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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 3843

EXPERIMENTAL STEADY-STATE YAWING DERIVATIVES OF A 60° DELTA-WING MODEL AS AFFECTED BY CHANGES IN VERTICAL POSITION OF THE WING AND IN RATIO OF FUSELAGE DIAMETER TO WING SPAN

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SUMMARY

An investigation was made in the Langley stability tunnel to determine the effects on the steady-state yawing derivatives of the vertical position of the wing for a 60° delta-wing model having ratios of fuselage diameter to wing span of 0.123, 0.165, and 0.246. The test Mach number was 0.13 and the Reynolds number was 1.65×10^6. The results of the investigation indicated that for angles of attack below the stall the steady-state damping in yaw decreased (the values of the yawing-moment coefficient due to yawing became less negative) when the wing was raised from a low to a middle or high position on each of the three fuselage sizes investigated. With the vertical tail on or off, the steady-state damping in yaw increased with an increase in fuselage size. The results of calculations of the oscillatory damping in yaw for the model with the large fuselage indicated that the effects of wing position would be opposite to those determined under steady-state conditions. For angles of attack below the stall, raising the wing produced a positive increment in the rolling moment due to yawing, and fuselage size had little effect on this parameter.

INTRODUCTION

Preliminary stability analyses of new airplane designs are, in many cases, made with stability derivatives which have been estimated by various means, such as in reference 1. As shown in references 2 and 3 the degree of accuracy of some of these estimated derivatives is always a matter of question; this is especially true in the moderate and high ranges of angle of attack or when a design departs from a simple body of revolution (ref. 4). Experience has shown, however, that in order to obtain more accurate estimates of the dynamic stability of a given design, it is necessary to use stability derivatives which are determined experimentally.

The present experimental investigation was undertaken, as a continuation of the work of reference 5, to show some effects of airplane design variables on the lateral derivatives of airplane models; namely, the effects on the steady-state yawing derivatives of changes in vertical position of the wing for a 60° delta-wing model having ratios of fuselage diameter to wing span of 0.123, 0.165, and 0.246. The derivatives were obtained at a Mach number of 0.13 and a Reynolds number of 1.65×10^6.

SYMBOLS

The data presented herein are referred to the stability system of axes shown in figure 1. The center of gravity was located at the projection of the wing 0.25 mean aerodynamic chord on a plane passing through the fuselage center line of each model. The coefficients and symbols used herein are defined as follows:

C_L lift coefficient, $\frac{\text{Lift}}{qS}$

C_Y side-force coefficient, $\frac{\text{Side force}}{qS}$

C_l rolling-moment coefficient, $\frac{\text{Rolling moment}}{qSb}$

C_n yawing-moment coefficient, $\frac{\text{Yawing moment}}{qSb}$

b wing span, ft

S wing area, sq ft

c local chord parallel to plane of symmetry, ft

$\bar{c}$ mean aerodynamic chord, $\frac{2}{S}\int_0^{b/2} c^2 dy$, ft

q dynamic pressure, $\frac{\rho V^2}{2}$, lb/sq ft

ρ mass density of air, slugs/cu ft

V airspeed, ft/sec

y spanwise distance measured from and perpendicular to plane of symmetry, ft

α angle of attack of fuselage center line, deg

β angle of sideslip, radians

t time, sec

ψ angle of yaw, radians

r yawing angular velocity, $d\psi/dt$, radians/sec

$\frac{rb}{2V}$ yawing-angular-velocity parameter, radians

$$C_{Y_r} = \frac{\partial C_Y}{\partial \frac{rb}{2V}}$$

$$C_{l_r} = \frac{\partial C_l}{\partial \frac{rb}{2V}}$$

$$C_{n_r} = \frac{\partial C_n}{\partial \frac{rb}{2V}}$$

$$C_{l_\beta} = \frac{\partial C_l}{\partial \beta}$$

$$C_{n_{r,\omega}} - C_{n_{\dot{\beta},\omega}} = \left(\frac{\partial C_n}{\partial \frac{rb}{2V}}\right)_\omega - \left(\frac{\partial C_n}{\partial \frac{\dot{\beta}b}{2V}}\right)_\omega$$

$$\dot{\beta} = \frac{\partial \beta}{\partial t}$$

Subscripts:

ω measured under oscillatory conditions

v vertical tail

APPARATUS, MODELS, AND TESTS

The tests of the present investigation were made in the 6- by 6-foot curved-flow test section of the Langley stability tunnel, in which the airstream is curved about a strut-supported model in order to simulate an airplane in curved flight. (See ref. 6.) The strut was rigidly attached to an electromechanical balance system.

The models used for the present investigation consisted of a 3-percent-thick, 60° delta wing mounted in a low, middle, or high position on each of three fuselages which had ratios of maximum fuselage diameter to wing span of 0.123, 0.165, and 0.246. The fuselage fineness ratios corresponding to the ratios of fuselage diameter to wing span were 12, 9, and 6, respectively. The length of each fuselage was 54.00 inches. A single vertical tail was common to all configurations, and this was the vertical tail designated as V_3 in reference 5. Since the fuselage diameter at the base varied when the ratio of fuselage diameter to wing span was changed, the overall span of the vertical tail also varied. The exposed area of the tail, however, was approximately the same for all models. The models did not have a horizontal tail. A drawing of the models is presented as figure 2, and additional data may be obtained from table I and from reference 5.

In order to obtain the steady-state yawing derivatives, the models were tested, tail on and off, through a range of angle of attack from -4° to 36° at values of $\frac{rb}{2V}$ of 0, -0.0316, -0.0670, and -0.0881. The angle of sideslip was 0° for all tests. The Mach number was 0.13, and the Reynolds number was 1.65×10^{6}.

CORRECTIONS

The angle of attack was corrected for the effects of the jet boundaries by the methods of reference 7. Corrections were applied to the derivatives to account for the pressure gradient associated with curved flow. (See ref. 6.) Blockage corrections were considered to be negligible and hence were not applied. The effects of the support strut on the derivatives were not determined but, on the basis of past experience, they were not expected to influence the accuracy of the data presented herein.

RESULTS AND DISCUSSION

Presentation of Results

The relation between the angle of attack and the lift coefficient is presented in figure 3 for each configuration. The effect of the vertical position of the wing on the steady-state yawing derivatives for the model with each of the three fuselage sizes is shown in figure 4 with the vertical tail on and in figure 5 with the vertical tail off. The effects of the wing position and the fuselage size on the value of C_{l_r} and C_{n_r} at an angle of attack of 0° are summarized in figure 6.

Effect of Wing Position and Fuselage Size on C_{l_r}

The effects of vertical position of the wing on the derivatives C_{l_r} (figs. 4 and 5) and $-C_{l_\beta}$ (ref. 5) are similar in the low and moderate ranges of angle of attack since an upward displacement of the wing produces a positive increment in these derivatives. In the low range of angle of attack, the data of figures 4 and 5 indicate that an increase in the ratio of fuselage diameter to wing span increases the magnitude of the effect on C_{l_r} produced by raising the wing. However, the data of figure 6, where the derivatives C_{l_r} and C_{n_r} are plotted against the ratio of wing height to wing span, indicate that only a small part of this increase is caused by the increase in fuselage diameter since the variation of C_{l_r} with ratio of wing height to wing span was about the same for all fuselage sizes. The major increase in C_{l_r} was caused by the increased wing height in terms of wing span since the ratio of wing height to fuselage diameter was 0.333 for all models. The tail contribution to C_{l_r} decreases slightly as the wing is moved from the low to the middle or high position and is essentially unaffected by changes in fuselage size (fig. 6). For the range of wing positions and fuselage sizes investigated, the value of C_{l_r} near $\alpha = 0^{\circ}$ varies linearly with changes in wing position.

Effect of Wing Position and Fuselage Size on C_{n_r}

With the tail on and for angles of attack below the stall, raising the wing decreases the value (becomes less negative) of C_{n_r}, and increasing the fuselage size increases the value of C_{n_r} (fig. 4). With the tail off, the effects of the wing position are smaller and somewhat different from the tail-on results in the low range of angle of attack inasmuch as the midwing position has the least value of C_{n_r} (figs. 5 and 6) and the high and low positions have values of C_{n_r} which are nearly equal for a given fuselage size. With the tail off (fig. 5), the effects of fuselage size on C_{n_r} are similar to those with tail on. The tail contribution to C_{n_r} decreases as the wing is raised and slightly increases as the fuselage size is increased (fig. 6).

Comparison of Steady-State and Oscillatory Damping in Yaw

The results of an investigation of a 45° swept-wing model (ref. 8) have indicated that raising the wing increased the damping in yaw determined under oscillatory conditions $C_{n_{r,\omega}} - C_{n_{\dot{\beta},\omega}}$, whereas in the present investigation the steady-state damping in yaw C_{n_r}, as determined from curved-flow tests, decreased when the wing was raised.

In order to indicate whether differences exist in the steady and oscillatory damping in yaw for the present models, calculations were made, by using the method of reference 8, and the sidewash data from reference 5, to determine the effect of wing position on the oscillatory damping in yaw for the model with the largest fuselage diameter. Since it was shown in reference 8 that almost equal values of the steady-state and oscillatory damping in yaw were obtained when the tail was removed ($\alpha = 0^\circ$), it was assumed that a similar circumstance occurred for the present investigation. In effect, therefore, the steady-state value of C_{n_r} and the value of C_{n_r} determined under pure yawing oscillatory conditions (no sideslipping) were assumed to be equal. The total oscillatory damping in yaw was obtained by the addition of the steady-state damping in yaw of the wing-fuselage combination and the tail contribution to the oscillatory damping in yaw as expressed by $\left(C_{n_{r,\omega}} - C_{n_{\dot{\beta},\omega}}\right)_V$.

The calculated results are shown in figure 6 and indicate that under oscillatory conditions, where the angle of sideslip is changing as well as the angle of yaw, the $\dot{\beta}$ contribution to the damping is quite appreciable, as was noted in reference 8, and the effects of wing position may be opposite to those for the steady-state condition. The results of additional calculations (not presented) indicate that the effect of fuselage size on the tail contribution to the oscillatory damping in yaw was negligibly small for the models of this investigation at an angle of attack of 0°.

CONCLUSIONS

Results of an investigation made in the Langley stability tunnel to determine the effects of wing position and fuselage size on the steady-state yawing derivatives of a 60° delta-wing model indicated the following conclusions:

1. For angles of attack below the stall, the steady-state damping in yaw decreased (the values of the yawing-moment coefficient due to yawing became less negative) when the wing was raised from a low to a middle or a high position on fuselages having ratios of the maximum diameter to wing span of 0.123, 0.165, and 0.246. This was the result of a decrease in the tail contribution to the steady-state damping in yaw when the wing was raised since the effect of wing position on the damping in yaw for the wing-fuselage combination was small. With the vertical tail on or off, the steady-state damping in yaw increased with an increase in fuselage size.

2. The results of calculations of the oscillatory damping in yaw for the model with the large fuselage indicated that the effects of wing position would be opposite to those determined under steady-state conditions.

3. For angles of attack below the stall, raising the wing produced a positive increment in the rolling moment due to yawing, and fuselage size had little effect on this parameter.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., July 25, 1956.

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5. Goodman, Alex, and Thomas, David F., Jr.: Effects of Wing Position and Fuselage Size on the Low-Speed Static and Rolling Stability Characteristics of a Delta-Wing Model. NACA Rep. 1224, 1955. (Supersedes NACA TN 3063.)

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TABLE I.- GEOMETRIC CHARACTERISTICS OF MODELS

Fuselage:			
Fineness ratio	12	9	6
Maximum diameter, in.	4.5	6.0	9.0
Ratio of maximum diameter to wing span	0.123	0.165	0.246
Wing:			
Aspect ratio			2.31
Taper ratio			0
Leading-edge sweep angle, deg			60
Dihedral angle, deg			0
Twist, deg			0
Area, sq in.			576.7
NACA airfoil section parallel to plane of symmetry			65A003
Vertical tail:			
Aspect ratio			2.18
Taper ratio			0
Leading-edge sweep angle, deg			42.5
Area for 12.00-inch span, sq in.			66.0
NACA airfoil section parallel to root			65-006
Tail length from center of gravity to 0.25 mean aerodynamic chord of tail, in.			21.5
Ratio of tail area to wing area			0.115
Ratio of tail length to wing span			0.59

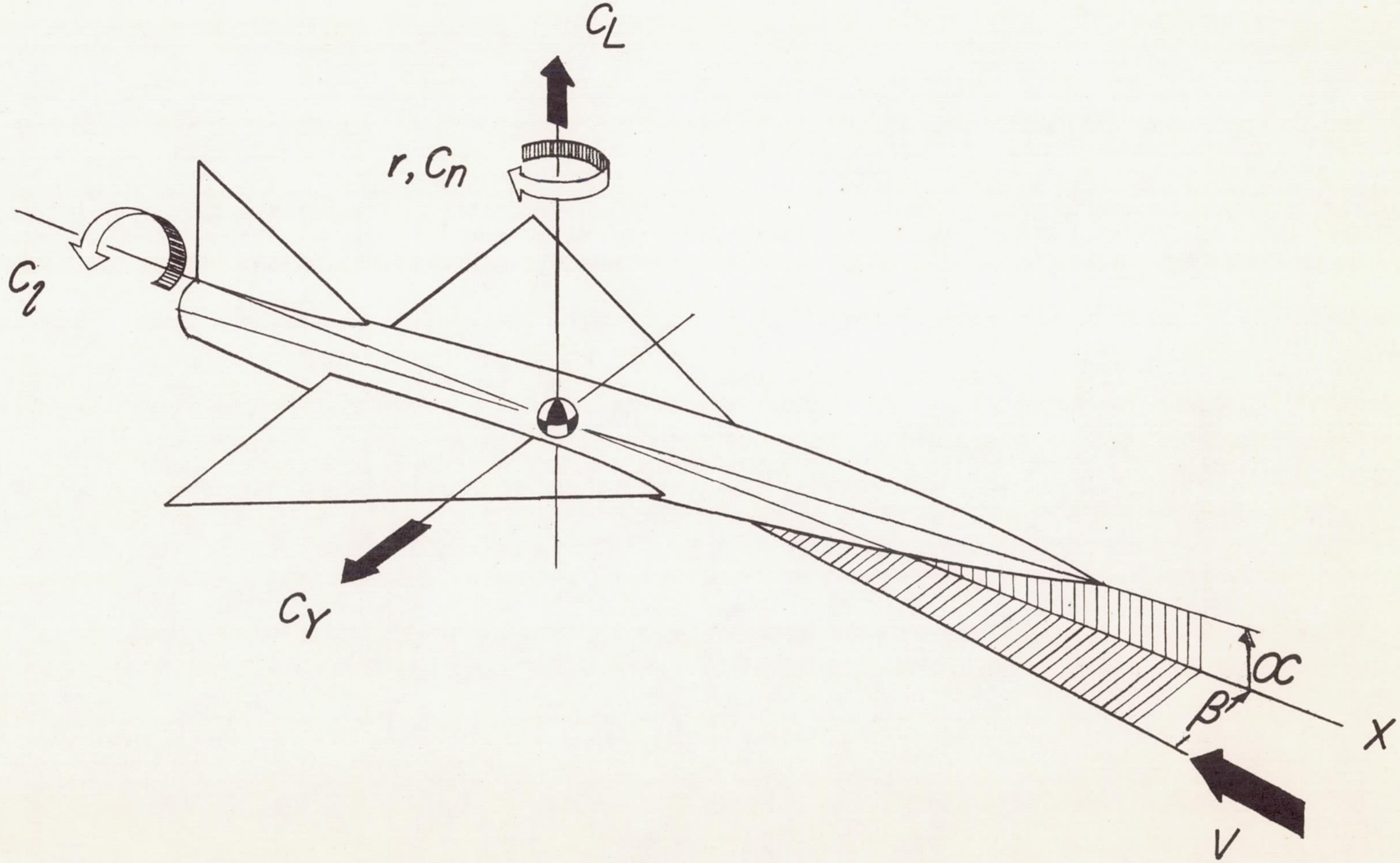


Figure 1.- Stability system of axes. Arrows indicate positive angles, velocities, and coefficients.

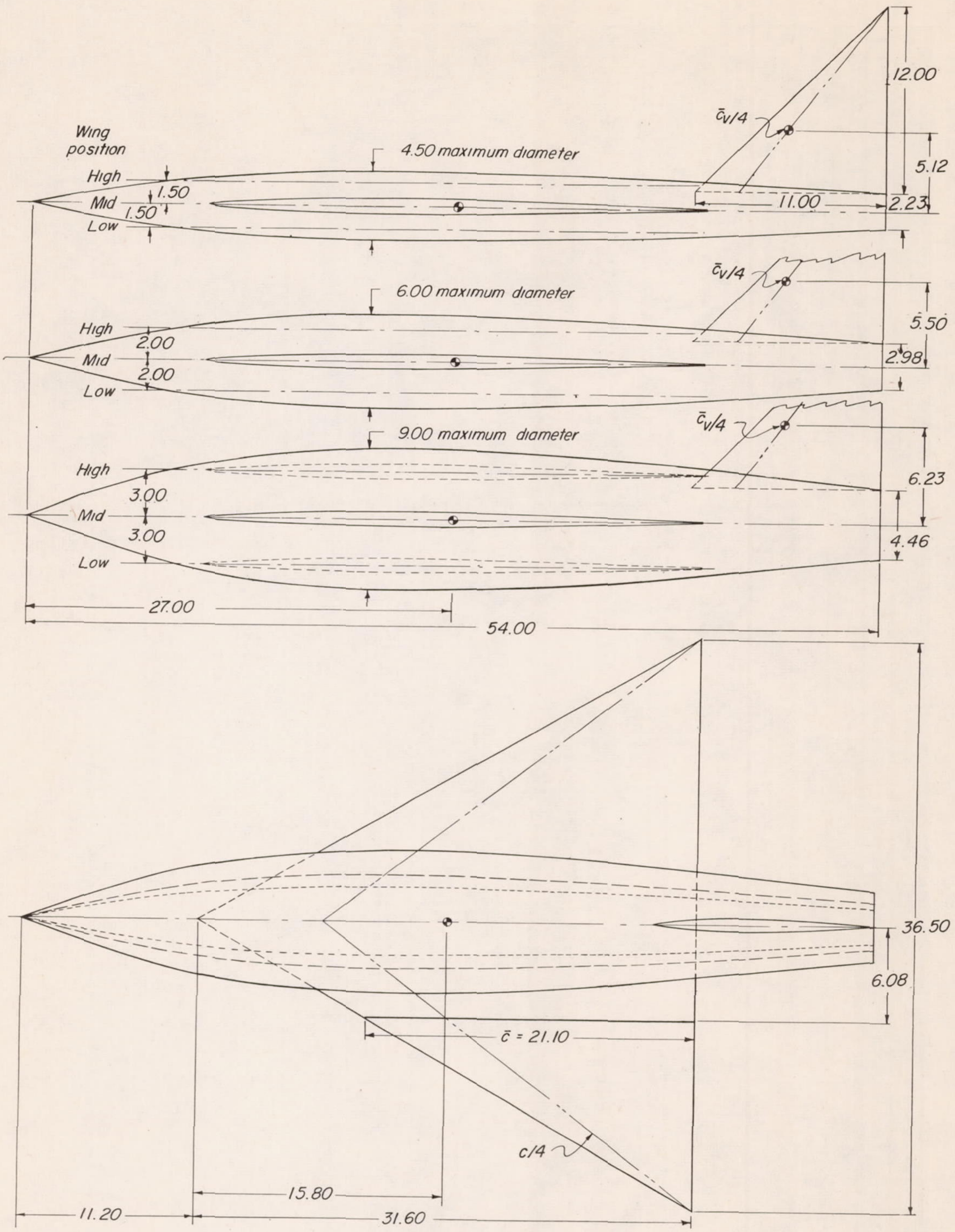
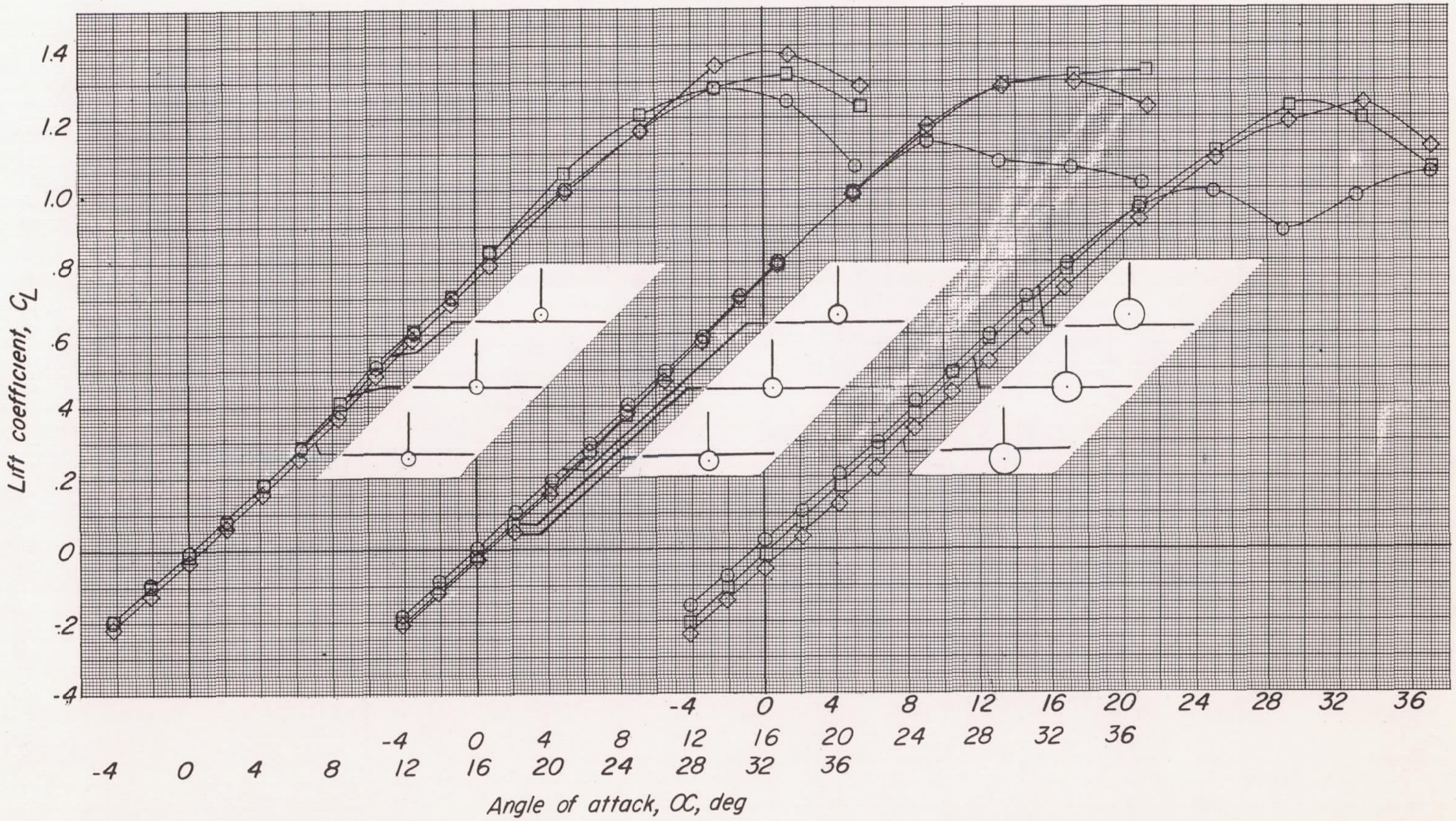
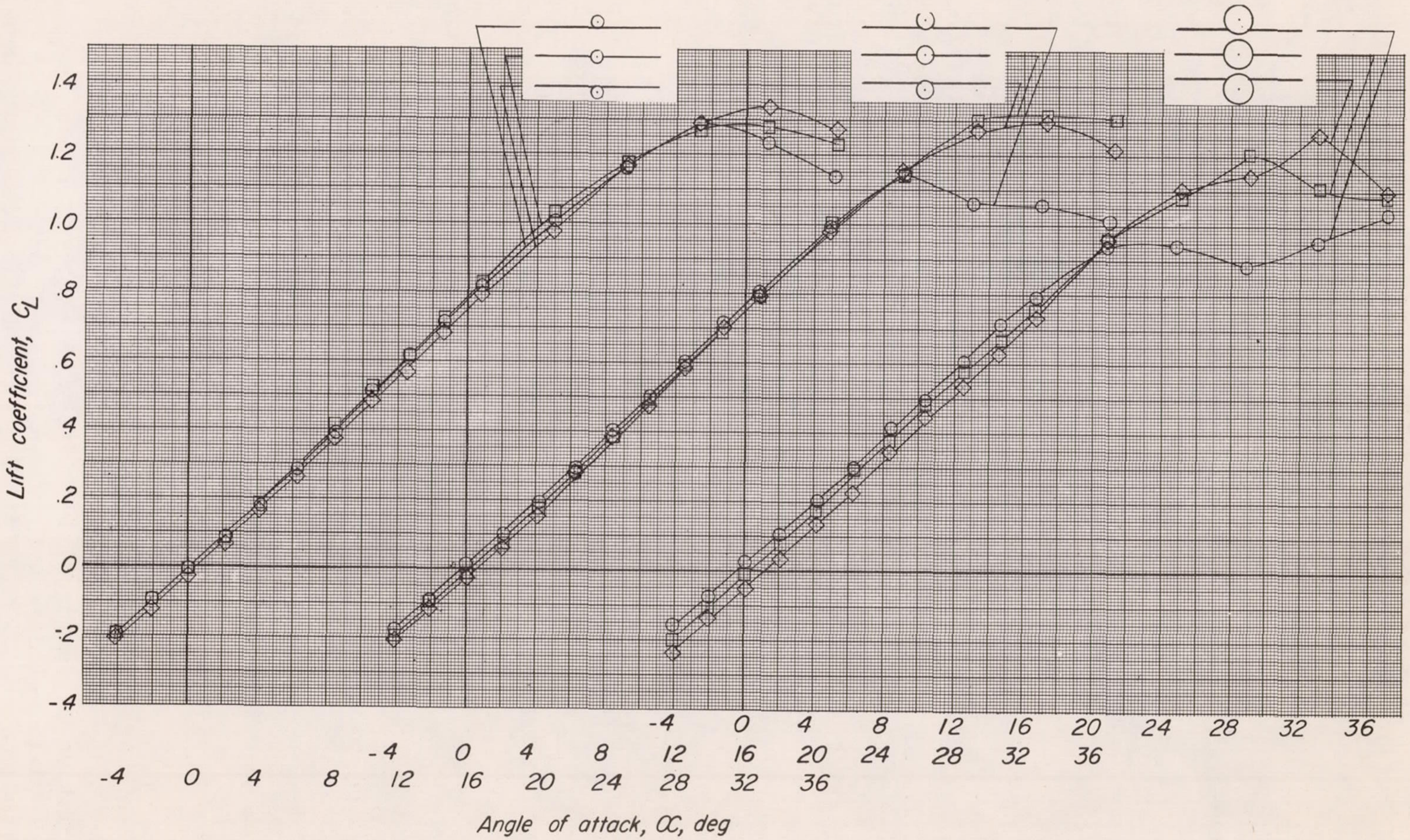


Figure 2.- Details of models. Dimensions are in inches.



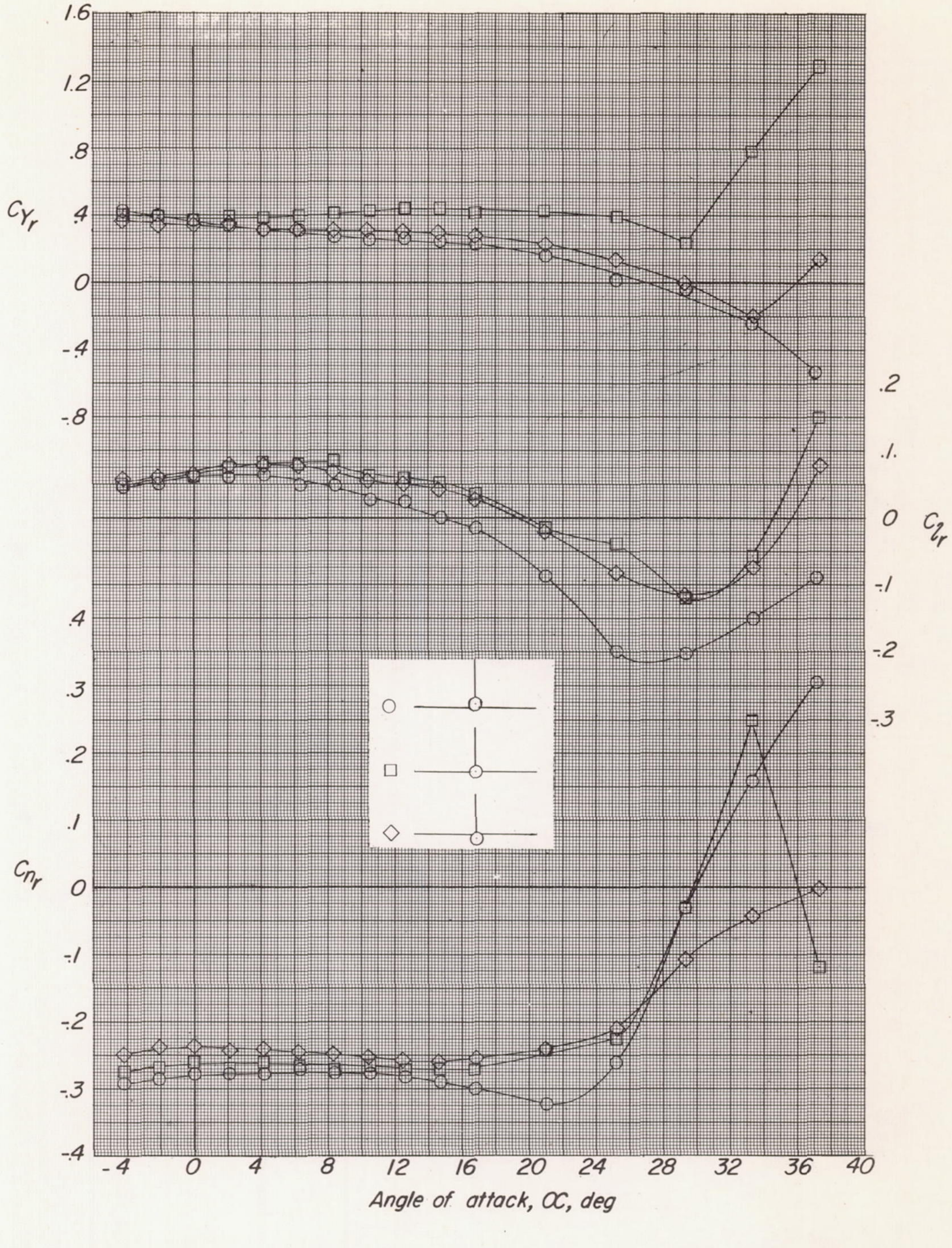
(a) Vertical tail on.

Figure 3.- Variation of lift coefficient with angle of attack for the various models. Ratios of fuselage diameter to wing span are 0.123, 0.165, and 0.246.



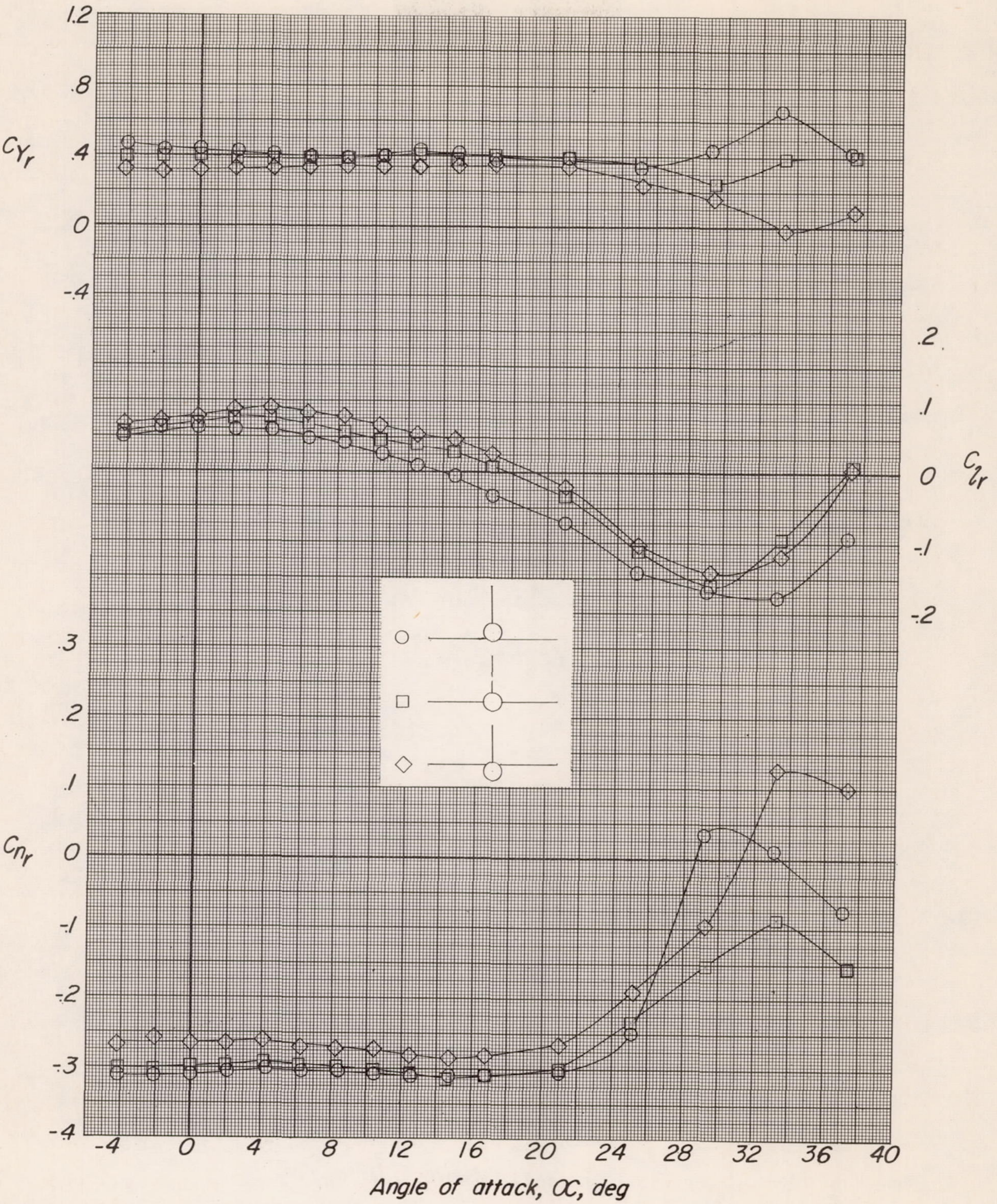
(b) Vertical tail off.

Figure 3.- Concluded.



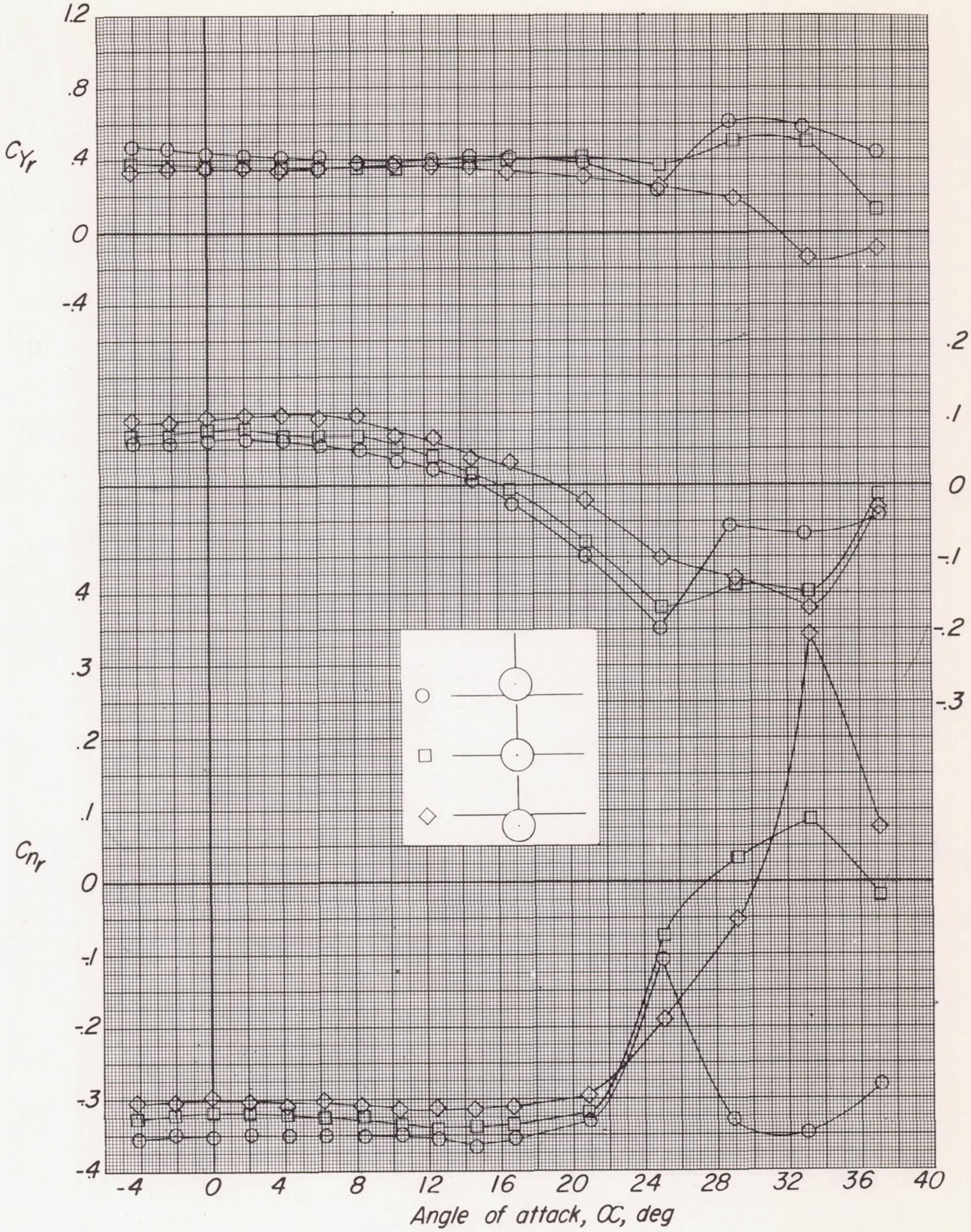
(a) Ratio of fuselage diameter to wing span of 0.123.

Figure 4.- Variation of yawing derivatives with angle of attack for the various models with a vertical tail on.



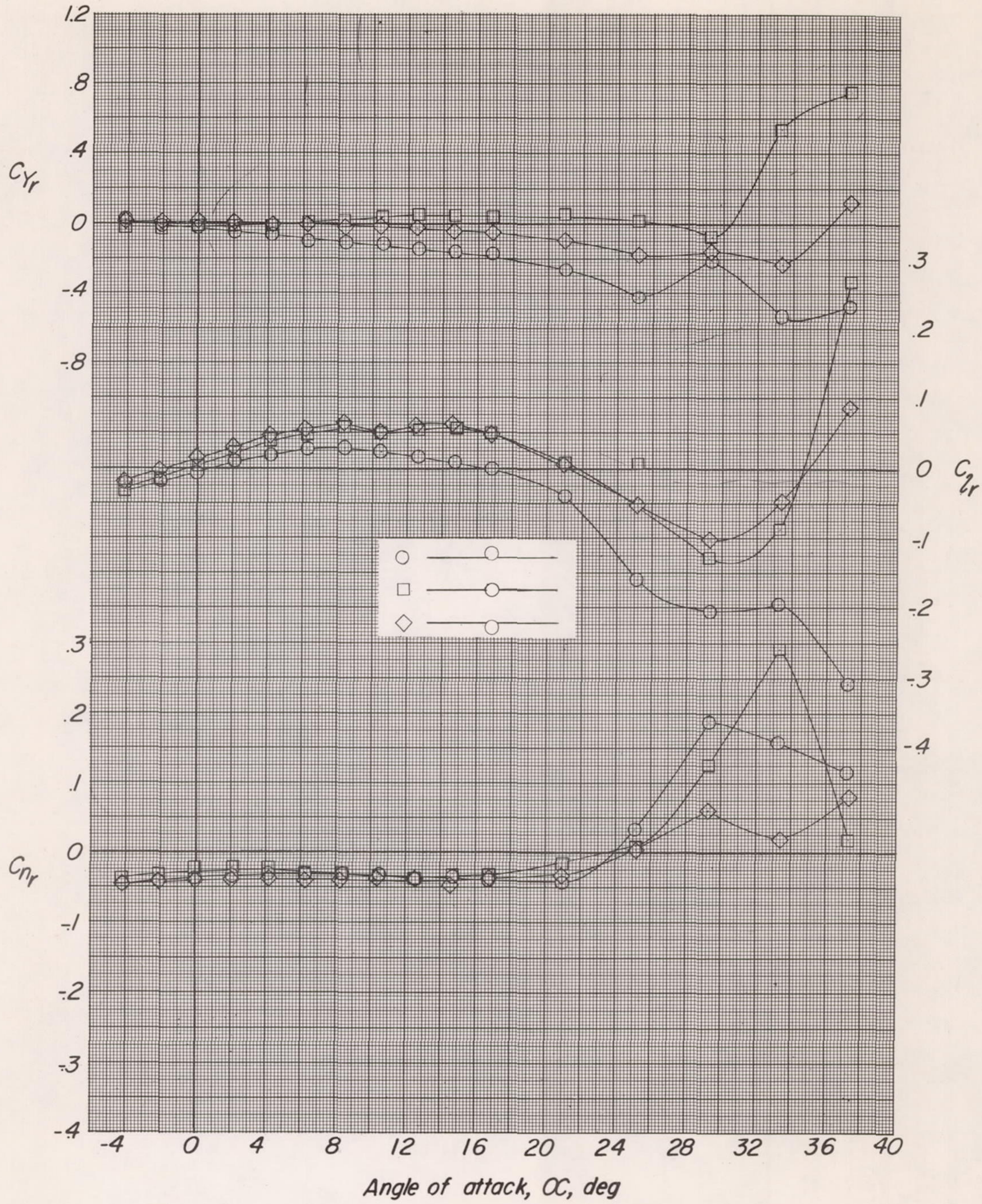
(b) Ratio of fuselage diameter to wing span of 0.165.

Figure 4.- Continued.



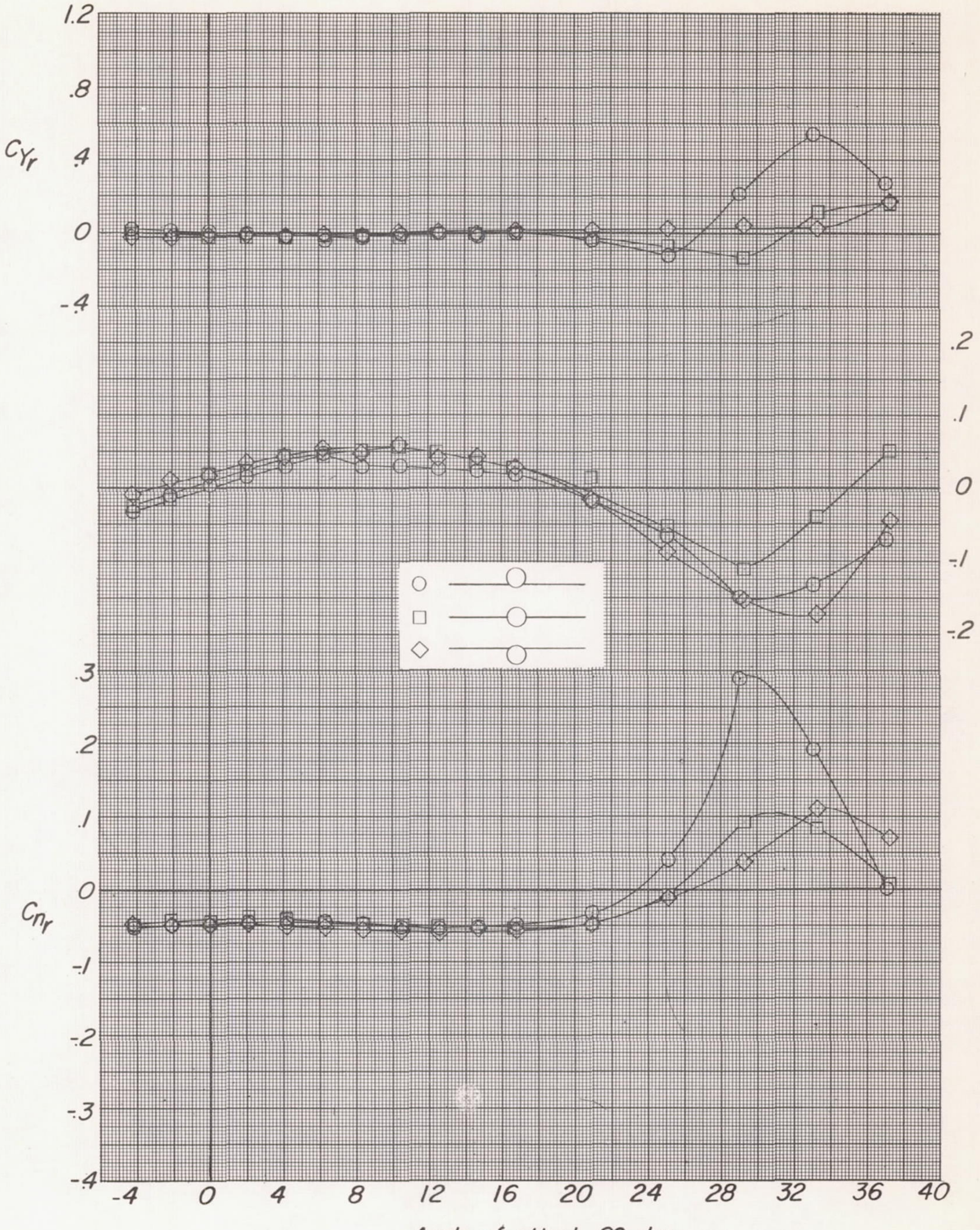
(c) Ratio of fuselage diameter to wing span of 0.246.

Figure 4.- Concluded.



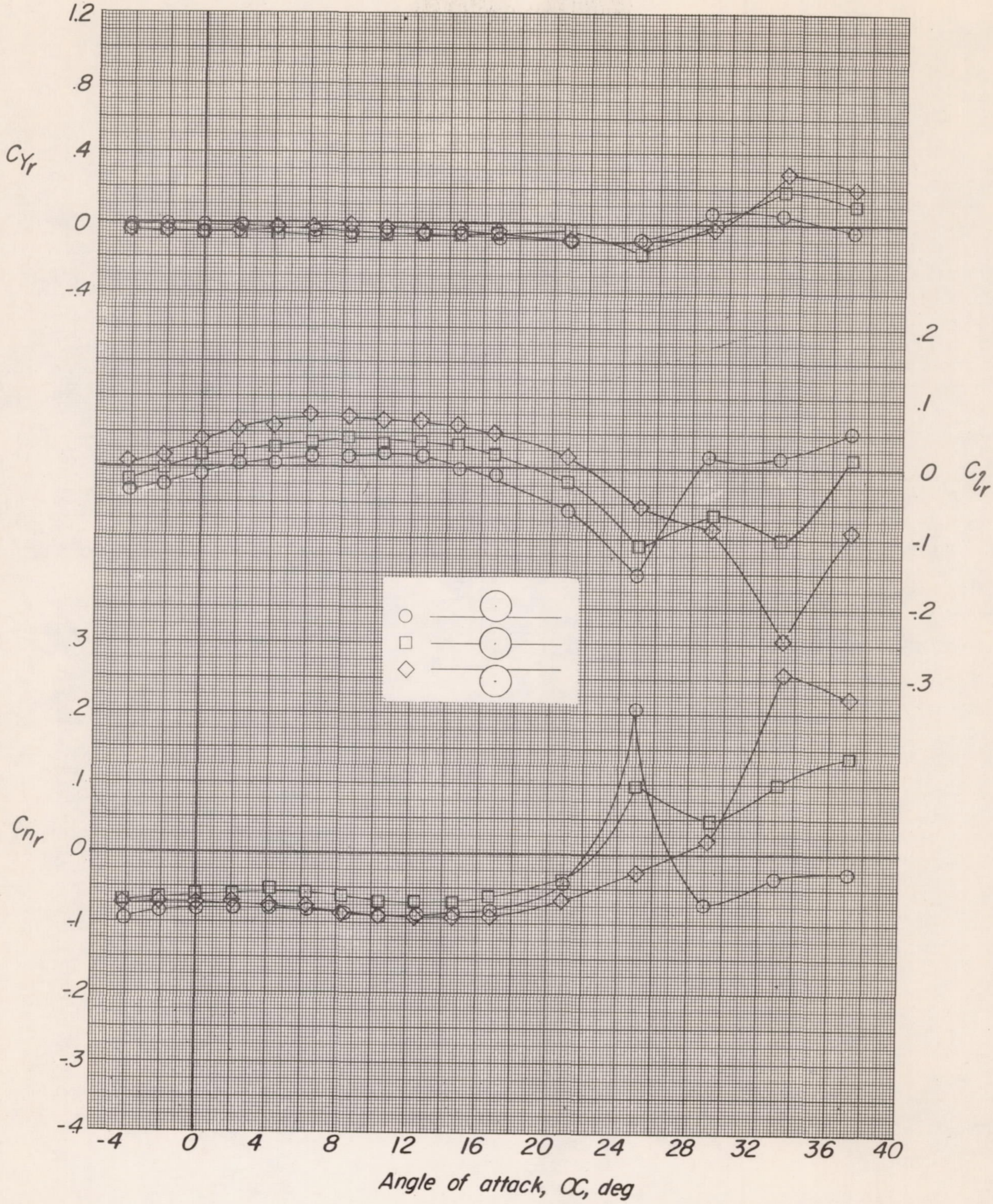
(a) Ratio of fuselage diameter to wing span of 0.123.

Figure 5.- Variation of yawing derivatives with angle of attack for the various models without a vertical tail.



(b) Ratio of fuselage diameter to wing span of 0.165.

Figure 5.- Continued.



(c) Ratio of fuselage diameter to wing span of 0.246.

Figure 5.- Concluded.

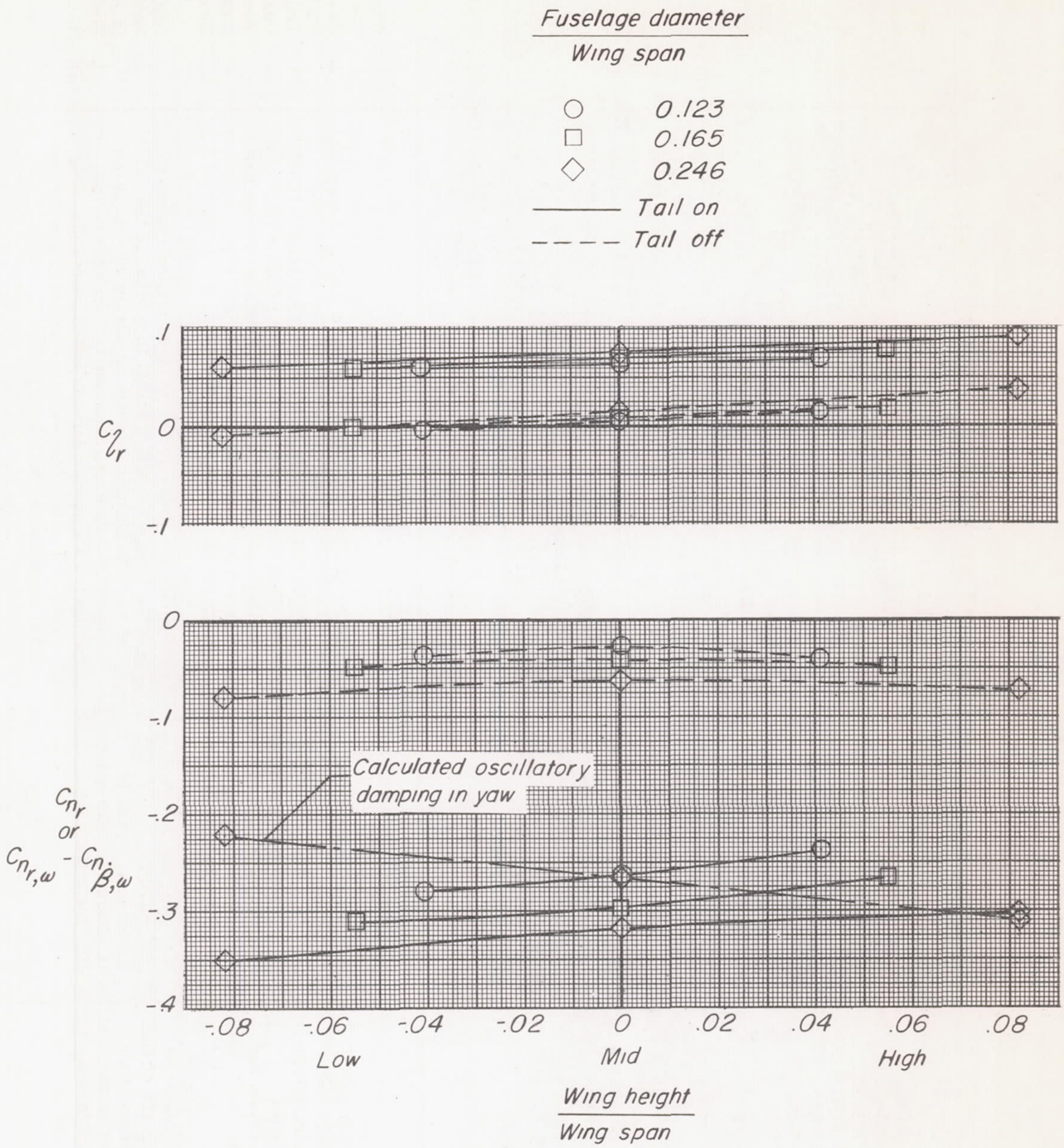


Figure 6.- Summary of effects of wing position and fuselage size on the steady-state yawing derivatives of a 60° delta-wing model. $\alpha = 0^{\circ}$.

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