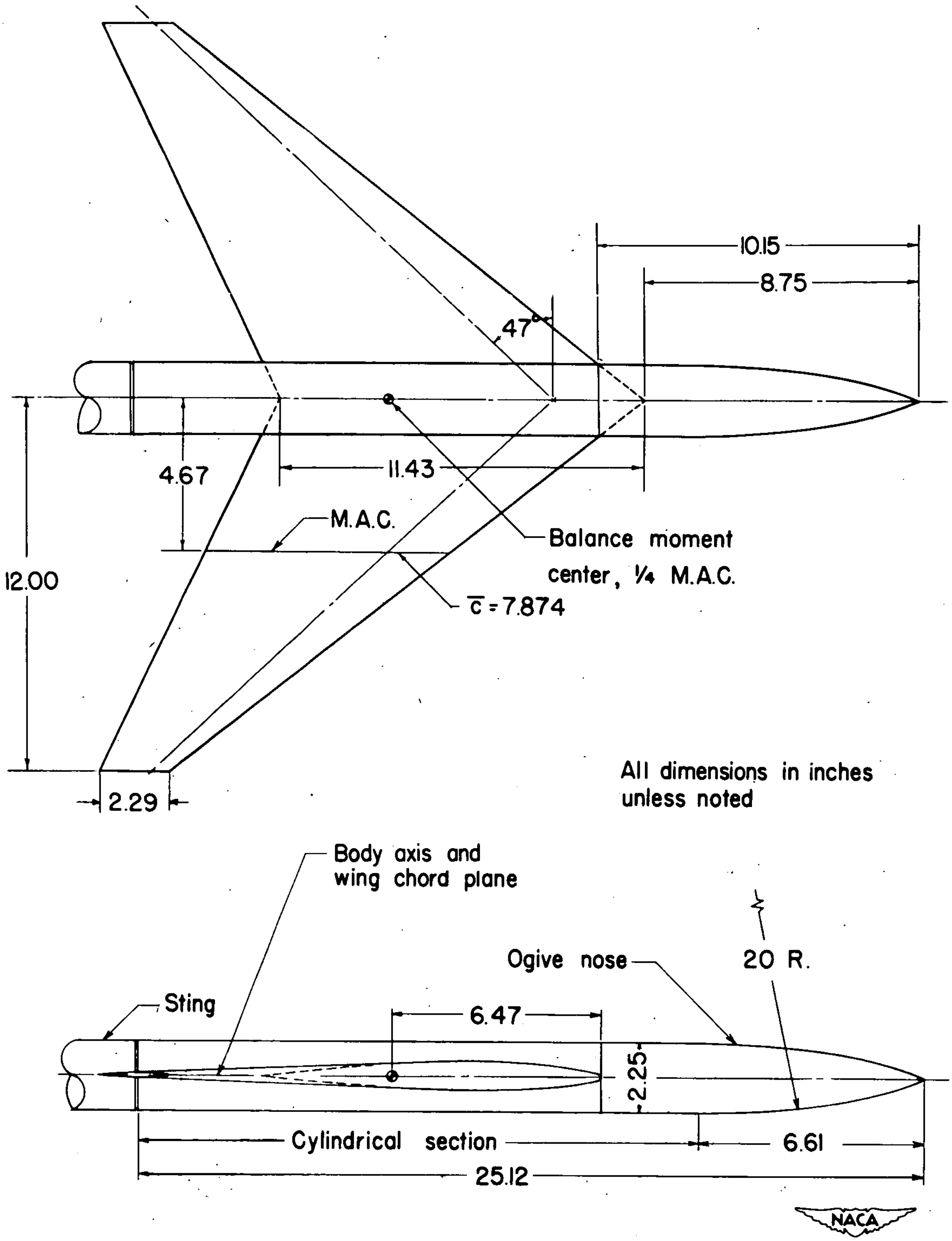
x/c	Root station	
	y/c	
	Upper surface	Lower surface
0	0.301	0
.5	1.120	.754
.75	1.335	.904
1.25	1.658	1.141
2.5	2.261	1.507
5.0	3.208	2.024
7.5	3.919	2.433
10	4.500	2.799
15	5.362	3.445
20	5.965	3.984
25	6.395	4.414
30	6.718	4.716
35	6.912	4.910
40	6.977	5.017
45	6.912	4.996
50	6.675	4.823
55	6.288	4.522
60	5.771	4.113
65	5.168	3.618
70	4.457	3.101
75	3.725	2.584
80	2.929	2.067
85	2.239	1.550
90	1.486	1.034
95	.732	.517
100	0	0
L.E. radius = 0.0099c		

TABLE II

SUMMARY OF THE LONGITUDINAL CHARACTERISTICS

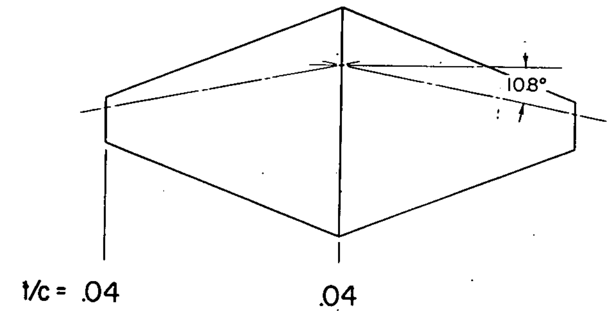
Λ deg	t/c	C_{L_α}	$C_{m_{C_L}}$	$C_{D_{min}}$	$\Delta C_{D_i}/C_L^2$	$(L/D)_{max}$	C_L for $(L/D)_{max}$	a.c.
(a) M = 1.60								
10.8	0.04	0.0525	-0.188	0.021	0.308	6.41	0.25	0.438
35	.04	.0535	-.230	.019	.308	6.97	.23	.480
47	.04	.053	-.258	.016	.288	7.65	.225	.508
47	.06	.052	-.259	.021	.310	6.28	.25	.509
[a]47	.06	.052	-.200	.022	.299	6.33	.27	.450
47	.09	.048	-.233	.0303	.330	5.10	.29	.483
47	.12, .06, .06	.050	-.260	.026	.308	5.71	.28	.510
Body alone		.0024	.770	.006	-----	----	-----	-.520
(b) M = 2.01								
10.8	0.04	0.0389	-0.170	0.0180	0.428	5.73	0.210	0.420
35	.04	.0388	-.204	.0170	.420	6.00	.200	.454
47	.04	.0400	-.236	.0150	.414	6.40	.190	.486
47	.06	.0388	-.230	.0200	.428	5.48	.210	.480
[b]47	.06	.0397	-.230	.0208	.435	5.32	.215	.480
[a]47	.06	.0401	-.190	.0205	.428	5.48	.210	.440
47	.09	.0375	-.207	.0290	.460	4.35	.255	.457
47	.12, .06, .06	.0381	-.240	.0245	.438	4.90	.240	.490
Body alone		.0013	1.46	.0037	-----	----	-----	-1.21
[c]Body alone		.0013	1.46	.0051	-----	----	-----	-1.21

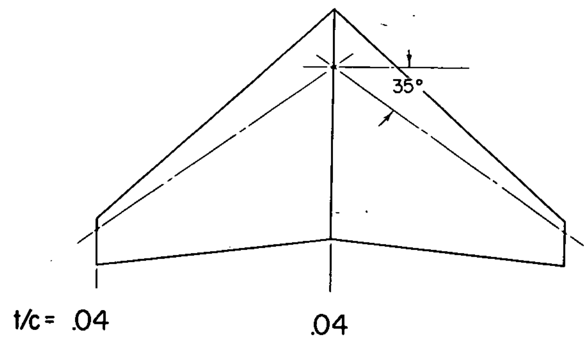
[a]Wing-body canard configuration.
[b]Wing-body with transition strip.
[c]Body with transition strip.



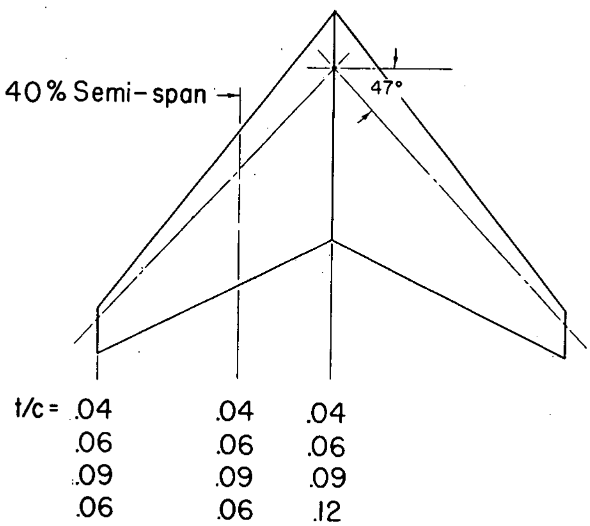
(a) Wing-body arrangement.

Figure 1.- Details of models. All dimensions in inches unless noted.



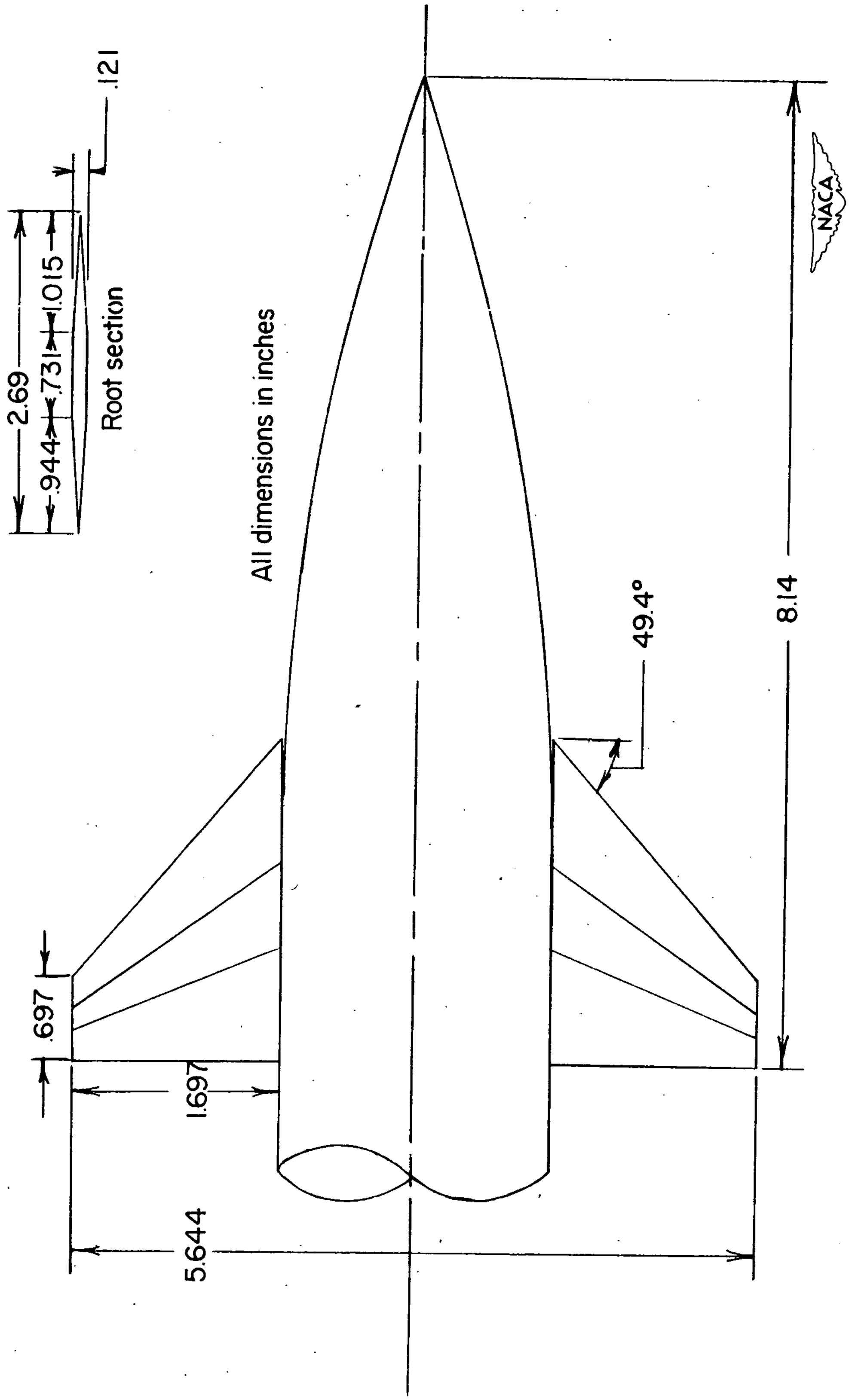


Aspect Ratio	3.5
Taper Ratio	0.2
Span, inches	24
Area, sq. feet	1.143



(b) Details of wings.

Figure 1.- Continued.



(c) Horizontal canard surface.

Figure 1.- Concluded.

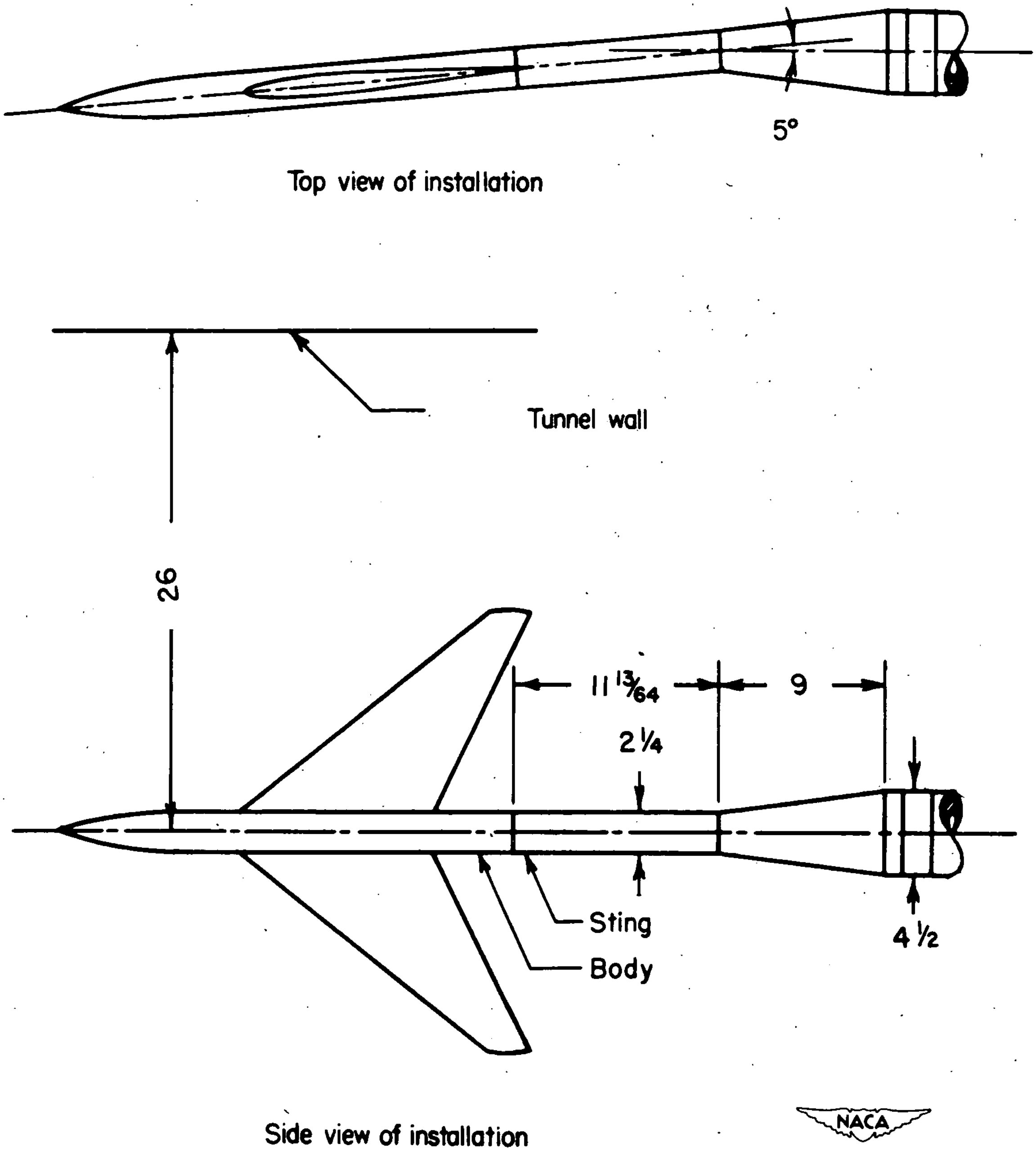


Figure 2.- Details of model sting support. All dimensions in inches unless otherwise noted.

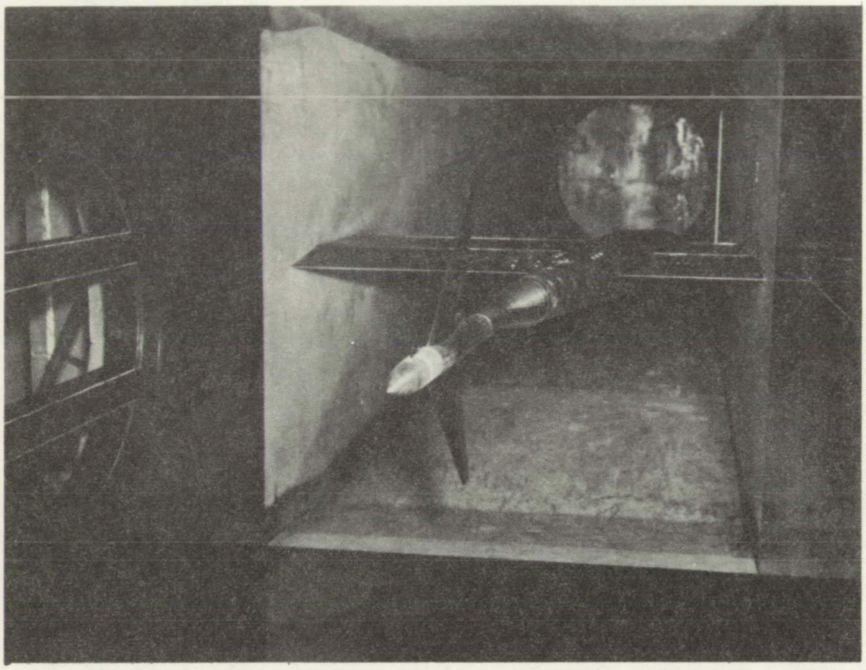


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Figure 3.- Model in Langley 4- by 4-foot supersonic pressure tunnel.

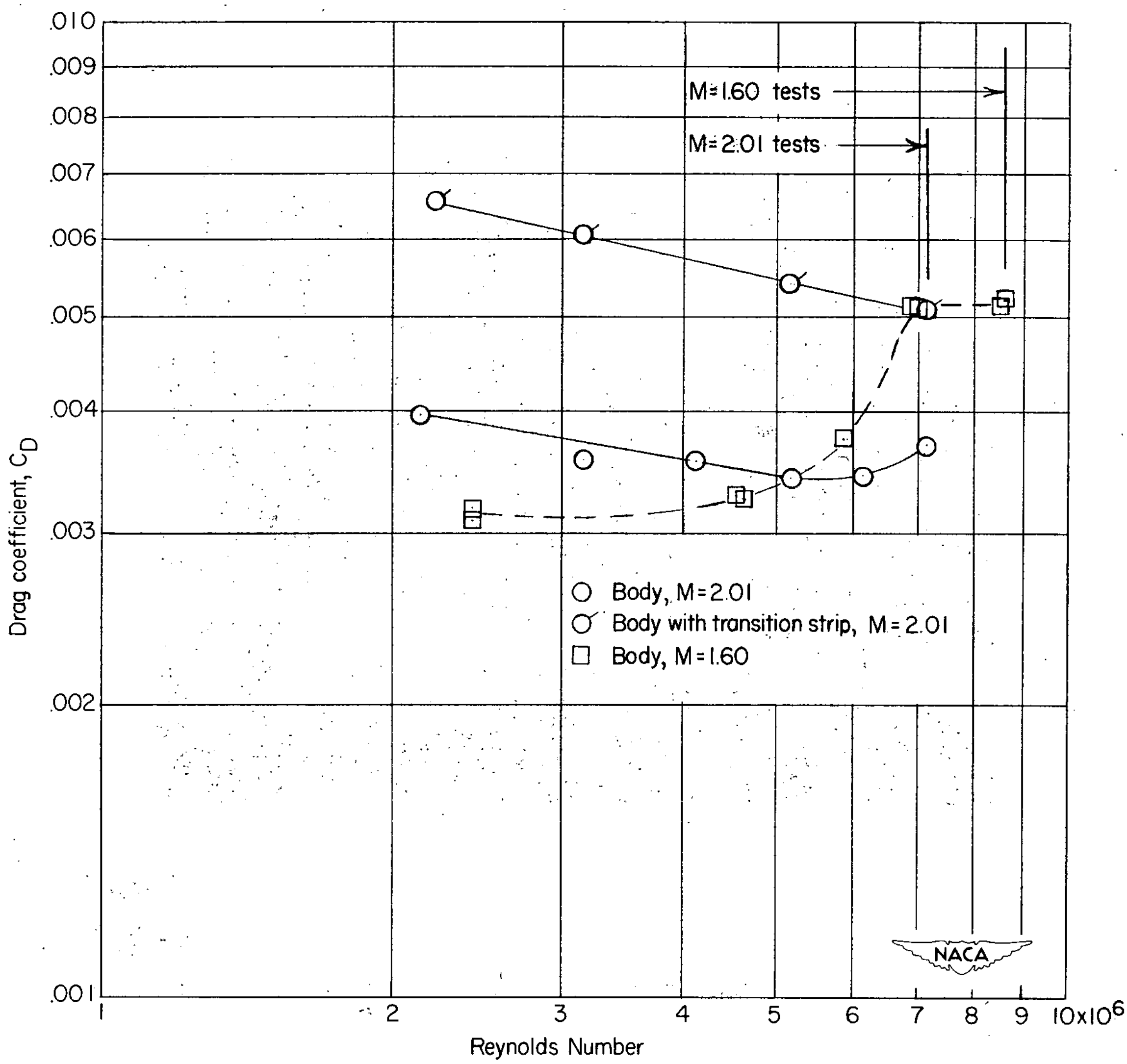


Figure 4.- Variation of body drag coefficient with Reynolds number based on body length. M = 1.60 and 2.01.

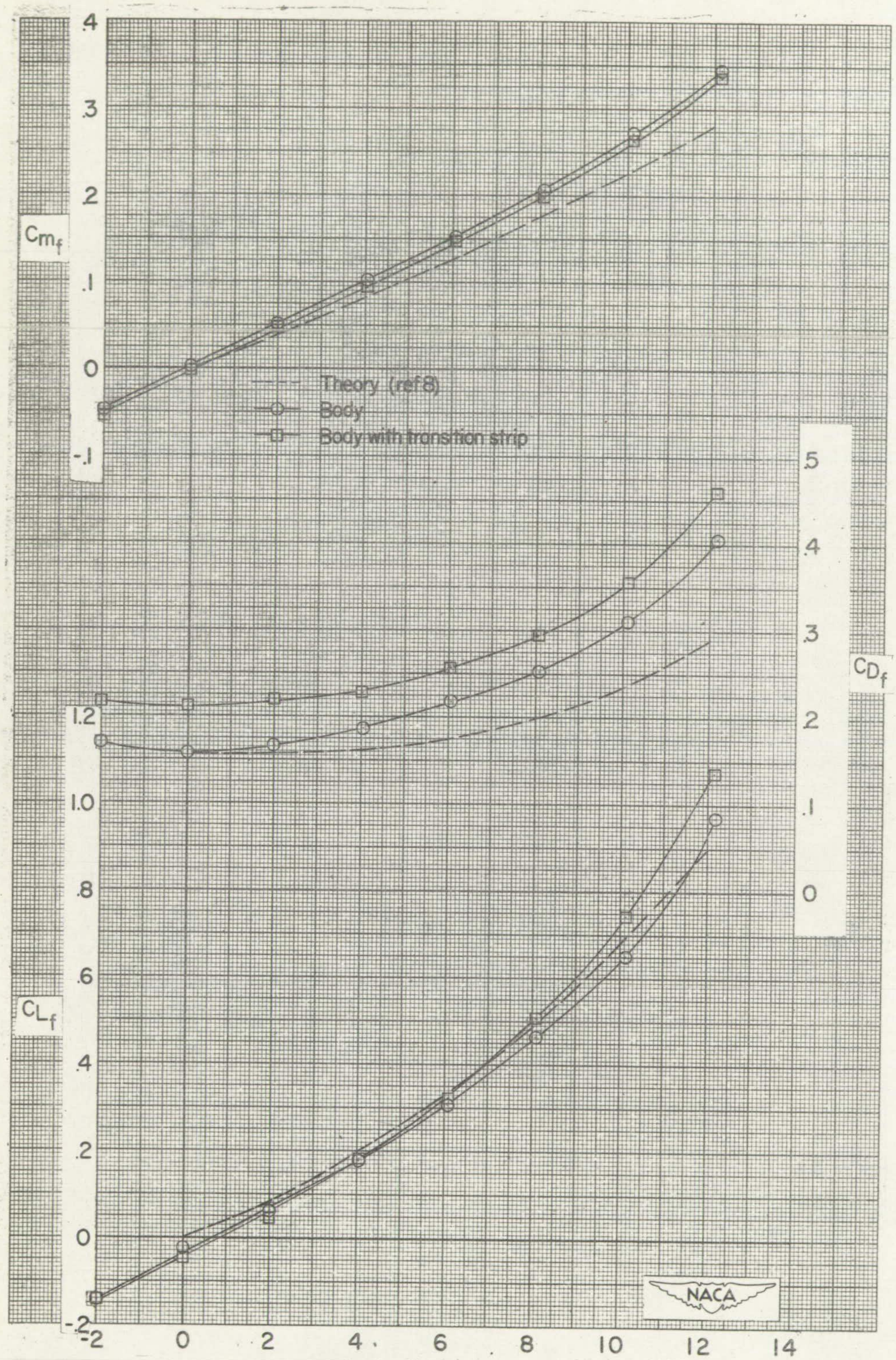
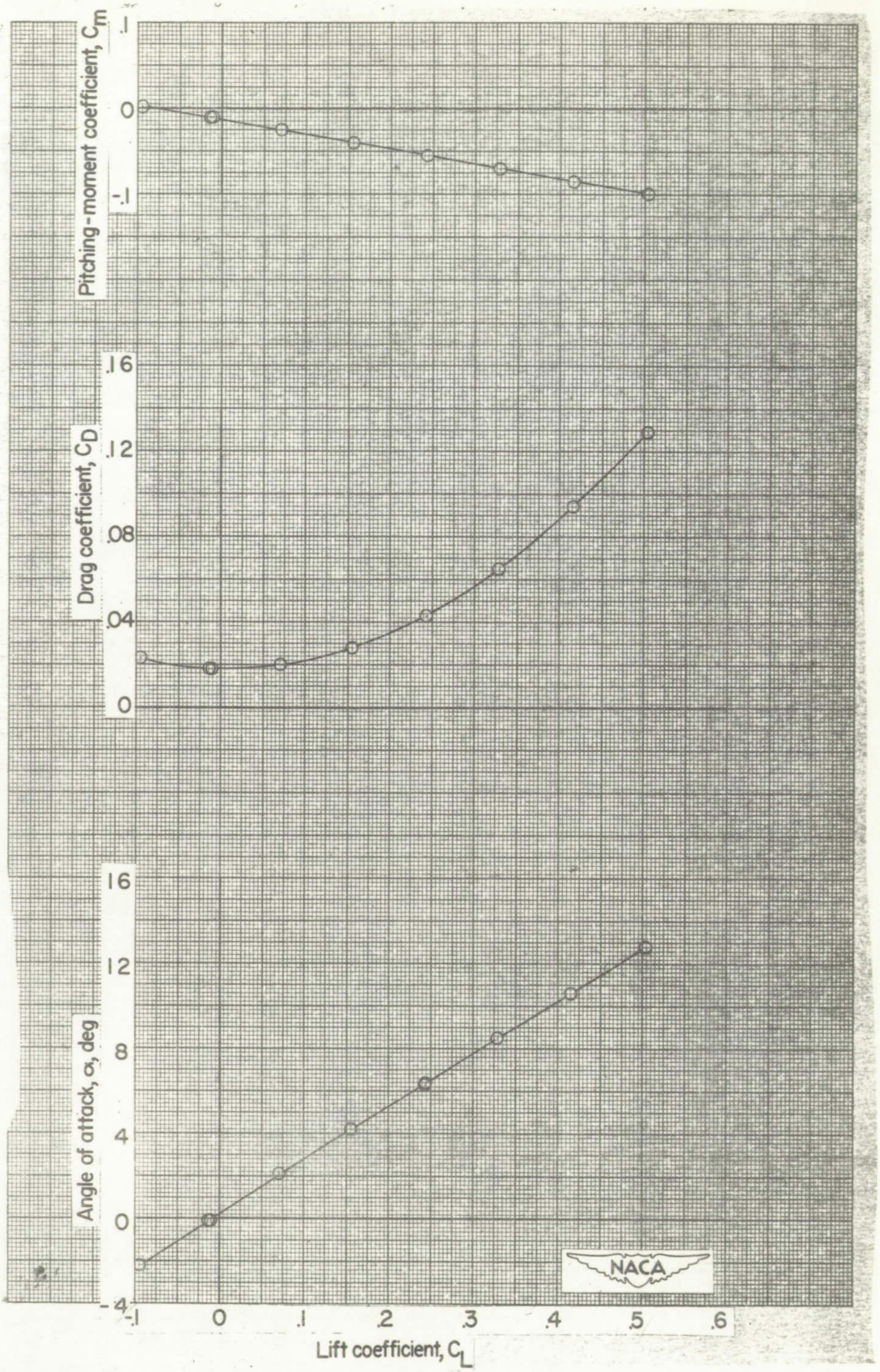
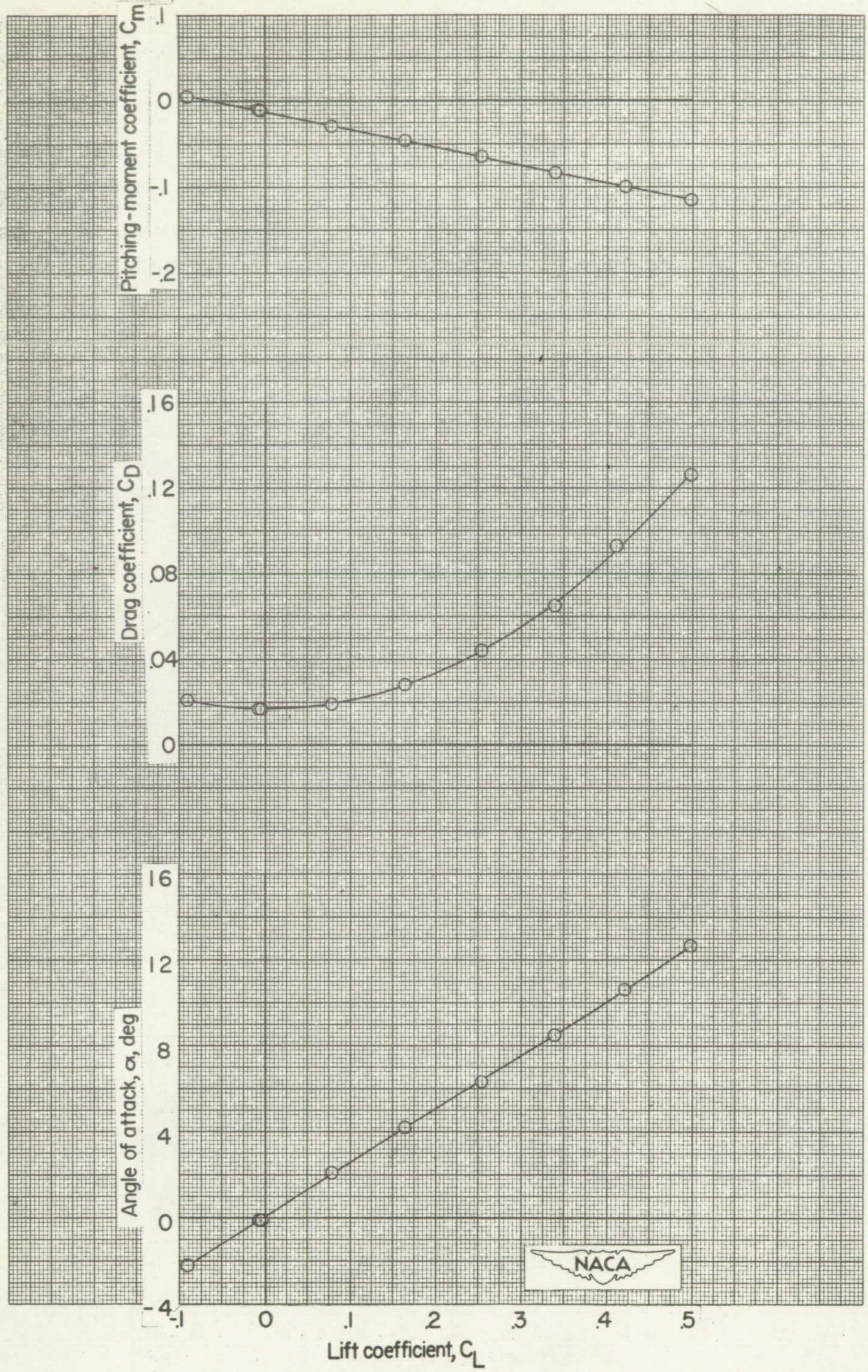


Figure 5.- Aerodynamic characteristics in pitch of body of revolution based on body frontal area and length. M = 2.01.



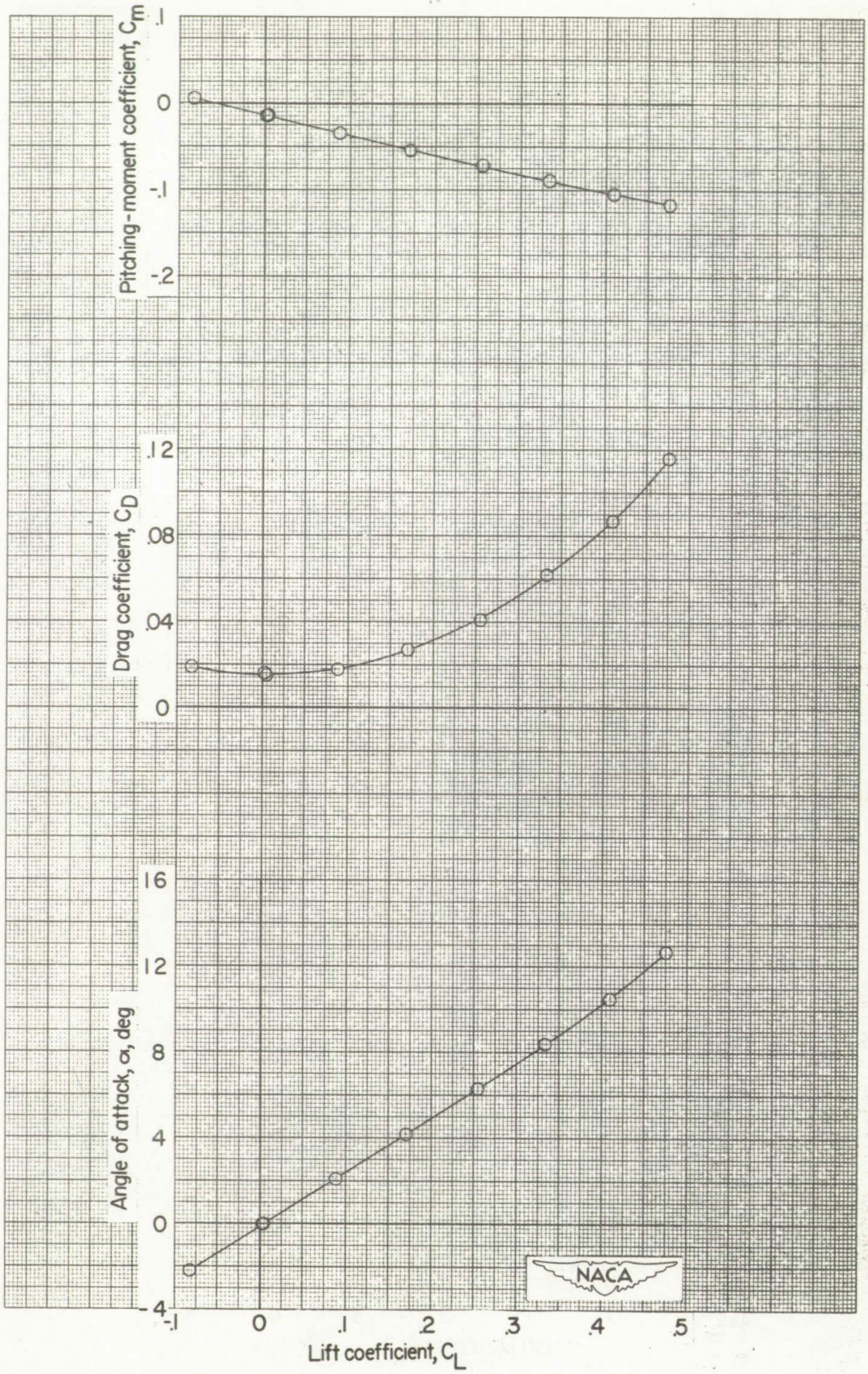
(a) $\Lambda = 10.8^{\circ}$; $\frac{t}{c} = 0.04$.

Figure 6.- Aerodynamic characteristics in pitch of the various wing-body combinations. M = 2.01.



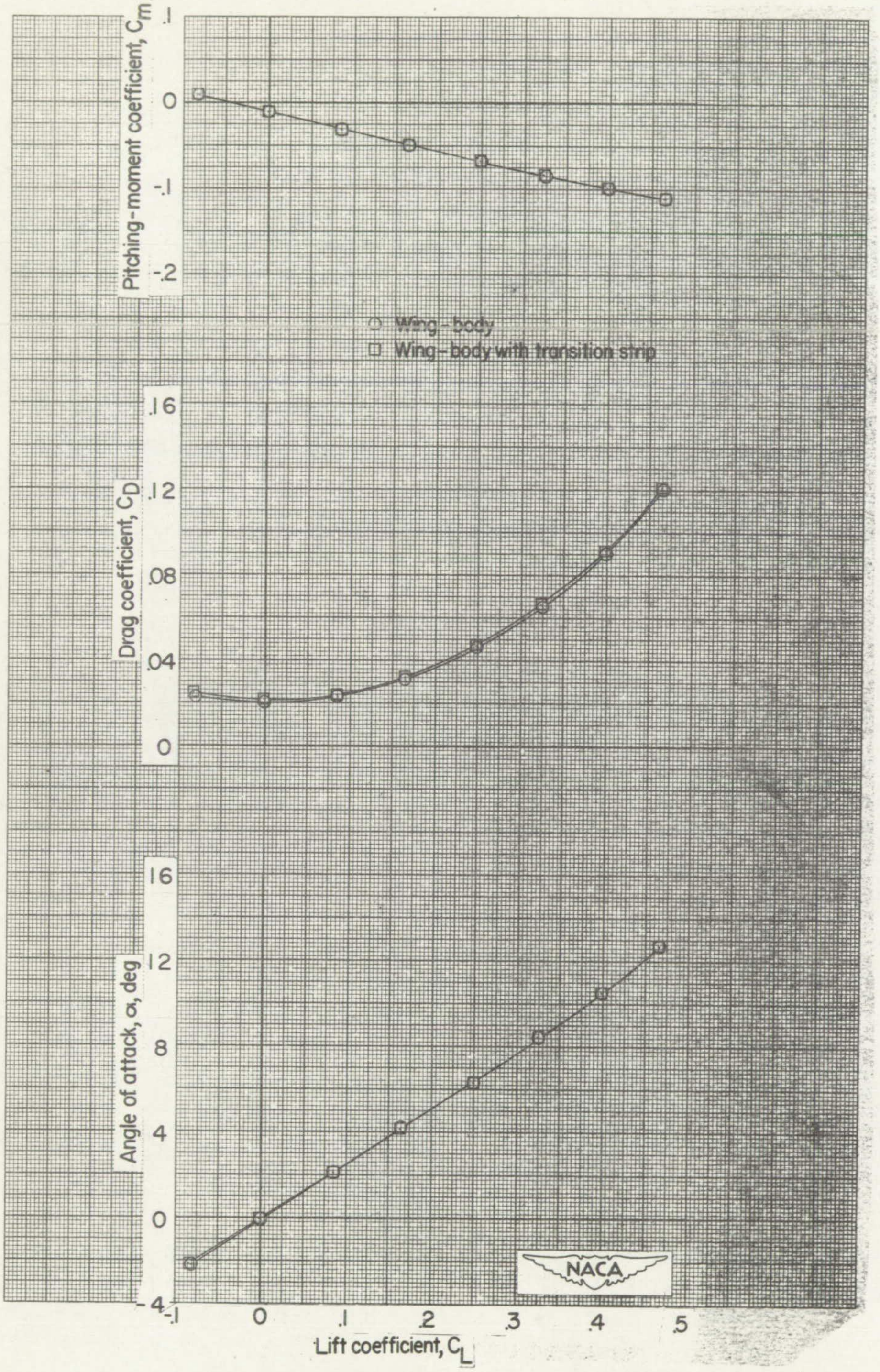
(b) $\Lambda = 35^{\circ}$; $\frac{t}{c} = 0.04$.

Figure 6.- Continued.



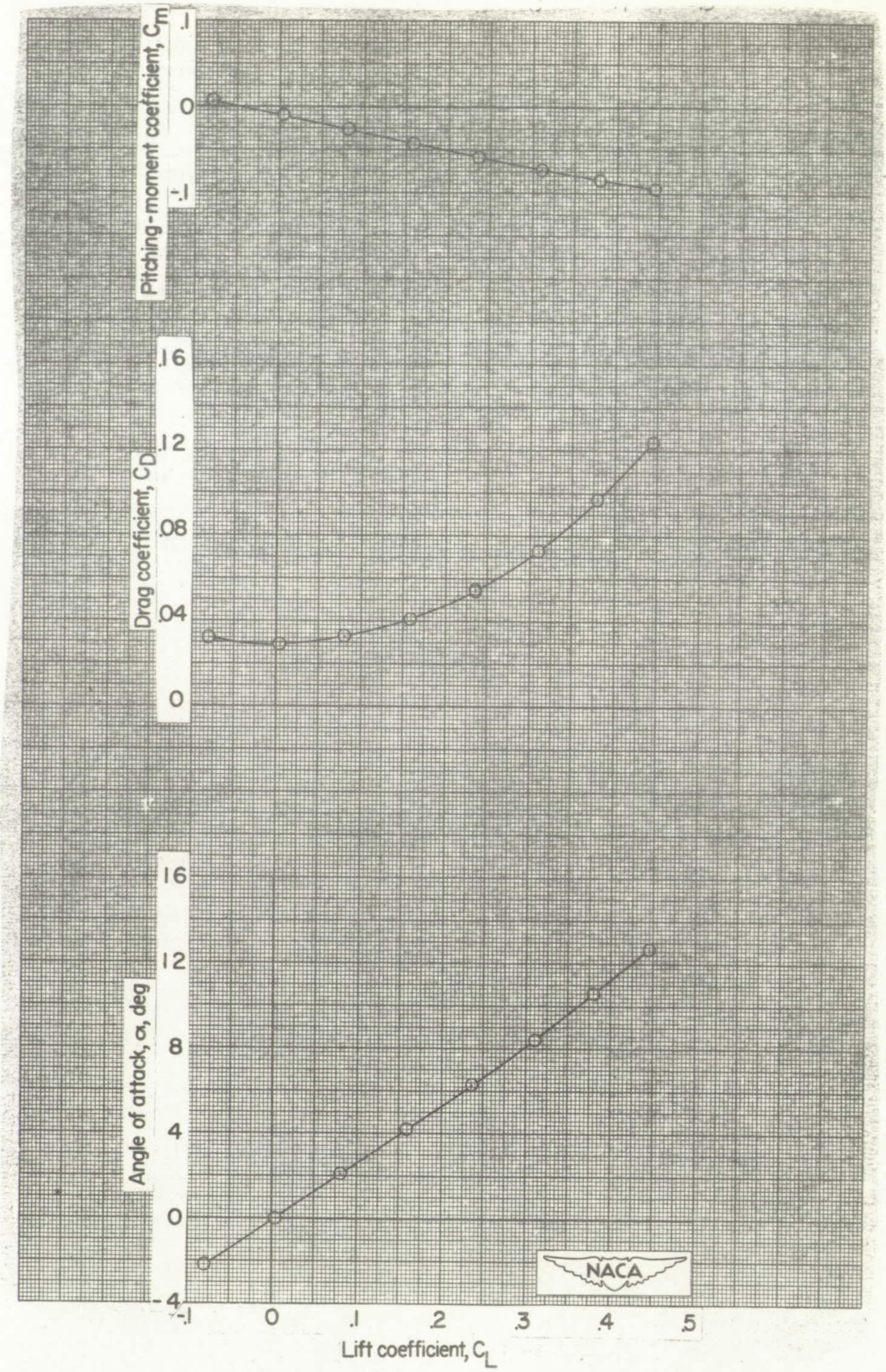
(c) $\Lambda = 47^\circ$; $\frac{t}{c} = 0.04$.

Figure 6.- Continued.



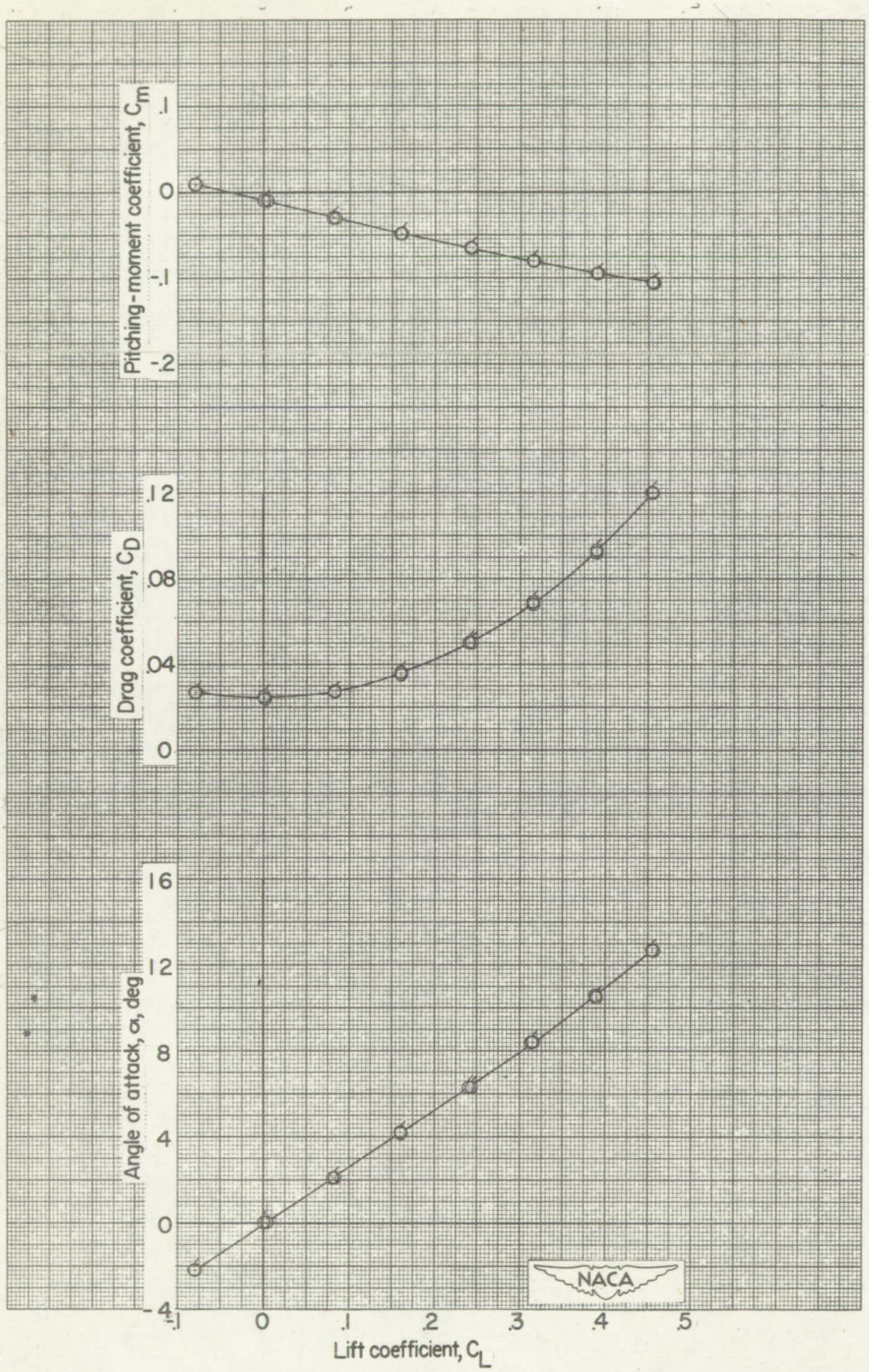
(d) $\Lambda = 47^{\circ}$; $\frac{t}{c} = 0.06$.

Figure 6.- Continued.



(e) $\Lambda = 47^{\circ}$; $\frac{t}{c} = 0.09$.

Figure 6.- Continued.



(f) $\Lambda = 47^{\circ}$; $\frac{t}{c}$ = 0.12, 0.06, 0.06.

Figure 6.- Concluded. Flagged symbols are data from a repeat run.

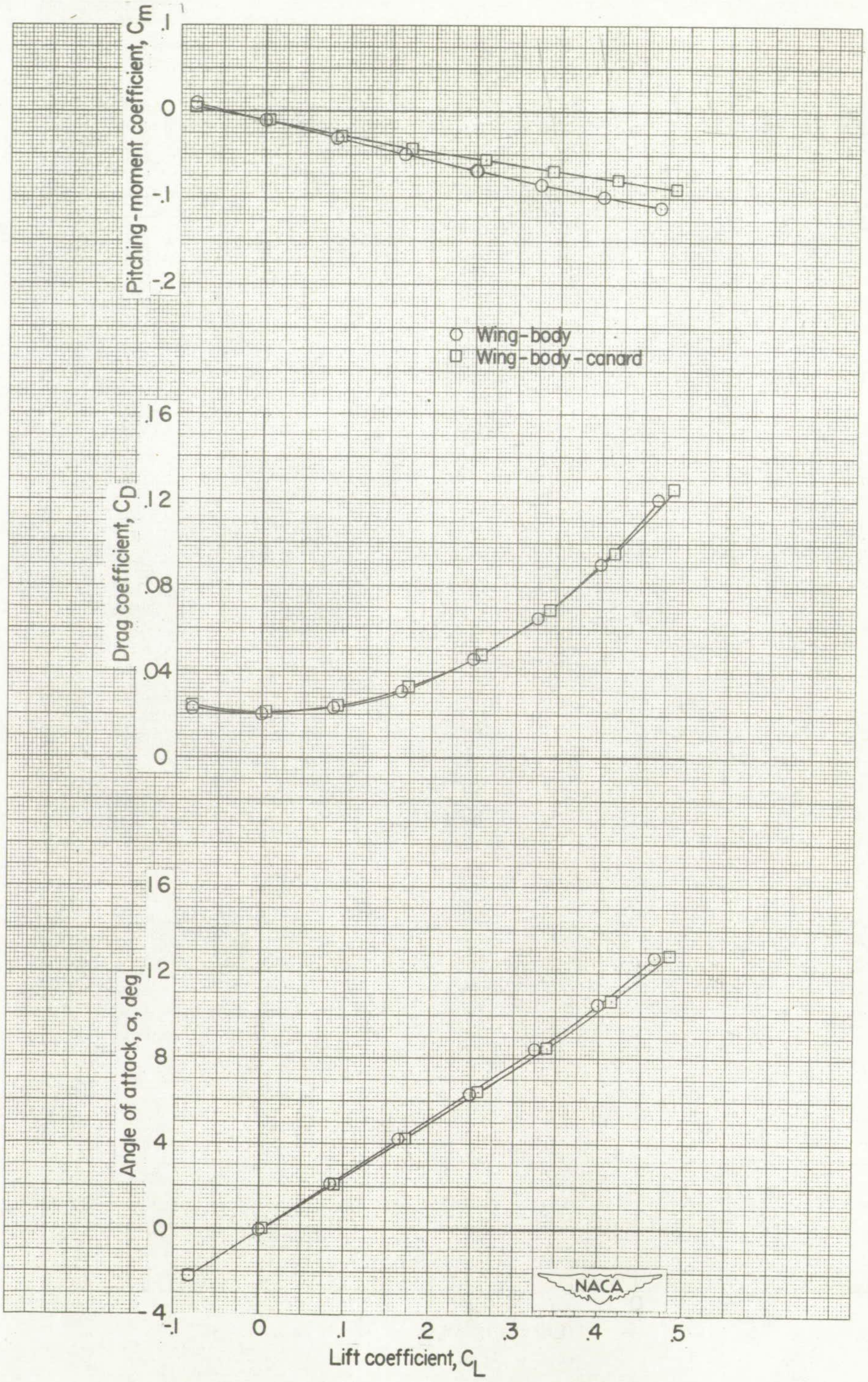
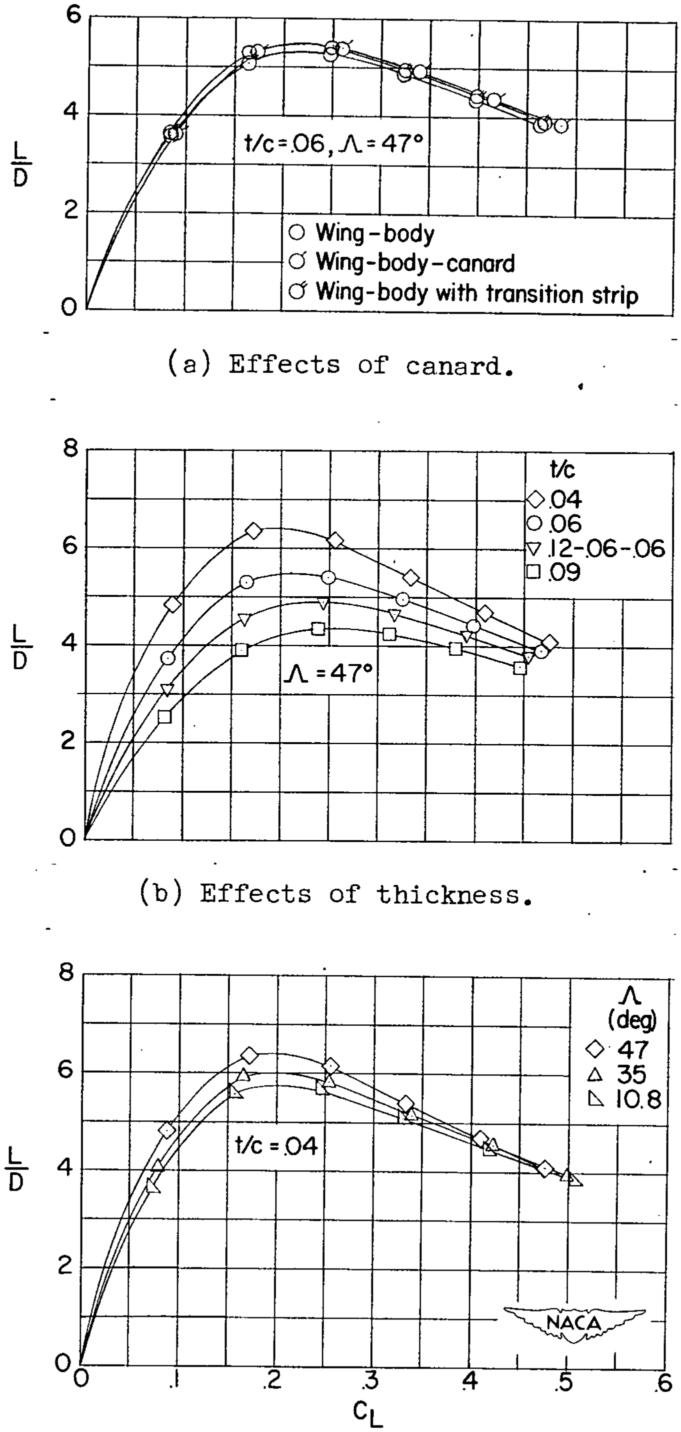


Figure 7.- Aerodynamic characteristics in pitch of a wing-body combination with and without canard. $\Lambda = 47^{\circ}$; $\frac{t}{c} = 0.06$.



(a) Effects of canard.

(b) Effects of thickness.

(c) Effects of sweep.

Figure 8.- Variation of lift-drag ratios with lift coefficient for the various wing-body configurations. M = 2.01.

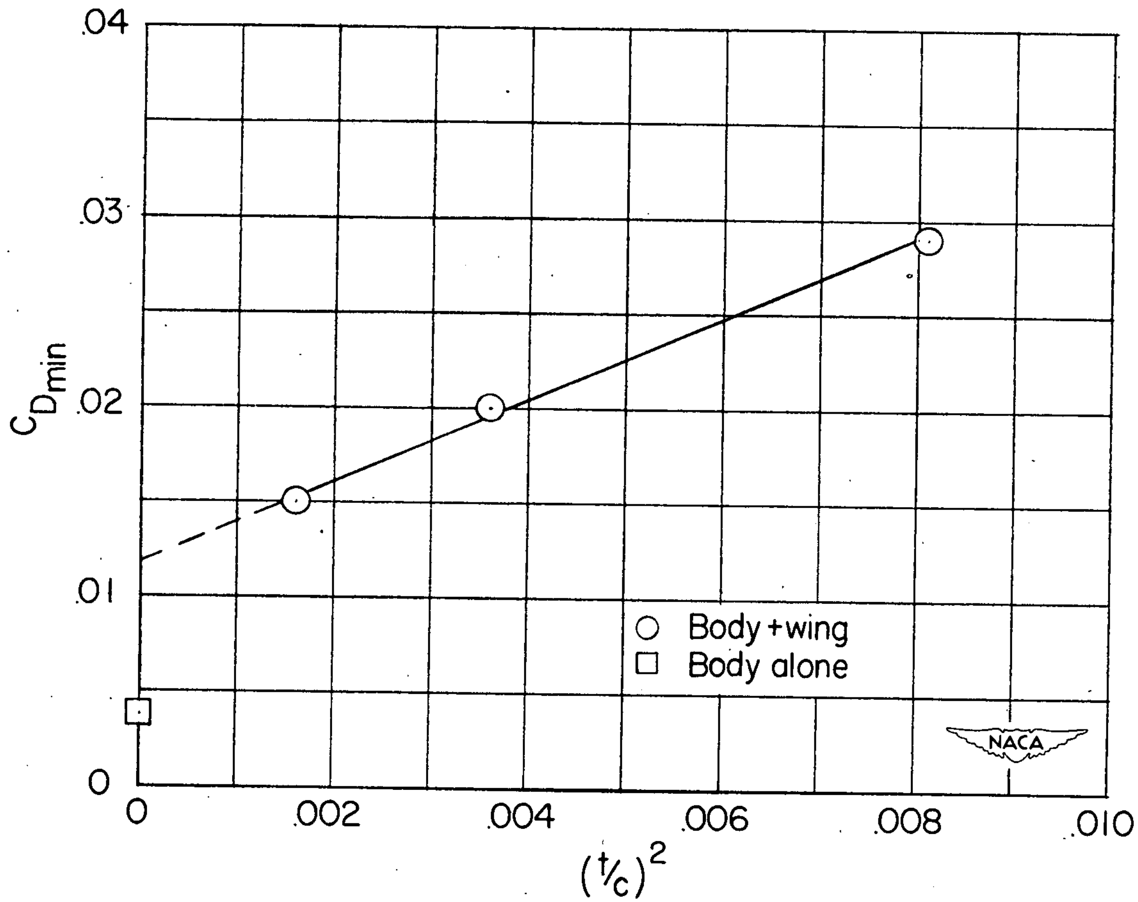


Figure 9.- Variation of minimum drag coefficient with the square of the thickness ratio. $\Lambda = 47^{\circ}$, M = 2.01.

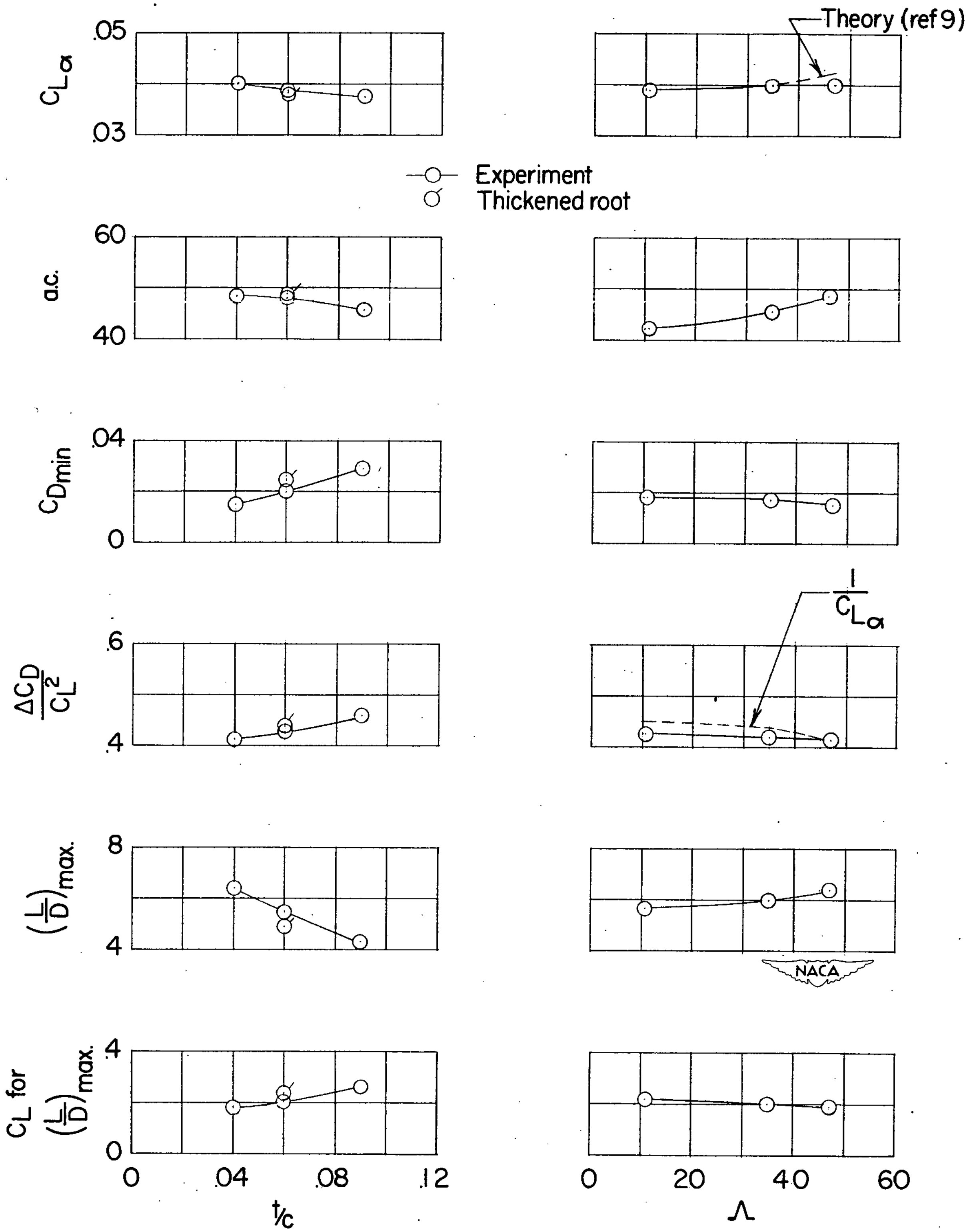


Figure 10.- Summary of the aerodynamic characteristics in pitch of the various wing-body configurations. M = 2.01.