



TECHNICAL NOTE

D-690

AERODYNAMIC LOADS AT MACH NUMBERS FROM 0.70 TO 2.22
ON AN AIRPLANE MODEL HAVING A WING AND CANARD
OF TRIANGULAR PLAN FORM AND EITHER SINGLE
OR TWIN VERTICAL TAILS

By Victor L. Peterson and Gene P. Menees

Ames Research Center
Moffett Field, Calif.

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SUMMARY

Results of an investigation of the aerodynamic loads on a canard airplane model are presented without detailed analysis for the Mach number range of 0.70 to 2.22. The model consisted of a triangular wing and canard of aspect ratio 2 mounted on a Sears-Haack body of fineness ratio 12.5 and either a single body-mounted vertical tail or twin wing-mounted vertical tails of low aspect ratio and sweptback plan form. The body, right wing panel, single vertical tail, and left twin vertical tail were instrumented for measuring pressures. Data were obtained for angles of attack ranging from -4° to $+16^{\circ}$, nominal canard deflection angles of 0° and 10° , and angles of sideslip of 0° and 5.3° . The Reynolds number was 2.9×10^6 based on the wing mean aerodynamic chord. Selected portions of the data are presented in graphical form and attention is directed to some of the results of the investigation. All of the experimental results have been tabulated in the form of pressure coefficients and integrations of the pressure coefficients and are available as supplements to this paper. A brief summary of the contents of the tabular material is given.

INTRODUCTION

The desire to achieve long-range supersonic flight has resulted in considerable effort being expended to develop configurations having high aerodynamic efficiency at supersonic speeds while maintaining acceptable characteristics at transonic and subsonic speeds. One important step in the development of such configurations is the choice of a longitudinal aerodynamic control producing low supersonic trim drag and sufficient control power for maneuvering and landing. Therefore, an experimental program has been undertaken at the Ames Research Center to obtain static stability and control data for various types of controls used in conjunction with several wing, body, and vertical-tail combinations. Considerable information was already available in the literature on the trailing-edge flap type of control and tail-aft controls so the program was directed primarily toward the investigation of canard controls.

The results of reference 1 show that the canard-type control has distinct geometric and aerodynamic advantages over the trailing-edge flap and tail-aft controls at supersonic speeds and that it can be used for trim and control at landing speeds. However, the desirability of using the canard-type control could be lessened in some instances because of unfavorable effects of the canard wake on other components of the configuration, especially the vertical tail. For instance, the loading on a single body-mounted vertical tail can be adversely affected by the canard wake whereas the over-all effects on the loadings of twin wing-mounted vertical tails can be favorable. A knowledge of these interference effects is essential to the design of a specific configuration. Sufficient force data were obtained in the other phases of this investigation to allow the determination of the integrated interference between various component parts of the configurations (e.g., see refs. 2 and 3). However, a complete understanding of the interference problem requires the additional knowledge of the local effects as well. It has been found that many of the analytical methods predict these integrated interference effects reasonably well but, in general, do not predict the local effects and therefore cannot always be used by the structural designer.

In view of the preceding considerations and of the continued interest shown in the canard control, the purpose of this report is to present pressure-distribution data for two canard airplane configurations. The model consisted of a wing, canard, body, and either a single vertical tail mounted on the body or twin vertical tails mounted on the wing panels. The wing and canard were both of triangular plan form and aspect ratio 2 and the body was a Sears-Haack shape of fineness ratio 12.5. The vertical tails were of low aspect ratio and sweptback plan form. Force results for the model with the single vertical tail have been reported in references 2 and 3. Also, the results of a pressure-distribution test of a canard airplane configuration having a trapezoidal wing are presented in reference 4.

The present investigation was conducted in the Ames 6- by 6-Foot Supersonic Wind Tunnel and covered Mach numbers from 0.70 to 2.22 with angles of attack to 16° at 0° and 5.3° sideslip. Canard deflection angles were nominally 0° and 10° . A limited amount of testing was also done with the single vertical tail deflected 5° , leading edge left with respect to the body centerline, for the purpose of determining loadings on the tail in the absence of large unsymmetrical vorticity effects produced by the body and canard.

To expedite publication of these data, no detailed analyses of the results have been made. However, selected portions of the data are presented in graphical form and attention is directed to some of the important results of the investigation. The complete results of the investigation have been tabulated in the form of pressure coefficients and integrations of the pressure coefficients and are presented in two supplements to this Technical Note. Because of the limited interest in

the complete results and their bulk, the supplements will be made available only upon request to the NASA. (A request form for this purpose is bound in the back of this paper.)

NOTATION

- a lower limit of integration (0.208 for single vertical tail;
0.011 for twin vertical tail)
- b span of surface
- c local chord of surface
- c_{av} average chord of surface, $\frac{S}{b}$
- $\bar{c}$ mean aerodynamic chord of wing
- $c_{m_{LE}}$ pitching-moment coefficient of wing section referred to the
leading edge of section, $-\int_0^{1.0} (C_{p_l} - C_{p_u}) \left(\frac{x}{c}\right) d\left(\frac{x}{c}\right)$
- c_{m_W} pitching-moment coefficient of wing section referred to the
projection of the 0.21 $\bar{c}$ point on the body reference line,
 $c_{m_{LE}} + \left(\frac{\bar{x}}{c}\right) c_{N_W}$
- $c_{n_{LE}}$ yawing-moment coefficient of vertical-tail section referred to
to the leading edge of section, $\int_0^{1.0} (C_{p_R} - C_{p_L}) \left(\frac{x}{c}\right) d\left(\frac{x}{c}\right)$
- c_{n_V} yawing-moment coefficient of vertical-tail section referred to
the projection of the 0.21 $\bar{c}$ point on the body reference line,
 $c_{n_{LE}} + c_{Y_V} \left(\frac{\bar{x}}{c}\right)$
- c_{N_B} body-section coefficient of force normal to the body surface
in the vertical plane of symmetry, positive upward,
 $-\int_{0.05}^{1.05} C_p \sin \theta d\left(\frac{\theta}{2\pi}\right)$
- c_{N_W} normal-force coefficient of wing section, $\int_0^{1.0} (C_{p_l} - C_{p_u}) d\left(\frac{x}{c}\right)$

- c_{Y_B} body-section coefficient of force normal to the body surface in the lateral plane of symmetry, positive right,

$$-\int_{0.05}^{1.05} C_p \cos \theta \, d\left(\frac{\theta}{2\pi}\right)$$
- c_{Y_V} side-force coefficient of vertical-tail section,

$$\int_0^{1.0} \left(C_{p_L} - C_{p_R}\right) d\left(\frac{x}{c}\right)$$
- C_{b_V} bending-moment coefficient of vertical tail referred to the body center line, $\frac{S_V}{S_W} \frac{b_V}{b_W} \int_a^{1.0} \left(\frac{c}{c_{av}}\right) \left(\frac{y}{b}\right) c_{Y_V} d\left(\frac{y}{b}\right)$
- C_{b_W} bending-moment coefficient of wing referred to the body center line, $-\int_{0.175}^{1.0} \left(\frac{c}{c_{av}}\right) \left(\frac{y}{b/2}\right) c_{N_W} d\left(\frac{y}{b/2}\right)$
- C_{m_B} pitching-moment coefficient of body referred to the projection of the 0.21 $\bar{c}$ point on the body reference line,

$$\frac{\pi l^2}{S_W} \frac{l}{\bar{c}} \int_0^{1.0} \left(\frac{d}{l}\right) \left(\frac{x}{l}\right) c_{N_B} d\left(\frac{x}{l}\right)$$
- C_{m_W} pitching-moment coefficient of wing referred to the projection of the 0.21 $\bar{c}$ point on the body reference line,

$$\int_{0.175}^{1.0} \left(\frac{c}{\bar{c}}\right) \left(\frac{c}{c_{av}}\right) c_{m_W} d\left(\frac{y}{b/2}\right)$$
- C_{n_B} yawing-moment coefficient of body referred to the projection of the 0.21 $\bar{c}$ point on the body reference line,

$$\frac{\pi l^2}{S_W} \frac{l}{b_W} \int_0^{1.0} \left(\frac{d}{l}\right) \left(\frac{x}{l}\right) c_{Y_B} d\left(\frac{x}{l}\right)$$
- C_{n_V} yawing-moment coefficient of vertical tail referred to the projection of the 0.21 $\bar{c}$ point on the body reference line,

$$\frac{S_V}{S_W} \frac{b_V}{b_W} \int_a^{1.0} \left(\frac{c}{c_{av}}\right) \left(\frac{c}{b}\right) c_{n_V} d\left(\frac{y}{b}\right)$$
- C_p pressure coefficient, $\frac{p - p_\infty}{q_\infty}$
- C_{N_B} normal-force coefficient of body, $\frac{\pi l^2}{S_W} \int_0^{1.0} \left(\frac{d}{l}\right) c_{N_B} d\left(\frac{x}{l}\right)$

C_{N_W}	normal-force coefficient of wing, $\int_{0.175}^{1.0} \left(\frac{c}{c_{av}} \right) c_{N_W} d\left(\frac{y}{b/2} \right)$
C_{Y_B}	side-force coefficient of body, $\frac{\pi l^2}{S_W} \int_0^{1.0} \left(\frac{d}{l} \right) c_{Y_B} d\left(\frac{x}{l} \right)$
C_{Y_V}	side-force coefficient of vertical tail, $\frac{S_V}{S_W} \int_a^{1.0} \left(\frac{c}{c_{av}} \right) c_{Y_V} d\left(\frac{y}{b} \right)$
d	local body diameter
l	body length
l_0	length of body for closure at both ends
M	free-stream Mach number
p	local pressure on model
p_∞	free-stream static pressure
q_∞	free-stream dynamic pressure
r	local body radius
r_0	maximum body radius
S	area of surface including portion covered by the body
x	distance measured from leading edge of wing or tail surface or from body nose (positive rearward)
$\bar{x}$	distance measured from the wing or vertical-tail leading edge or a body station to the projection of the 0.21 $\bar{c}$ point on the body center line (positive rearward)
y	distance measured from body center line (positive away from body)
α	angle of attack of wing root chord, deg
β	angle of sideslip measured between relative wind and vertical plane of symmetry, deg

- δ angle of deflection of canard surface relative to wing plane
(positive when trailing edge is down), deg
- θ meridian angle on body, radians (measured clockwise from left
wing chord plane looking downstream)

Subscripts

- cp distance to section chordwise center of pressure
- l wing lower surface
- u wing upper surface
- V vertical tail
- B body
- L left side of vertical tail
- R right side of vertical tail
- W wing

Configurations are denoted by the following letters used in combination:

- B body
- C canard
- V single or twin vertical tails
- V₅ single vertical tail deflected 5° relative to the body center
line, leading edge left
- W wing

MODEL DESCRIPTION

The model consisted of a wing, an all-movable canard, a body and either a single vertical tail mounted on the body or twin vertical tails mounted on the wing panels. The wing and canard were both of triangular plan form and aspect ratio 2. The body was a Sears-Haack shape of fineness ratio 12.5. The vertical tails were of aspect ratio 1.35 and had swept

and tapered plan forms. The combined plan-form area of the twin vertical tails was equal to the plan-form area of the single vertical tail extended to the body center line.

The model was sting mounted in the wind tunnel as shown in the photographs of figures 1(a) and 1(b). Dimensional sketches of the complete model with the single vertical tail and the twin vertical tails are shown in figures 1(c) and 1(d), respectively, and the canard is detailed in figure 1(e). The wing and vertical tails had NACA 0003-63 sections streamwise and the canard consisted of a flat plate with beveled leading and trailing edges. The canard hinge line, passing through the 0.35 point of its mean aerodynamic chord, was located in the extended wing chord plane 1.21 wing mean aerodynamic chord lengths ahead of the reference center of moments (0.215). The ratio of the exposed area of the canard to the total area of the wing was 6.9 percent and the ratio of the total areas was 12.9 percent.

Pressure orifices of 0.30-inch diameter were installed flush with the model surface at 181 positions on the right wing panel, at 160 positions on the body, at 139 positions on the single vertical tail, and at 106 positions on the left-side twin vertical tail. The locations of these orifices are given in figure 2. The wing pressure lines were placed in grooves milled in the solid steel panel. These grooves were then filled with a plastic re-enforced with glass fiber which was faired to the wing ordinates. The vertical tails were also constructed of steel and re-enforced plastic. A steel spar to which the pressure lines were attached was wrapped with a cloth impregnated with plastic and then shaped to the specified coordinates. The hollow steel sting used to mount the model was machined integral with the body and served as a conduit for all the pressure tubes. The canard, wing, and vertical tails were removable and provision was also made for deflecting the single vertical tail to 5° , leading edge left with respect to the body center line.

TESTS AND PROCEDURES

Ranges of Test Variables

Table I lists the various configurations tested. Mach numbers of 0.70, 0.90, 0.95, 1.00, 1.05, 1.10, 1.30, 1.50, 1.70, 1.90, and 2.22 were covered in the investigation. Data were obtained at angles of sideslip of 0° and 5.3° for nominal angles of attack ranging from -4° to $+16^\circ$ at Mach numbers of 0.70, 1.30, and 2.22. All the data for the remaining Mach numbers were obtained at 0° sideslip and a nominal angle of attack of 4° . Canard deflection angles were nominally 0° and 10° . A limited amount of data was obtained with the vertical tail deflected 5° , leading edge left with respect to the body center line. The test Reynolds number based on the wing mean aerodynamic chord was 2.9×10^6 .

It should be noted that boundary-layer transition was not induced on the model as was done for the force measurements (refs. 2 and 3). However, for the test conditions being considered the effects of artificially inducing transition are generally small and are primarily related to changes in model friction drag.

Reduction of Data

The pressures were measured on multitube manometers filled with tetrabromoethane (specific gravity = 2.97 at 70° F) and were photographically recorded on film. The film was read and the data were recorded on punchcards. Automatic digital computing machines reduced the data from the punchcards to the final coefficients.

Force and moment coefficients were obtained from integrations of the pressure coefficients. To perform the various integrations, it was necessary to extrapolate certain quantities; for instance, because of physical limitations, no orifices could be located on the wing or vertical tails at values of $x/c = 0$ or 1.0. Therefore, to provide consistent results for the wing and tail section coefficients, the values of C_p at the leading and trailing edges were obtained mathematically by passing a second degree curve through the three experimental data points nearest the unknown value and extrapolating to the desired value of x/c . The extrapolated pressure coefficients for x/c values of 0 and 1.0 on the wing and vertical tails have been included in the tabulated data. The section coefficients were then obtained from the pressure coefficients integrated in the following manner. A second degree curve was fitted to the first three pressure coefficients for a section and the area under the curve through these three points was computed analytically. Then a second degree curve was fitted to the second, third, and fourth pressure coefficients and the area under the curve between the third and fourth points was integrated analytically. This process was repeated a sufficient number of times to include all the necessary data points, and the summation of the partial areas resulted in the total area or section coefficient. The same general scheme was also used in computing the total coefficients. First the necessary section coefficient data were extrapolated and then the integrations were performed. The extrapolations of section coefficient data for the wing and single vertical tail were simplified by considering the body to be cylindrical in the vicinity of the wing-body and tail-body junctures. The diameter of the cylinder was computed by stipulating that the cylindrical section of body was to have the same volume as the section of body it replaced. The extrapolated section coefficient data have also been included in the tabulated data. The total coefficients of the body and vertical tails are based on wing area, and the wing-panel total coefficients are based on wing-panel area.

The angles of attack have been adjusted for wind-tunnel stream inclinations in the model pitch plane which were less than $\pm 0.3^\circ$ at all Mach numbers. No corrections to model sideslip angle were applied for wind-tunnel stream angularities in the lateral plane.

The pitching- and yawing-moment coefficients for each component part of the model have been referred to the projection of the wing 0.21 $\bar{c}$ point on the body center line. This particular moment reference center was chosen to be consistent with the force results for the same configuration reported in references 2 and 3.

Accuracy

All the data tabulated herein were plotted by an automatic machine utilizing punchcards. The plots were then examined for data points obviously in error and corrections were made whenever possible. It is therefore believed that any erroneous data points have either been corrected or removed and that the following estimates of accuracy apply to all of the data presented.

M	± 0.005
α and δ , deg	± 0.10
β , deg	± 0.30
C_p	± 0.005

RESULTS AND DISCUSSION

Presentation of Results

All of the experimental results for this investigation are tabulated in the form of pressure coefficients and integrations of the pressure coefficients in two supplements to this paper. The supplements, designated as NASA TN D-690-I and D-690-II, present the results for the single-tail and twin-tail configurations, respectively. Limited amounts of the data have been selected for presentation in graphical form in this paper and are presented in figures 3 through 12. The effects illustrated by these plotted data are typical of those at other Mach numbers and angles of attack and, therefore, can be used to focus attention on some of the results of the investigation.

Discussion

Loads on the wing.- The chordwise distributions of pressure at several spanwise stations of the wing for the single-tail and twin-tail

configurations, both with and without the canard, are shown in figures 3 and 4, respectively. These data are presented for a sideslip angle of 0° and a nominal angle of attack of 8° . The results in figures 3 and 4 show that, as might be expected, the canard wake influences the chordwise distributions of pressure differently at different spanwise stations on the wing. At the two most inboard stations, for example, the effects of canard interference are concentrated almost entirely forward of the midpoints of the local chords while on stations further outboard the effects tend to be distributed over the full length of each local chord. However, even though local pressure disturbances due to canard induced interference are in some cases greater at outboard stations on the wing than at inboard stations, the most pronounced effects on the span-load distribution occur in the inboard region of the wing where the chord lengths are large. This effect is illustrated by the results of figures 5 and 6 wherein the wing span-load distributions at several angles of attack for the model with and without the canard are presented for both the single- and twin-tail configurations.

The results of figures 5 and 6 show, for a constant angle of attack, that the interference of the canard wake with the wing generally has the effect of reducing the spanwise loading over most of the wing with the most severe effects, as pointed out previously, occurring at inboard stations. When the canard is deflected and the model is at low angles of attack, the relatively strong canard trailing vorticity is positioned near the surface of the wing. Under these circumstances, the change in loading inboard of a line drawn parallel to the body center line from the canard tip to the wing is particularly large; for example, refer to figure 5(a) ($\alpha = 3.7^\circ$, $\delta = 9.6^\circ$) and figure 6(a) ($\alpha = 3.8^\circ$, $\delta = 10.2^\circ$).

A comparison of the spanwise load distributions of the single-tail model (fig. 5) with those of the twin-tail model (fig. 6) shows that at the supersonic Mach numbers, $M = 2.22$ and 1.30 , the wing loading characteristics up to angles of attack of 16° with the canard off or on are not significantly affected by mounting the twin tails on the wing panels. At the subsonic Mach number of 0.70 , however, this is true only up to an angle of attack of about 8° . At the higher angles of attack of approximately 12° and 16° , the presence of the twin tails on the wing panels causes the span loading to be reduced. This reduction in loading is particularly large on the outboard portions of the wing when the canard is off. Thus, at these angles of attack, the effect of adding the canard is to increase the loading on the wing outboard of the vertical tail.

Loads on the body.—Coefficients of force normal to the body surface in the vertical plane of symmetry are presented as a function of body length in figures 7 and 8 for the single- and twin-tail configurations, respectively. These data are shown for 0° sideslip and a nominal angle of attack of 8° . For the single-tail configuration without the wing (fig. 7(a)), the effects of canard interference are confined to the forward 40 percent of the body throughout the Mach number range investigated. Most of these effects can be attributed to a direct carry-over

of lifting pressure from the canard to the body rather than interference of the canard wake with the body. For the single-tail configuration with the wing (fig. 7(b)), the effects of canard-induced interference are appreciable over a much larger region of the body. The interference on the forward portion of the body is much the same as with the wing off. However, the additional effects in the vicinity of the wing might be expected in view of the previously mentioned canard-wing interference. It will be remembered that at a given angle of attack the canard causes the wing loading to be reduced, especially near the wing-body juncture. This, in turn, reduces the amount of carry-over loading from the wing to the body. The results for the twin-tail configuration with the wing on, presented in figure 8, are very similar to those of the single-tail configuration (fig. 7(b)).

Loads on the vertical tails.- Chordwise distributions of pressure on several spanwise stations of the single tail and the left-side twin tail are presented in figures 9 and 10, respectively, for 5.3° sideslip and a nominal angle of attack of 8° both with and without the canard. The canard interference effects on the single tail (fig. 9) are, in general, distributed over the full chord length at each spanwise station. Furthermore, the canard-induced disturbances in the local pressures on the single vertical tail are generally smaller than those measured on the wing. The results of figure 10 for the left-side twin tail also show that the canard interference effects are distributed over the full chord length at each spanwise station. However, in contrast to the results for the single tail, the canard-induced changes in the local pressures on the left-side twin tail are quite large.

The relative location of the vertical tails with respect to the canard vortex wake would be expected to influence the spanwise distribution of canard-induced interference effects. This is, indeed, verified by the results of figures 11 and 12 which present the spanwise load distributions on the single tail and left-side twin tail, respectively, resulting from the chordwise pressure distributions shown in figures 9 and 10.

With the canard in the undeflected position, the lower region of the single tail is in a destabilizing induced flow field while the upper portion is in a relatively weaker stabilizing flow field. Deflection of the canard results in a downward movement of its wake with respect to the vertical tail and, in this case, a greater portion of the tail span is in the stabilizing flow field. In fact, the results of figure 11 indicate that for the conditions of α , β , and δ selected to illustrate the effects of the deflected canard, practically all of the single vertical tail is in a stabilizing induced flow field.

Comparison of the results of figure 12 for the left-side twin tail with those for the single tail (fig. 11) reveals that the wing-mounted tail is not very effective as a stabilizing surface when the canard is

off. When the canard is added at 0° deflection, the loading on the twin tail is increased considerably over the half of the tail nearest the wing. However, no significant change in the loading on the outboard half of the twin tail is noted. Increasing the strength of the canard trailing vorticity by deflecting the canard produces further large increases in the loading on the lower portion of the twin tail. However, the region of the tail experiencing the increased loading due to canard deflection is somewhat smaller than the portion influenced by the undeflected canard. This can probably be attributed to the fact that as the canard is deflected, the canard wake moves downward with respect to the tail.

Ames Research Center

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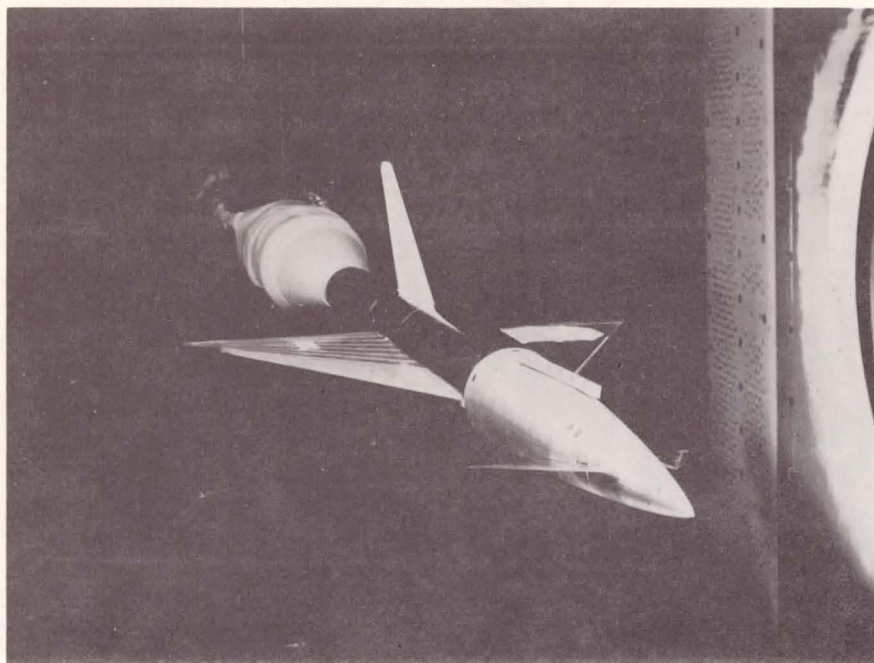
REFERENCES

1. Hall, Charles F., and Boyd, John W.: Effects of Canards on Airplane Performance and Stability. NACA RM A58D24, 1958.
2. Boyd, John W., and Peterson, Victor L.: Static Stability and Control of Canard Configurations at Mach Numbers From 0.70 to 2.22 - Longitudinal Characteristics of a Triangular Wing and Canard. NACA RM A57J15, 1958.
3. Peterson, Victor L., and Menees, Gene P.: Static Stability and Control of Canard Configurations at Mach Numbers From 0.70 to 2.22 - Lateral-Directional Characteristics of a Triangular Wing and Canard. NACA RM A57J18, 1958.
4. Driver, Cornelius, and Jacocks, James L.: Tabulated Data From a Pressure-Distribution Investigation at Mach Number 2.01 of the Wing of a Canard Airplane Model. NASA TM X-140, 1959.

TABLE I.- CONFIGURATIONS TESTED

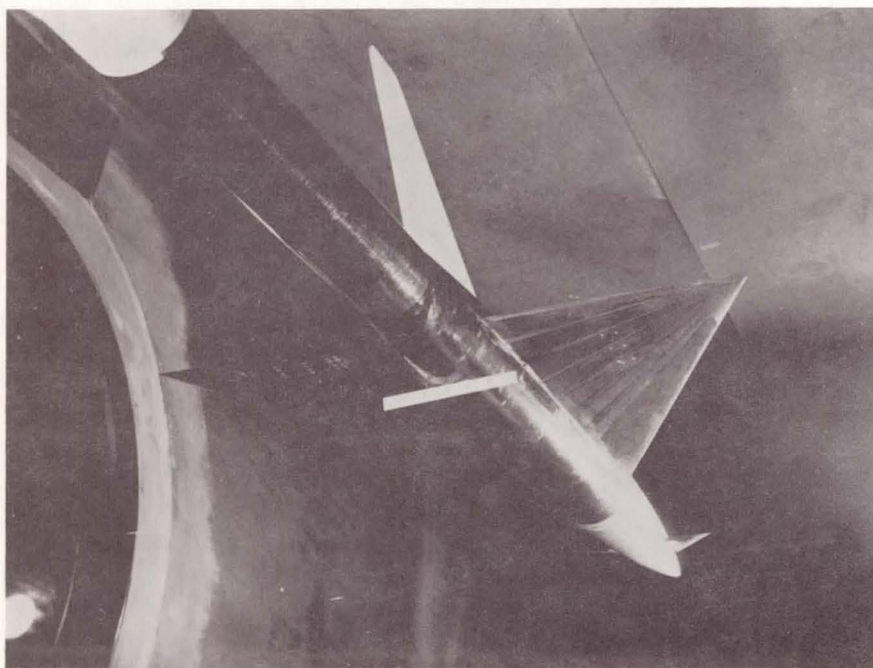
Single-tail model			Twin-tail model		
Configuration	δ , deg	β , deg	Configuration	δ , deg	β , deg
BVW		0 5.3	BVW		0 5.3
BVWC	-0.4 0.5	0 5.3	BVWC	-0.5 0.5	0 5.3
BVWC	9.6 9.9	0 5.3	BVWC	10.2 10.7	0 5.3
BV ₅ W		0			
BV ₅ WC	0.4	0			
BW		0	BW		0
BV		0			
BVC	-0.1	0			
BVC	9.7	0			
BV ₅		0			
BV ₅ C	-0.1	0			
B		0 5.3			

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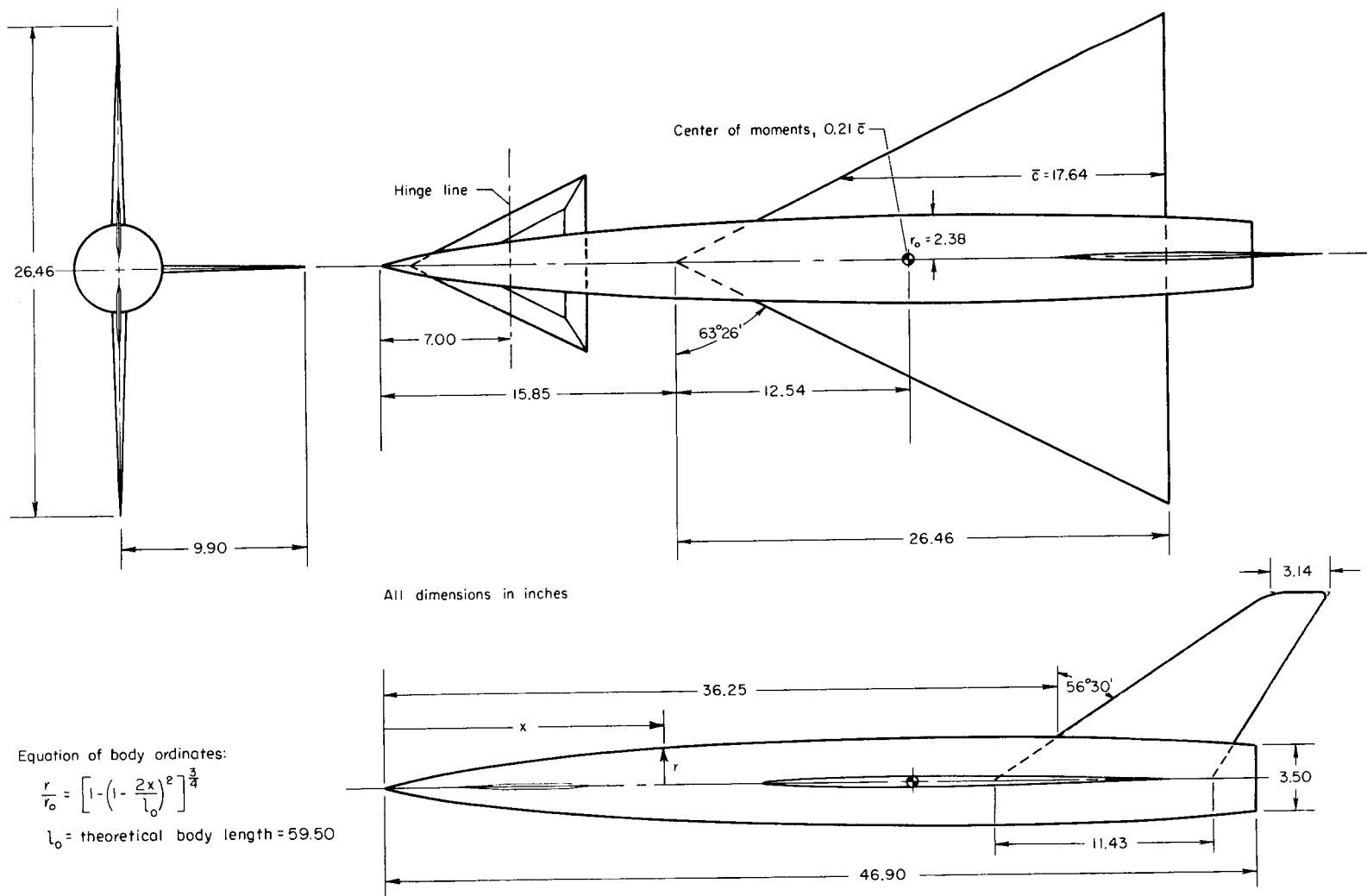
(a) Three-quarter front view of the single-tail model mounted in the wind tunnel.



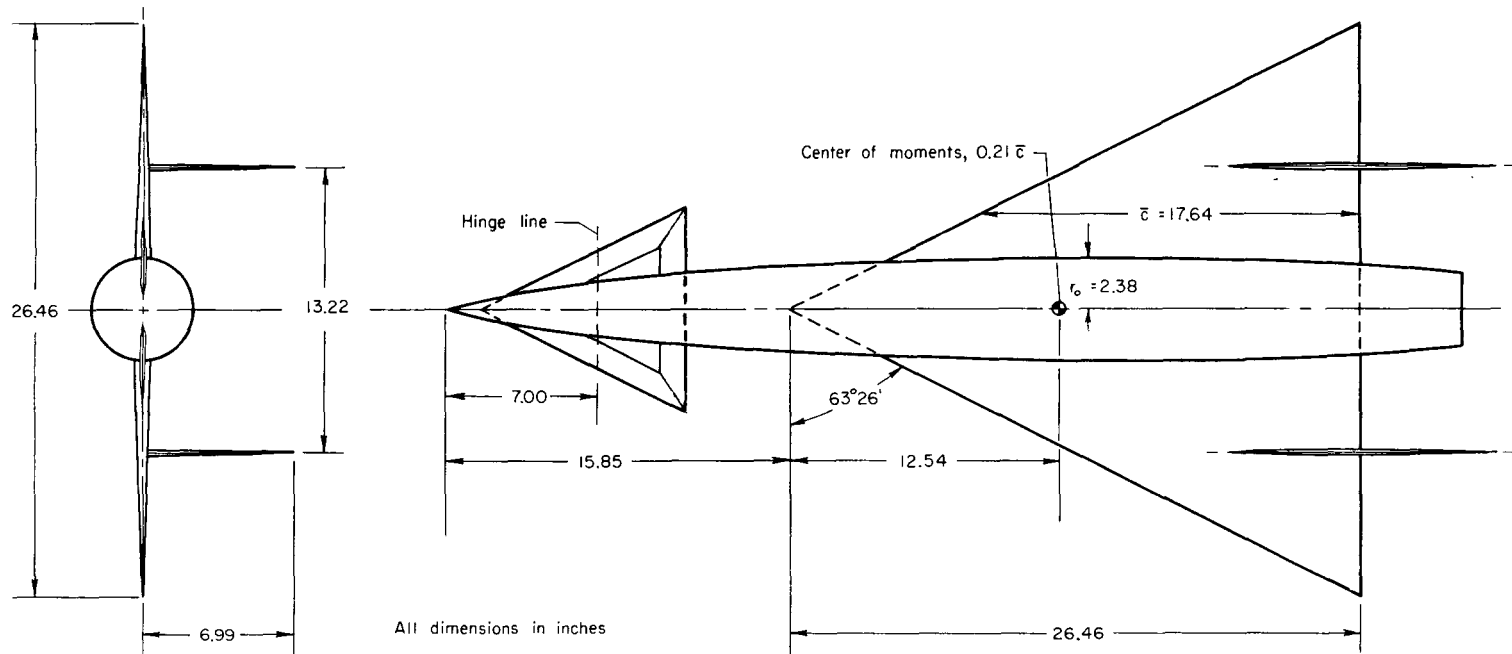
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(b) Three-quarter rear view of the single-tail model mounted in the wind tunnel.

Figure 1.- Model details and dimensions.



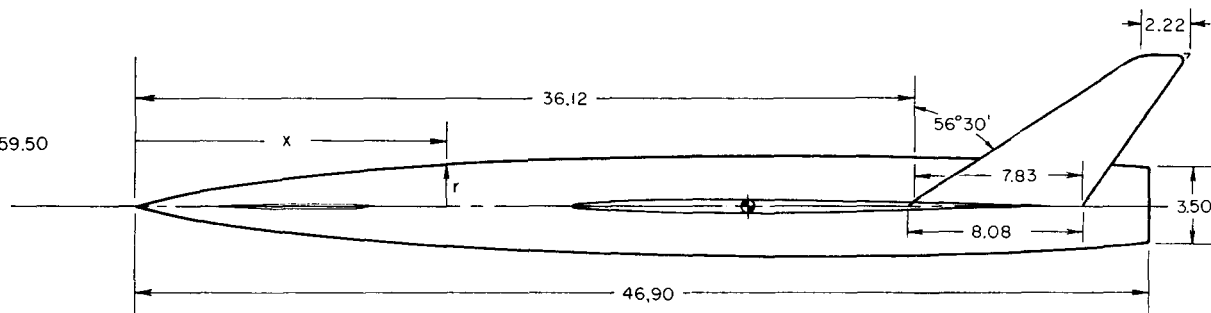
(c) Dimensional sketch of the single-tail model.



Equation of body ordinates:

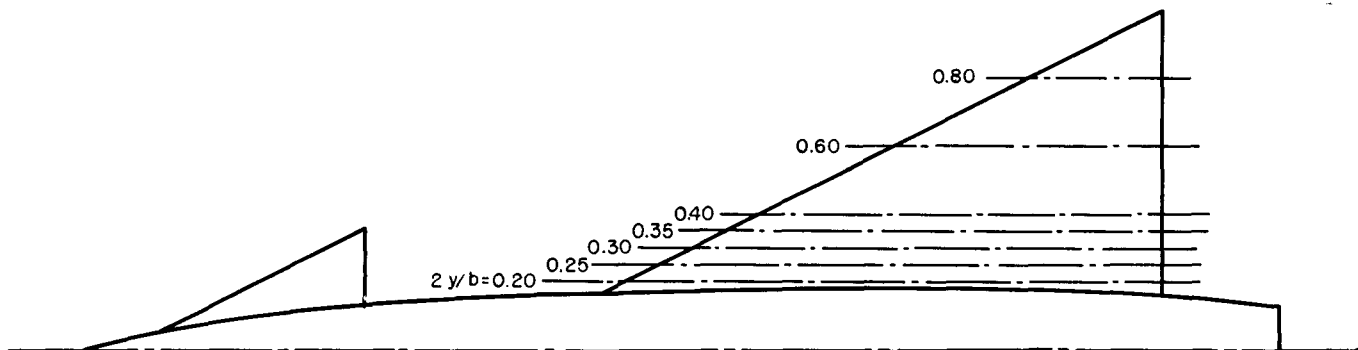
$$\frac{r}{r_o} = \left[1 - \left(\frac{2x}{l_o} \right)^2 \right]^{\frac{3}{4}}$$

l_o = theoretical body length = 59.50



(d) Dimensional sketch of the twin tail model.

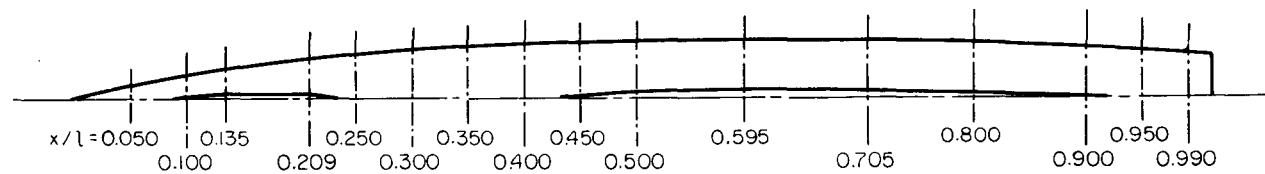
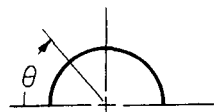
Figure 1.- Continued.



Wing orifice locations													
$2y/b = 0.20$		$2y/b = 0.25$		$2y/b = 0.30$		$2y/b = 0.35$		$2y/b = 0.40$		$2y/b = 0.60$		$2y/b = 0.80$	
$(x/c)_u$	$(x/c)_l$	$(x/c)_u$	$(x/c)_l$	$(x/c)_u$	$(x/c)_l$	$(x/c)_u$	$(x/c)_l$	$(x/c)_u$	$(x/c)_l$	$(x/c)_u$	$(x/c)_l$	$(x/c)_u$	$(x/c)_l$
0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	
.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250	
.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	0.0420
.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750	
.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.0640
.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	
.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.0810
.3000	.3000	.3000	.3000	.3000	.3000			.3000	.3000	.3000	.3000	.3000	.1000
.4000	.4000	.4000	.4000	.4000	.4000			.4000	.4000	.4000	.4000	.4000	.1500
.5000	.5000	.5000	.5000	.5000	.5000			.5000	.5000	.5000	.5000	.5000	.2000
.6000	.6000	.6000	.6000	.6000	.6000			.6000	.6000	.6000	.6000	.6000	.3000
.7000	.7000	.7000	.7000	.7000	.7000			.7000	.7000	.7000	.7000	.7000	.4000
.8000	.8000	.8000	.8000	.8000	.8000			.8000	.8000	.8000	.8000	.8000	.5000
.9000	.9000	.9000	.9000	.9000	.9000			.9000	.9000	.9000	.9000	.9000	.6000
												.7000	.7000
												.8000	.8000
												.9000	.9000

(a) Wing.

Figure 2.- Locations of pressure orifices.



At each longitudinal body station orifices
are located peripherally as follows:

$\theta/2\pi = 0.05$
 .15
 .25
 .35
 .45
 .55
 .65
 .75
 .85
 .95

(b) Body.

Figure 2.- Continued.



Vertical-tail orifice locations									
$y/b = 0.25$		$y/b = 0.40$		$y/b = 0.55$		$y/b = 0.70$		$y/b = 0.85$	
$(x/c)_L$	$(x/c)_R$	$(x/c)_L$	$(x/c)_R$	$(x/c)_L$	$(x/c)_R$	$(x/c)_L$	$(x/c)_R$	$(x/c)_L$	$(x/c)_R$
0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125	0.0125
.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250	.0250
.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500
.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750	.0750
.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000
.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500
.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000
.3000	.3000	.3000	.3000	.3000	.3000	.3000	.3000	.3000	.3000
.4000	.4000	.4000	.4000	.4000	.4000	.4000	.4000	.4000	.4000
.5000	.5000	.5000	.5000	.5000	.5000	.5000	.5000	.5000	.5000
.6000	.6000	.6000	.6000	.6000	.6000	.6000	.6000	.6000	.6000
.7000	.7000	.7000	.7000		.6170	.7000	.7000	.7000	
.8000	.8000	.8000	.8000	.7000	.7000	.8000	.8000		.7100
.9000	.9000	.9000	.9000	.8000	.8000	.9000	.9000	.8000	
				.9000	.9000				.8100
								.9000	.9000

(c) Single vertical tail.

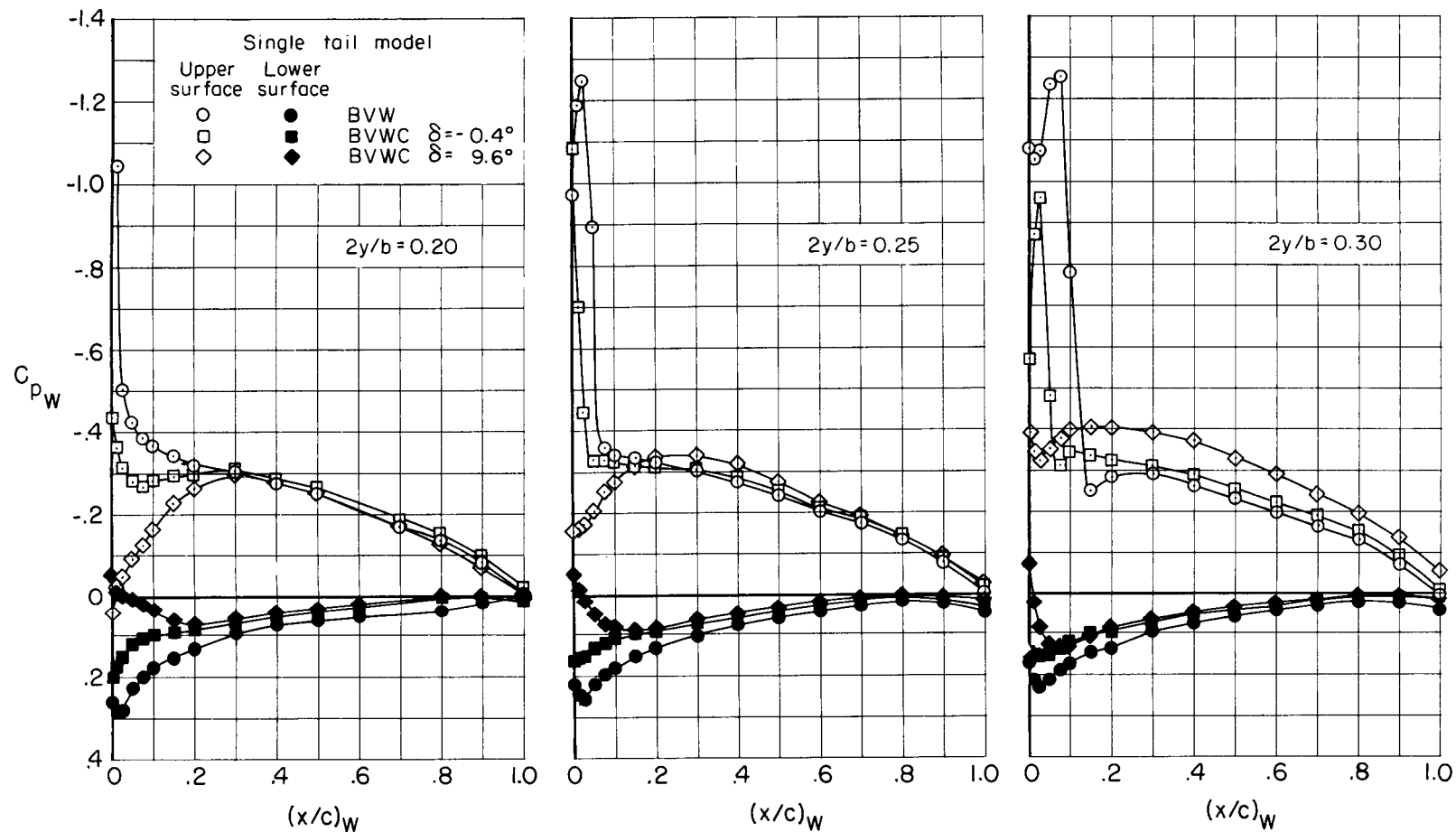
Figure 2.- Continued.



Vertical-tail orifice locations									
y/b = 0.15		y/b = 0.30		y/b = 0.45		y/b = 0.60		y/b = 0.75	
(x/c) _L	(x/c) _R	(x/c) _L	(x/c) _R	(x/c) _L	(x/c) _R	(x/c) _L	(x/c) _R	(x/c) _L	(x/c) _R
0.0250	0.0250	0.0250	0.0250	0.0250	0.0250	0.0250	0.0250		
.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500	.0500
.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000	.1000
.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500	.1500
.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000	.2000
.3000	.3000	.3000	.3000	.3000	.3000	.3000	.3000	.3000	.3000
.4000	.4000	.4000	.4000	.4000	.4000	.4000	.4000	.4000	.4000
.5000	.5000	.5000	.5000	.5000	.5000	.5000	.5000	.5000	.5000
.6000	.6000	.6000	.6000	.6000	.6000	.6000	.6000	.6000	.6000
.7000	.7000	.7000	.7000	.7000	.7000	.7000	.7000	.7000	.7000
.8000	.8000	.8000	.8000	.8000	.8000	.8000	.8000		

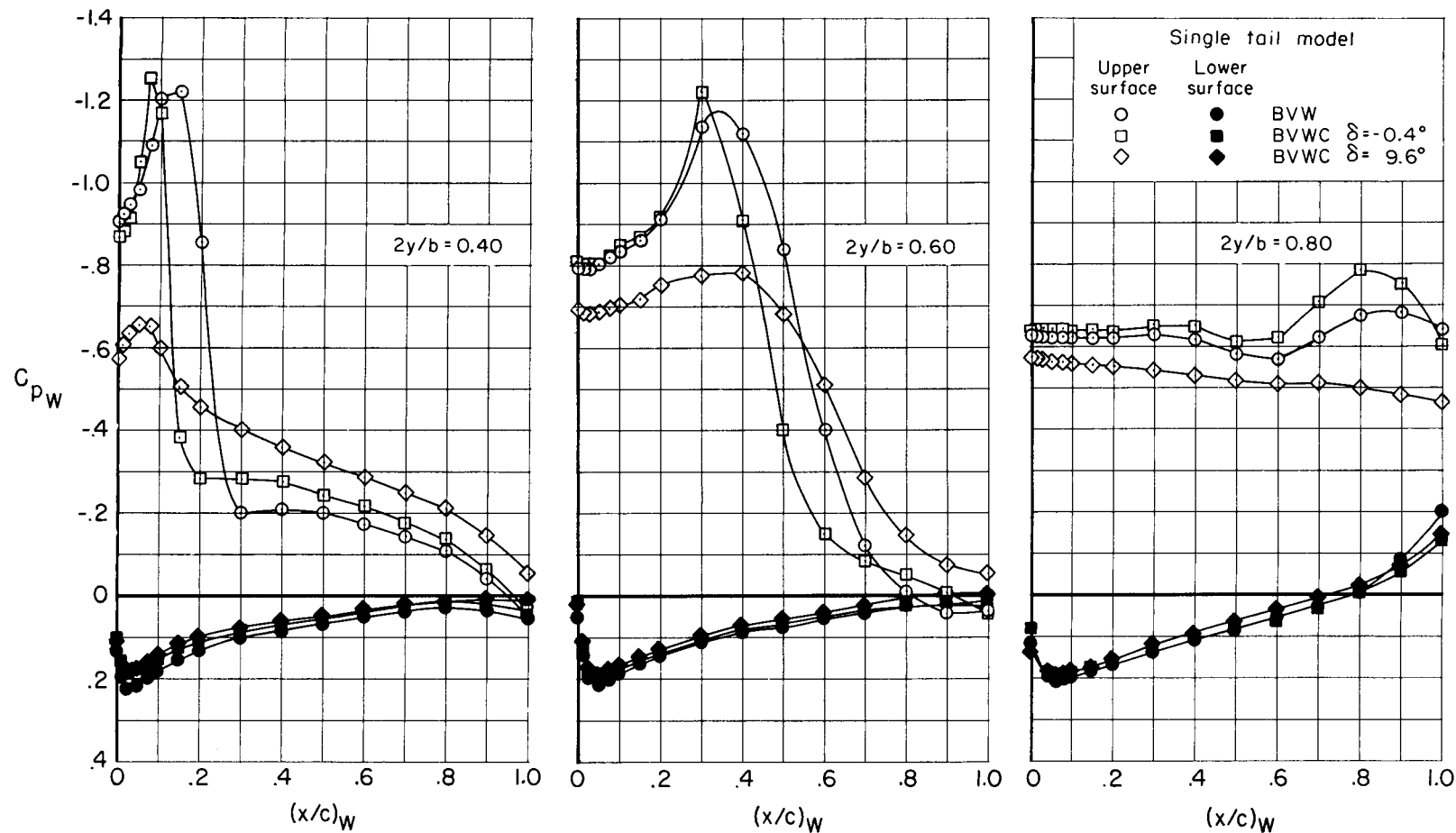
(d) Twin vertical tail.

Figure 2.- Concluded.



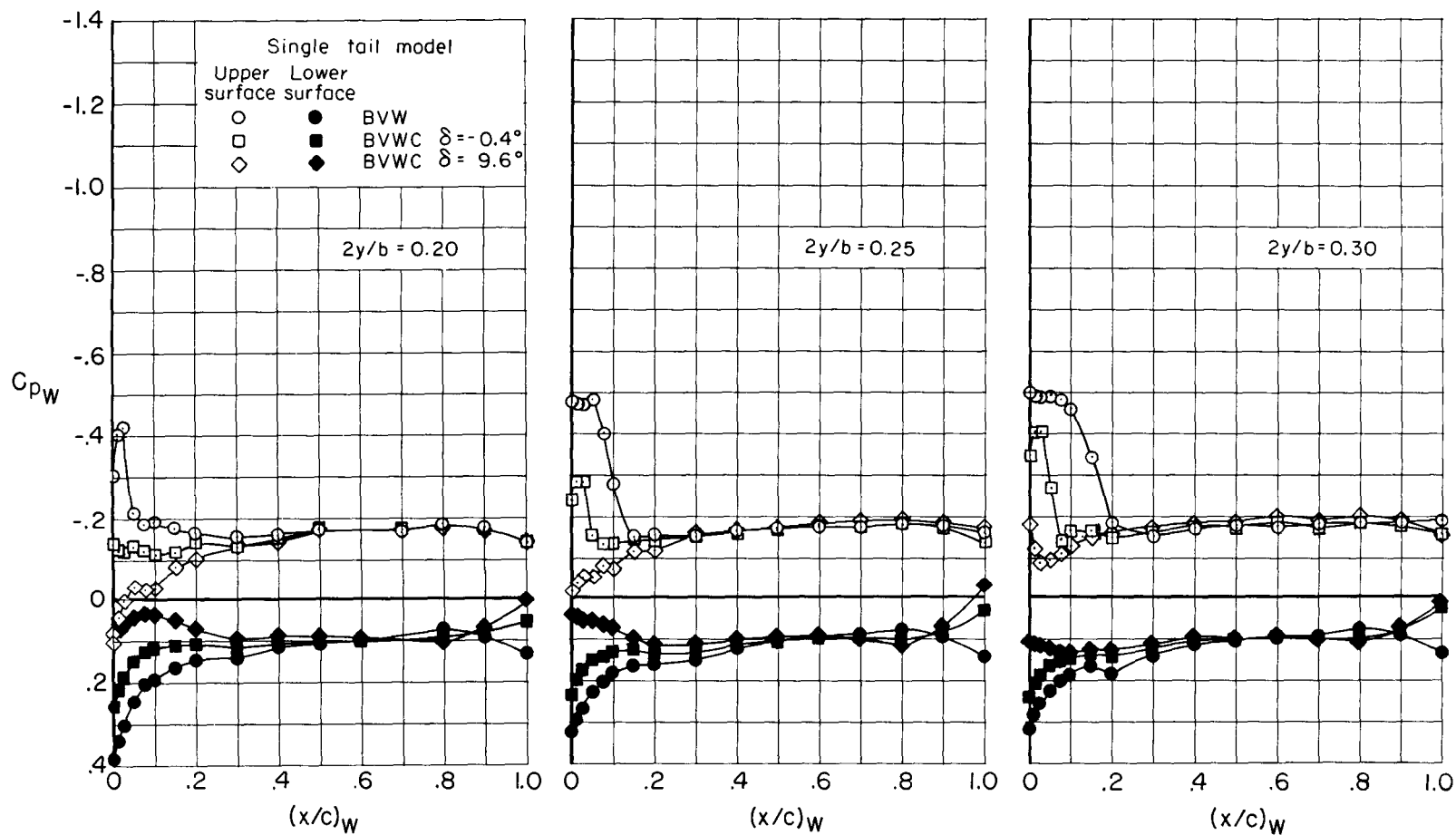
(a) $M = 0.70$, $\alpha = 7.7^\circ$

Figure 3.- Chordwise distributions of pressure on the wing of the model with the single vertical tail; $\beta = 0^\circ$. Values at $(x/c)_w$ of 0 and 1.0 were obtained by extrapolation.



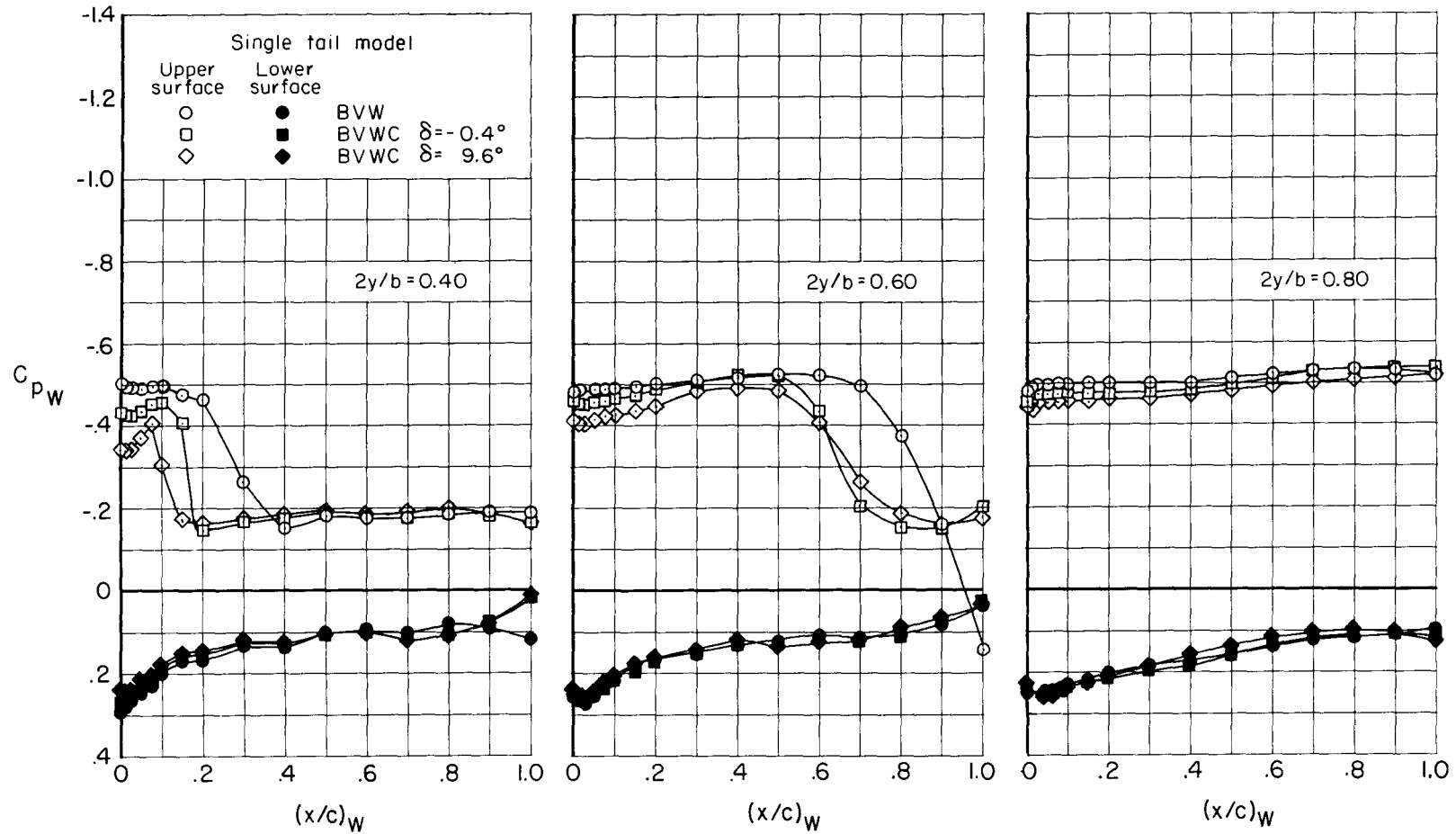
(a) $M = 0.70$, $\alpha = 7.7^\circ$ - Concluded.

Figure 3.- Continued.



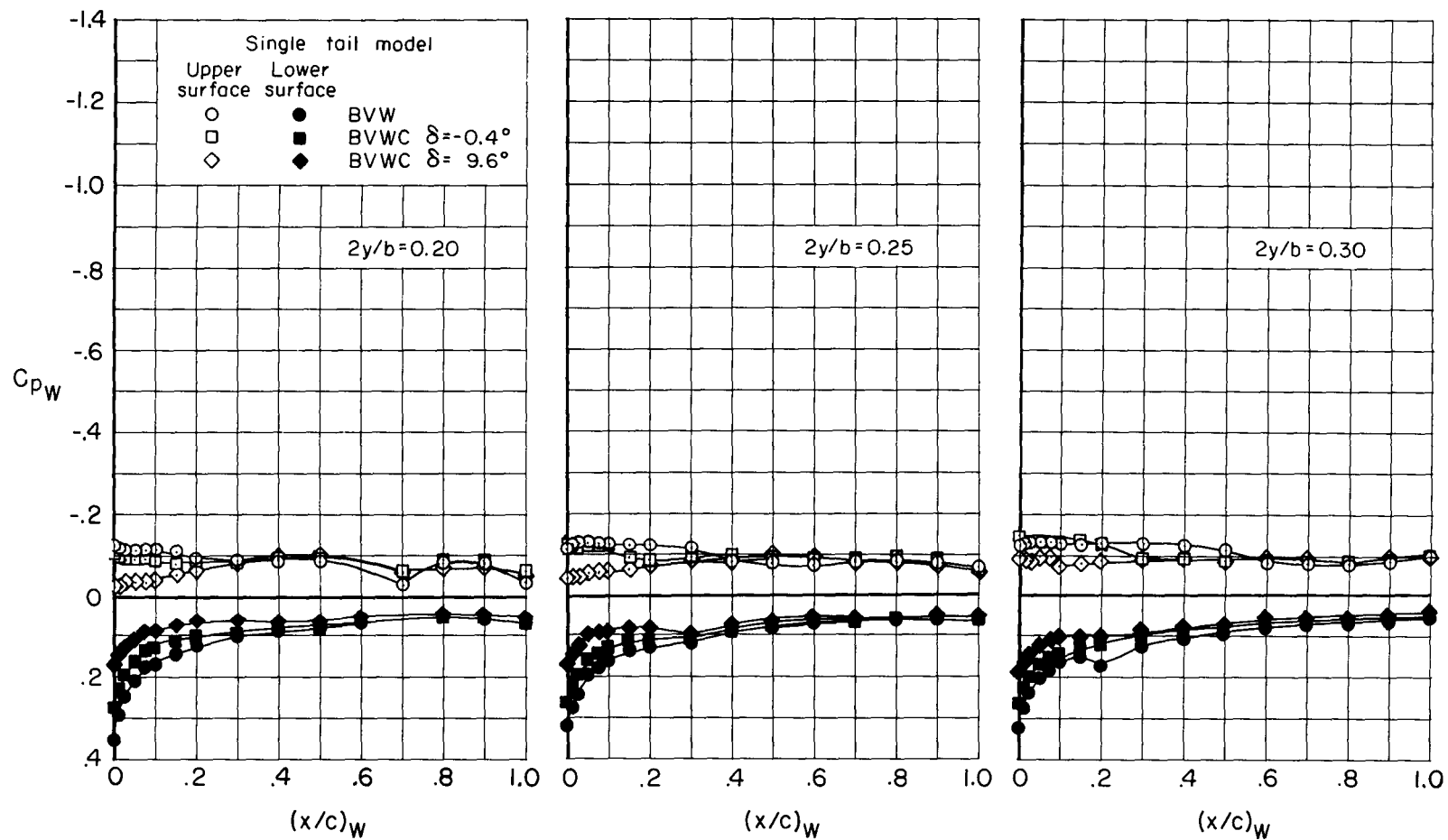
(b) $M = 1.30$, $\alpha = 7.8^\circ$

Figure 3.- Continued.



(b) $M = 1.30$, $\alpha = 7.8^\circ$ - Concluded.

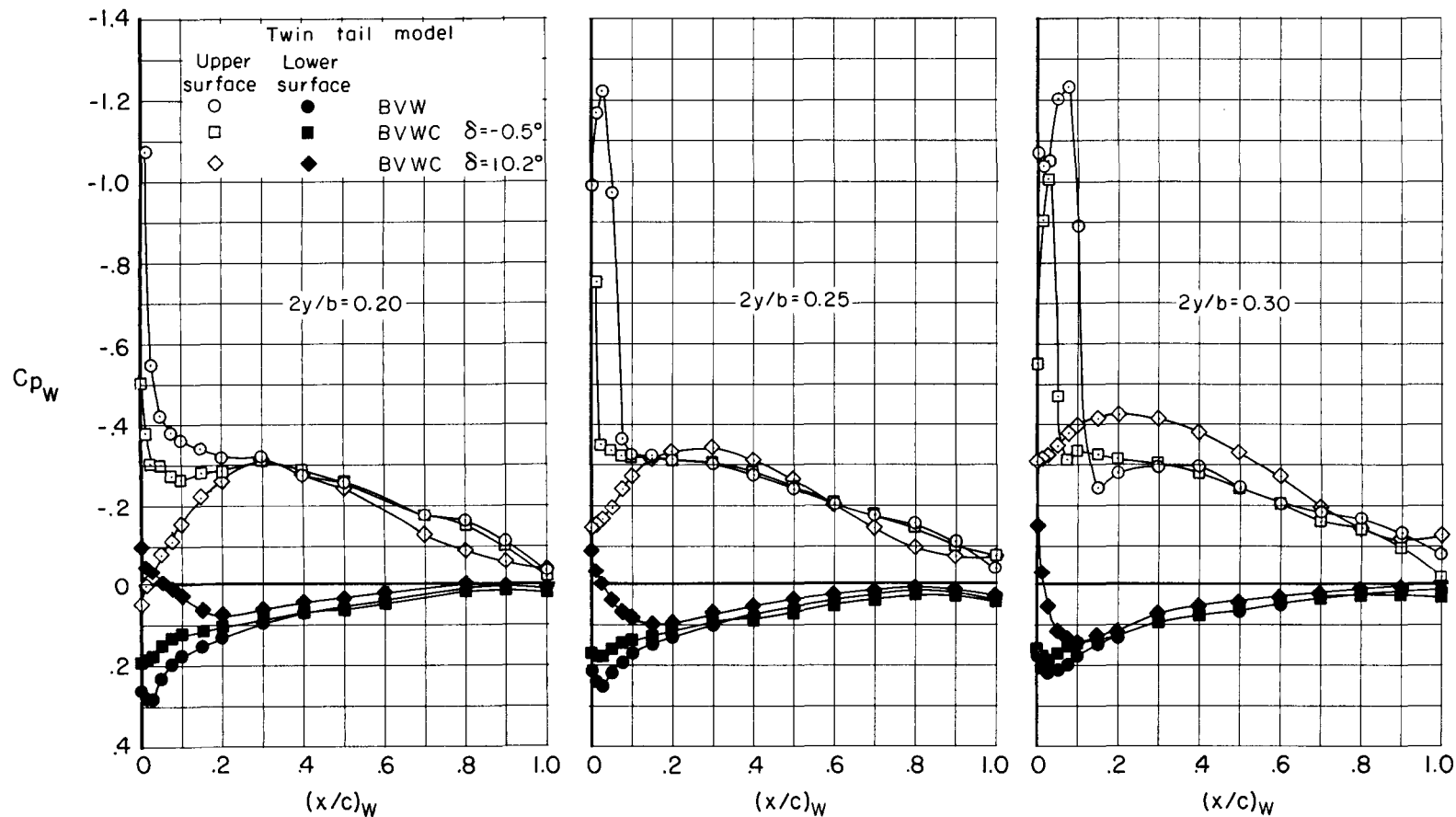
Figure 3.- Continued.



(c) $M = 2.22$, $\alpha = 8.3^\circ$

Figure 3.- Continued.

Figure 3.- Concluded.



(a) $M = 0.70$, $\alpha = 7.7^\circ$

Figure 4.- Chordwise distributions of pressure on the wing of the model with the twin vertical tails; $\beta = 0^\circ$. Values at $(x/c)_w$ of 0 and 1.0 were obtained by extrapolation.

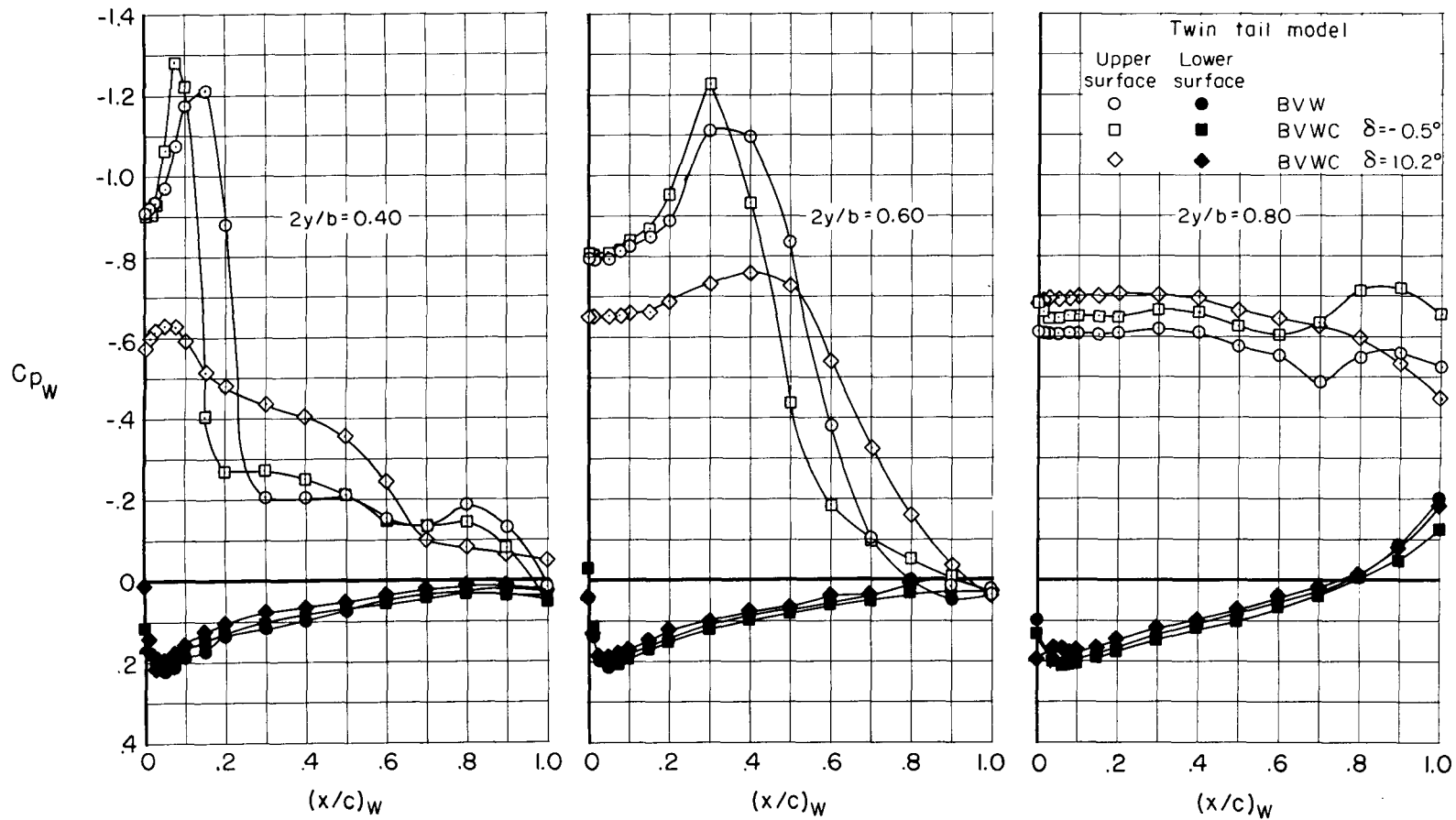
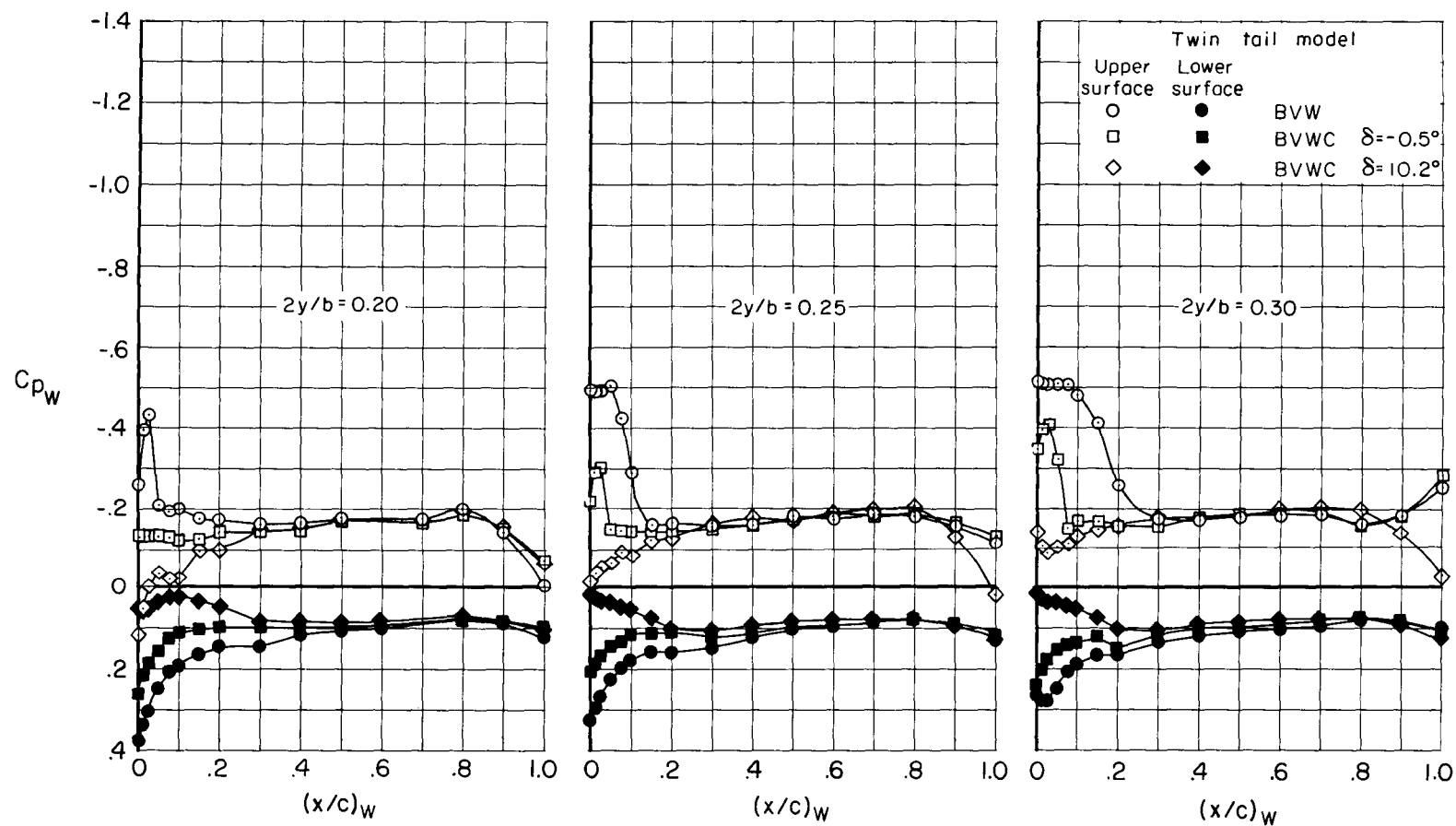
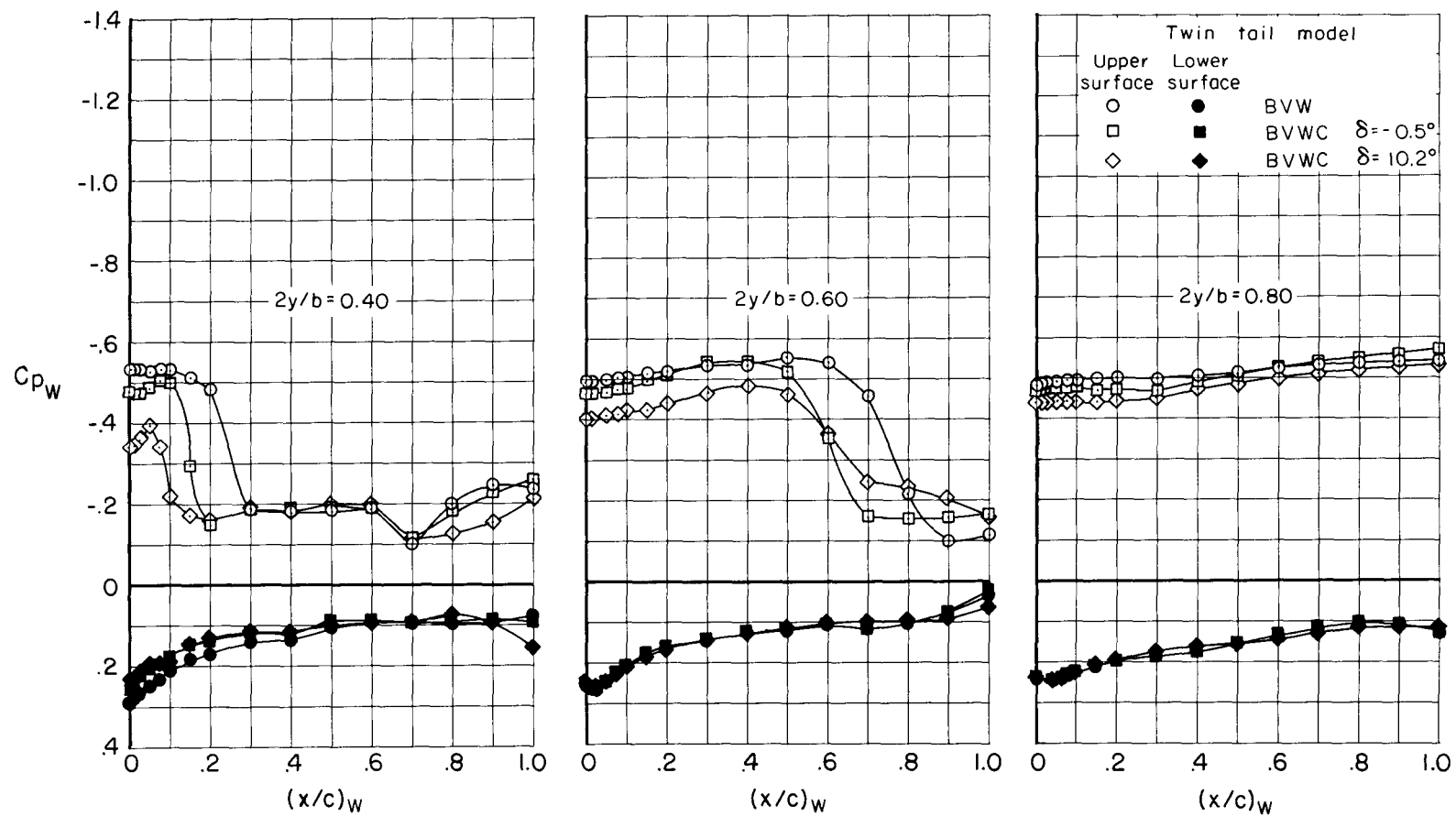
(a) $M = 0.70$, $\alpha = 7.7^\circ$ - Concluded.

Figure 4.- Continued.



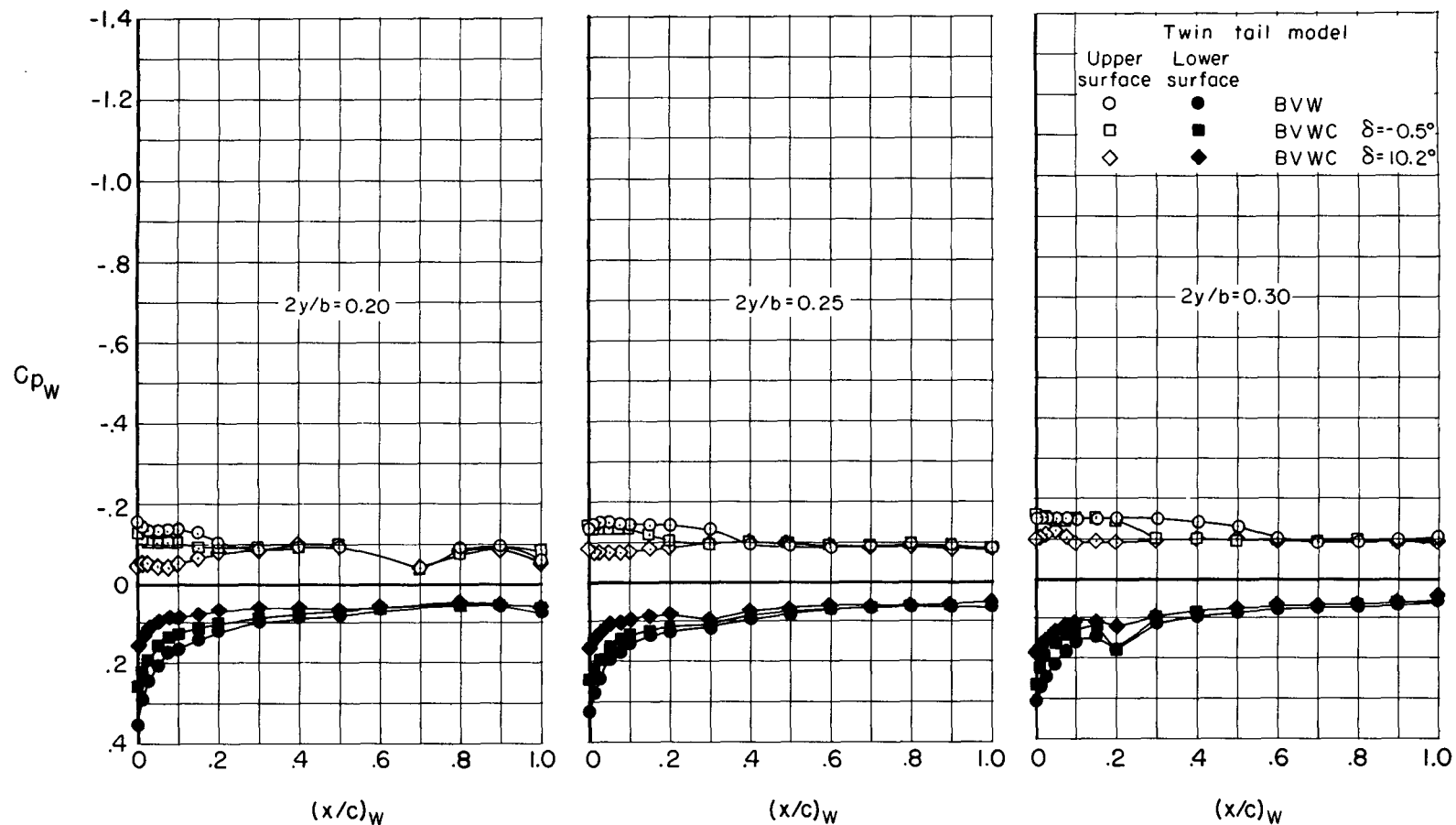
(b) $M = 1.30$, $\alpha = 7.9^\circ$.

Figure 4.- Continued.



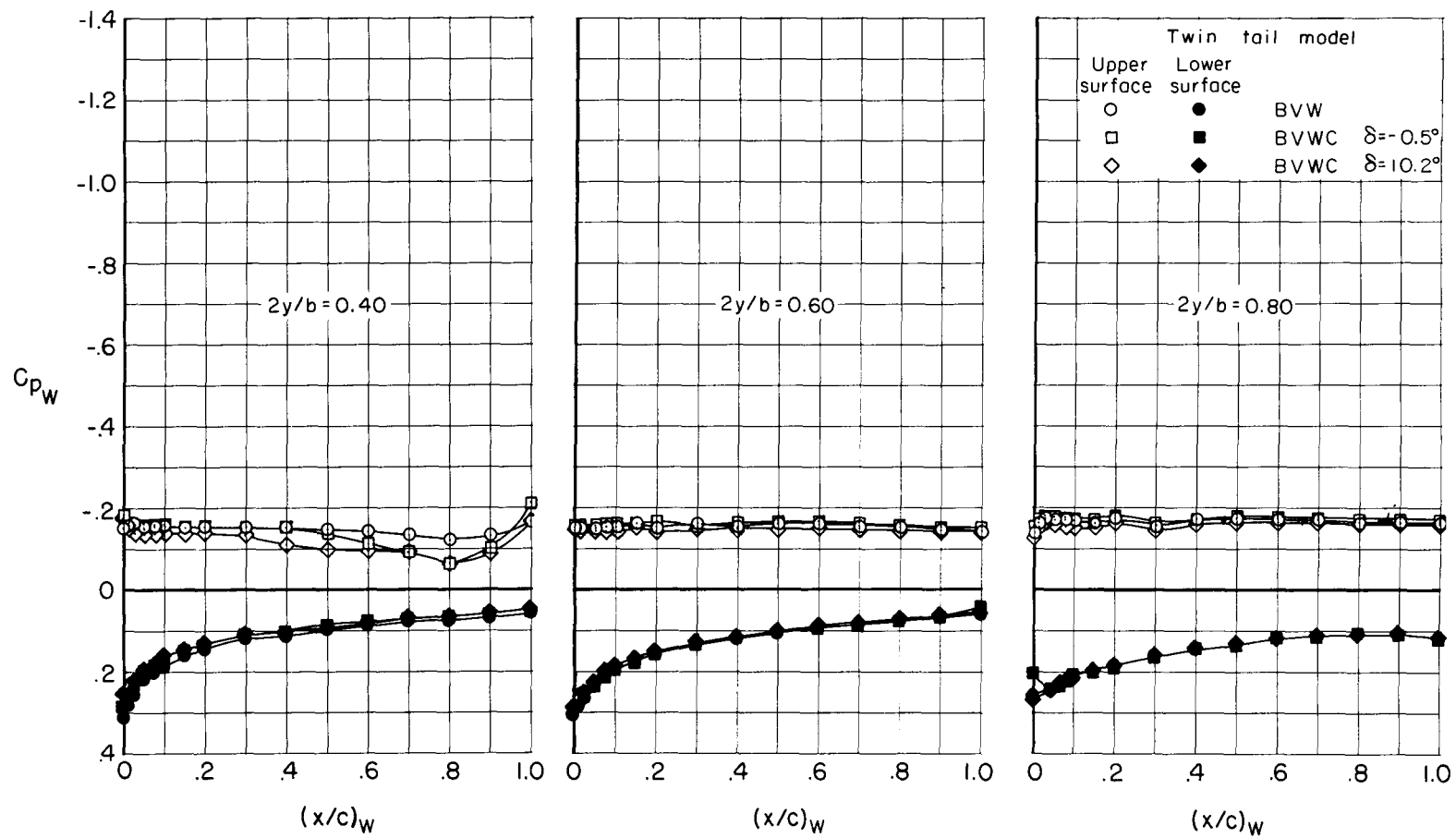
(b) $M = 1.30$, $\alpha = 7.9^\circ$ - Concluded.

Figure 4.- Continued.



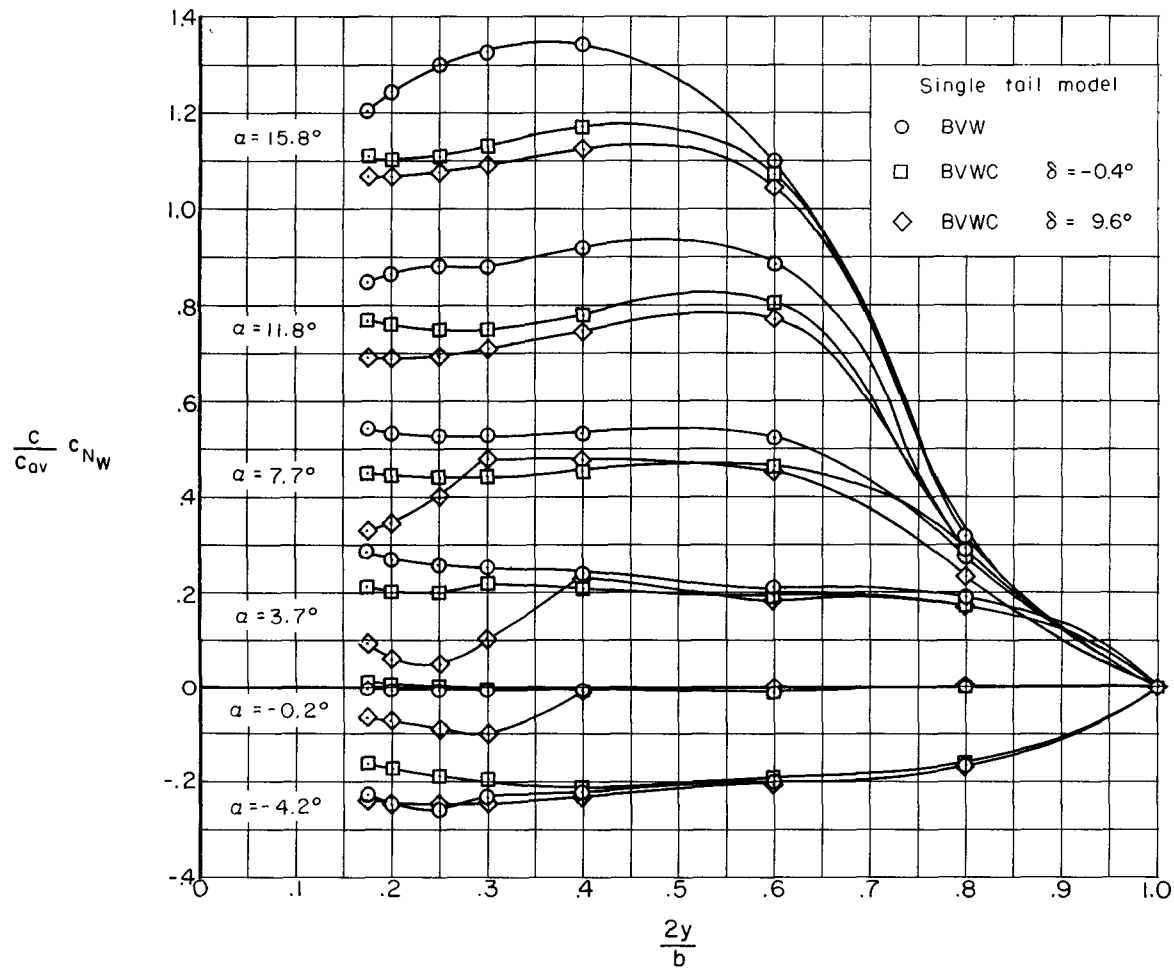
(c) $M = 2.22$, $\alpha = 8.2^\circ$

Figure 4.- Continued.



(c) $M = 2.22$, $\alpha = 8.2^\circ$ - Concluded.

Figure 4.- Concluded.



(a) $M = 0.70$

Figure 5.- Wing span loadings on the single-tail model; $\beta = 0^\circ$. Values at $2y/b$ of 0.175 and 1.0 were obtained by extrapolation.

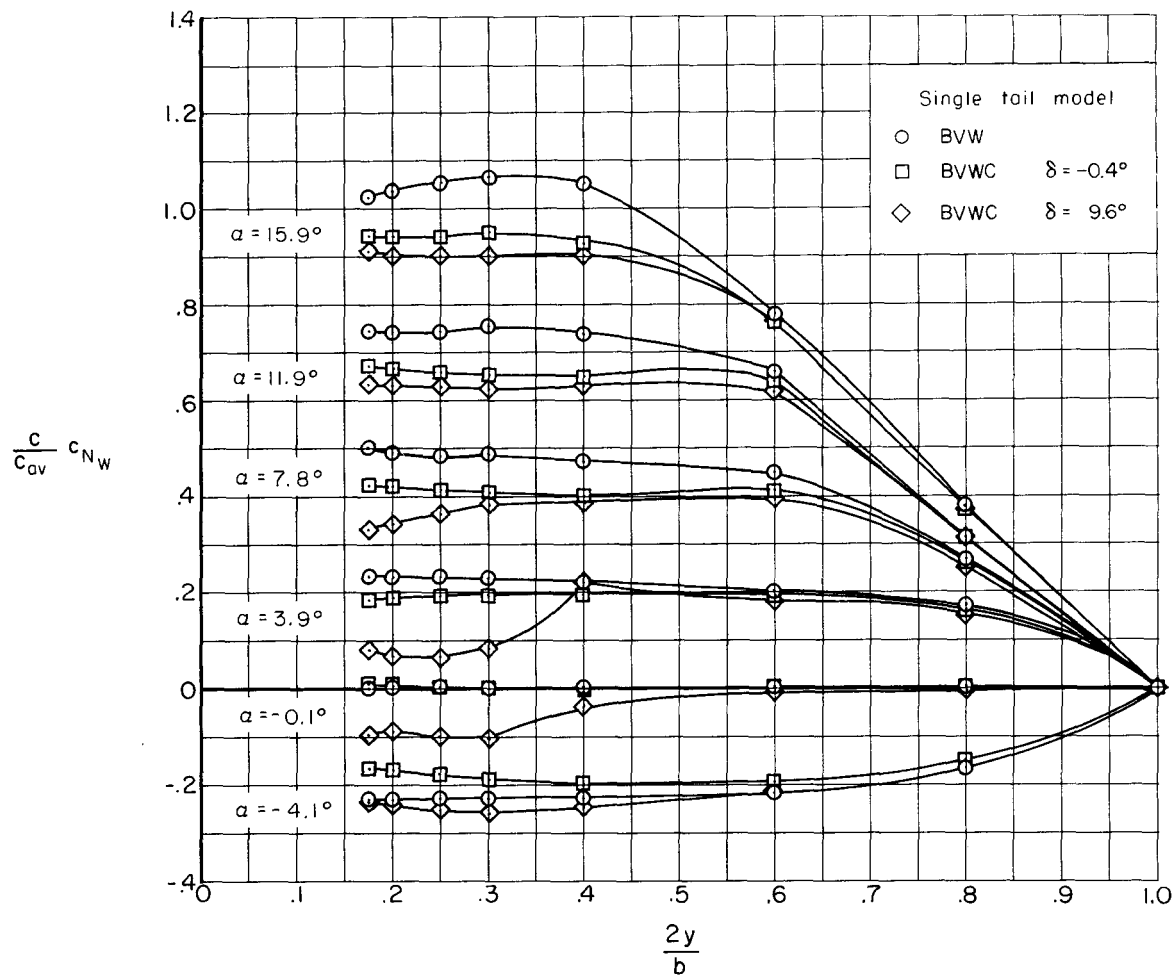
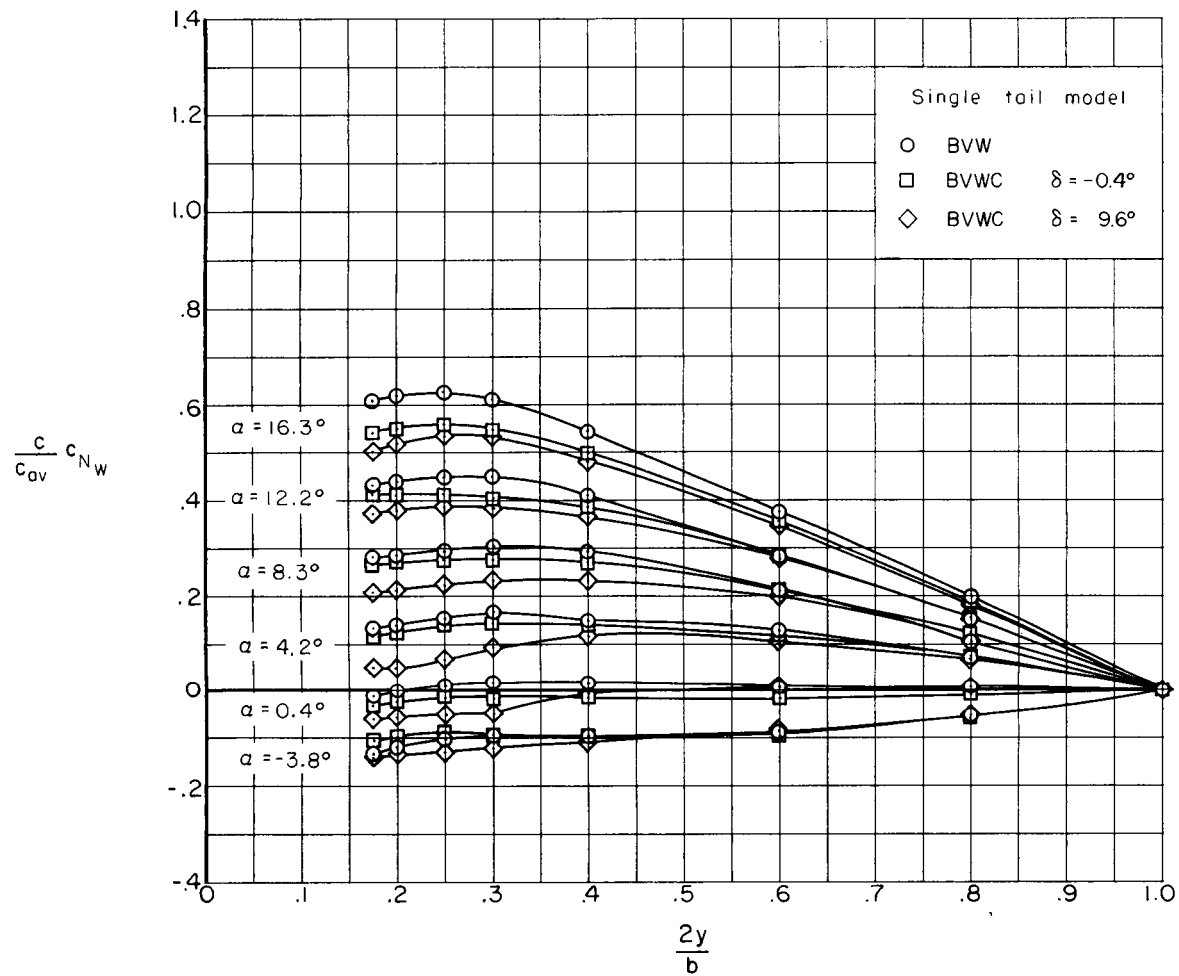
(b) $M = 1.30$

Figure 5.- Continued.



(c) $M = 2.22$

Figure 5.- Concluded.

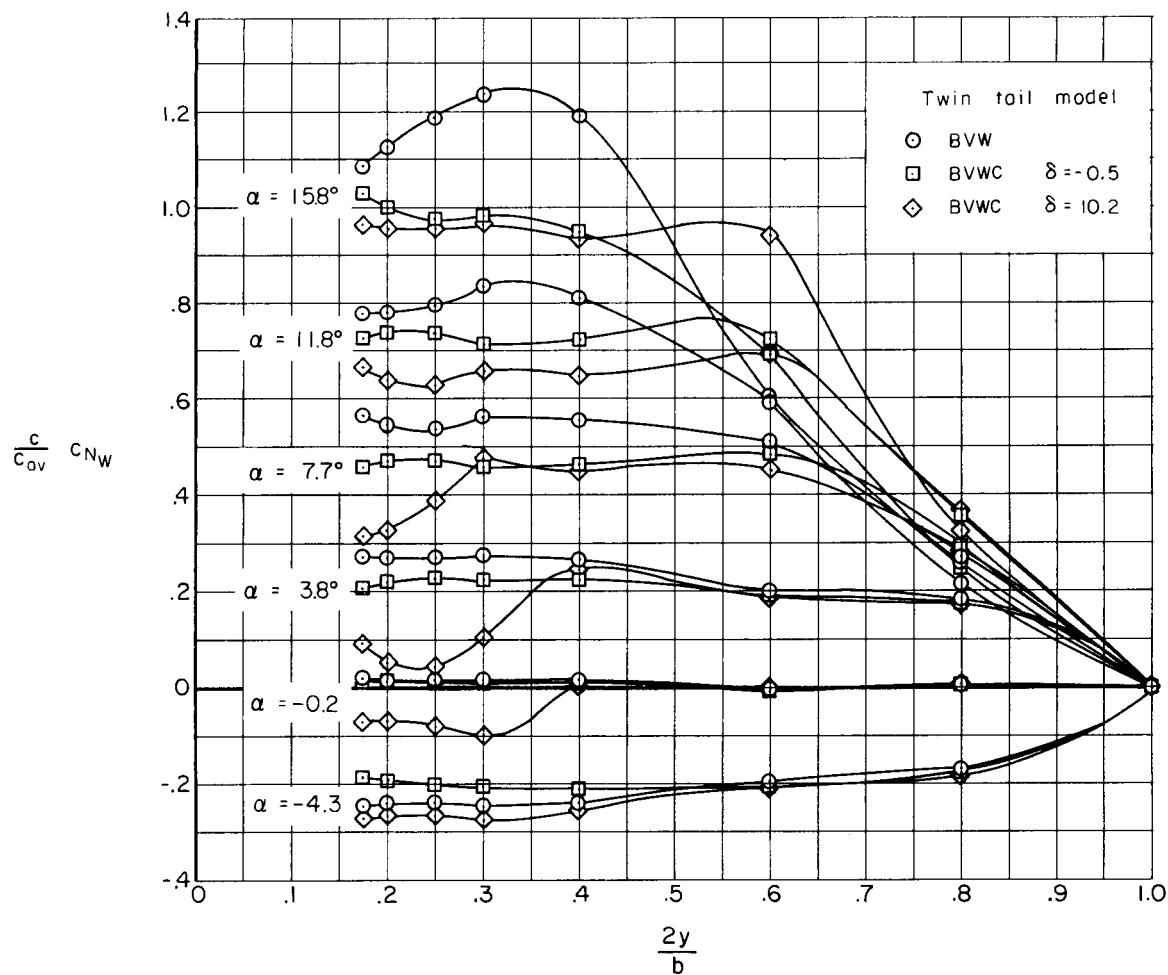
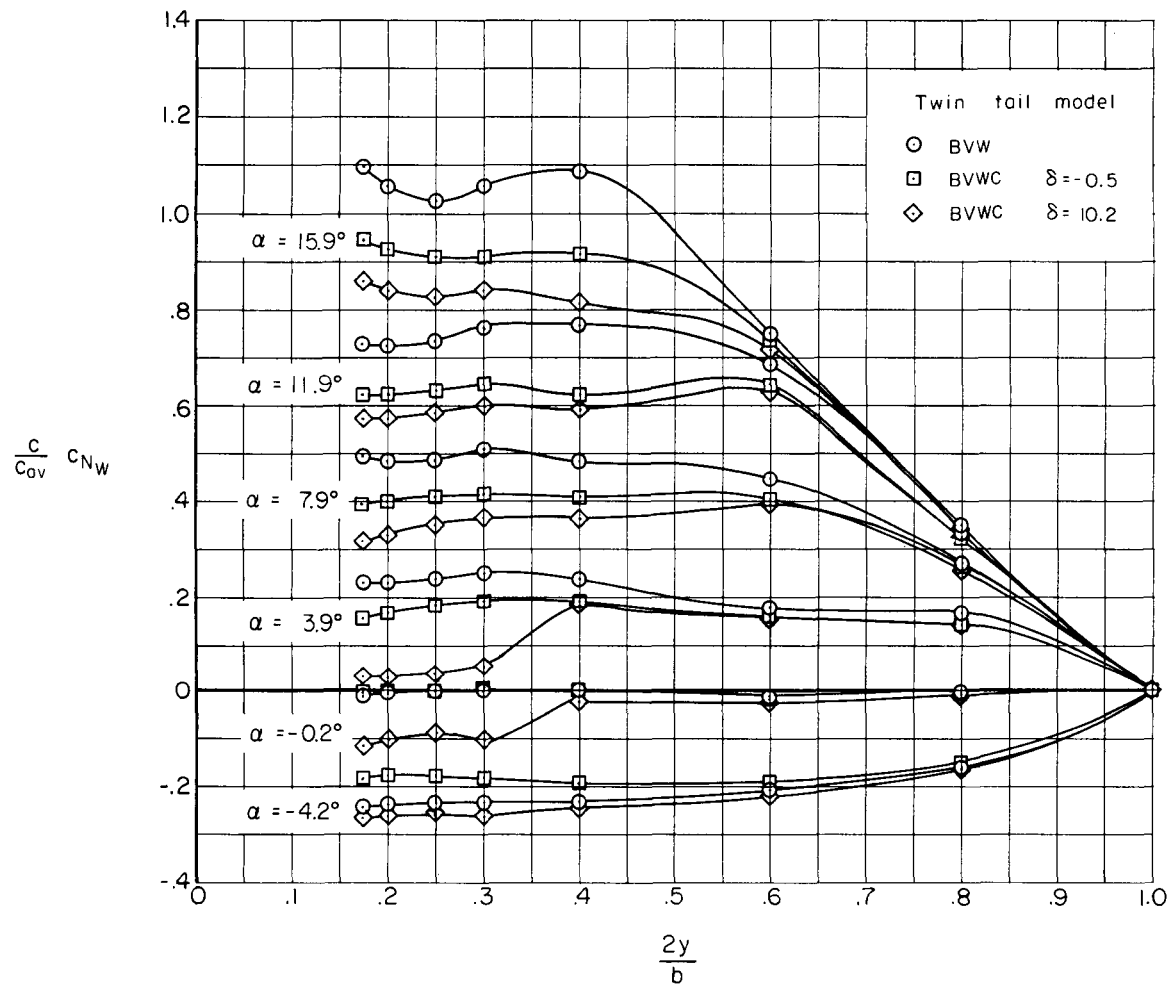
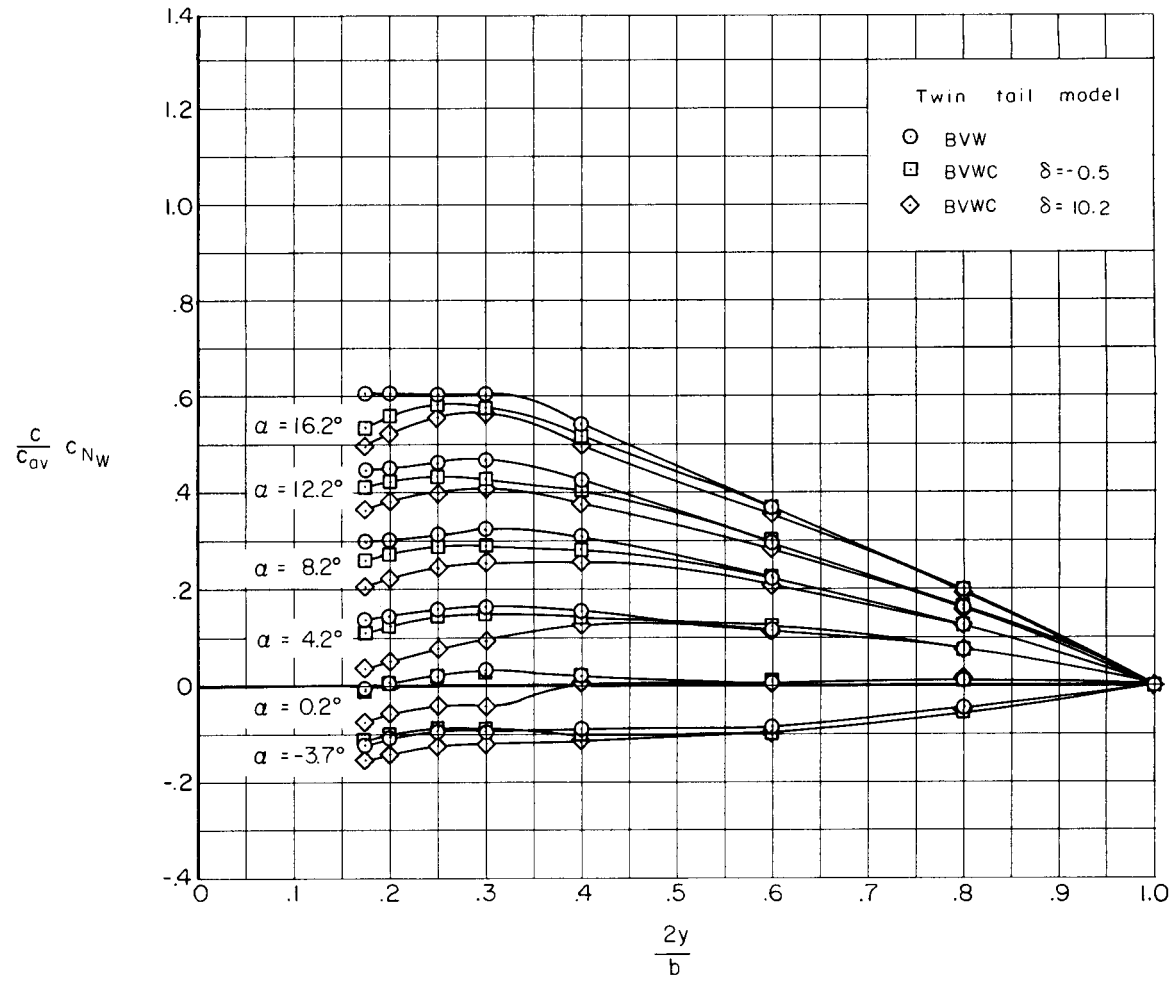
(a) $M = 0.70$

Figure 6.- Wing span loadings on the twin-tail model; $\beta = 0^\circ$. Values at $2y/b$ of 0.175 and 1.0 were obtained by extrapolation.



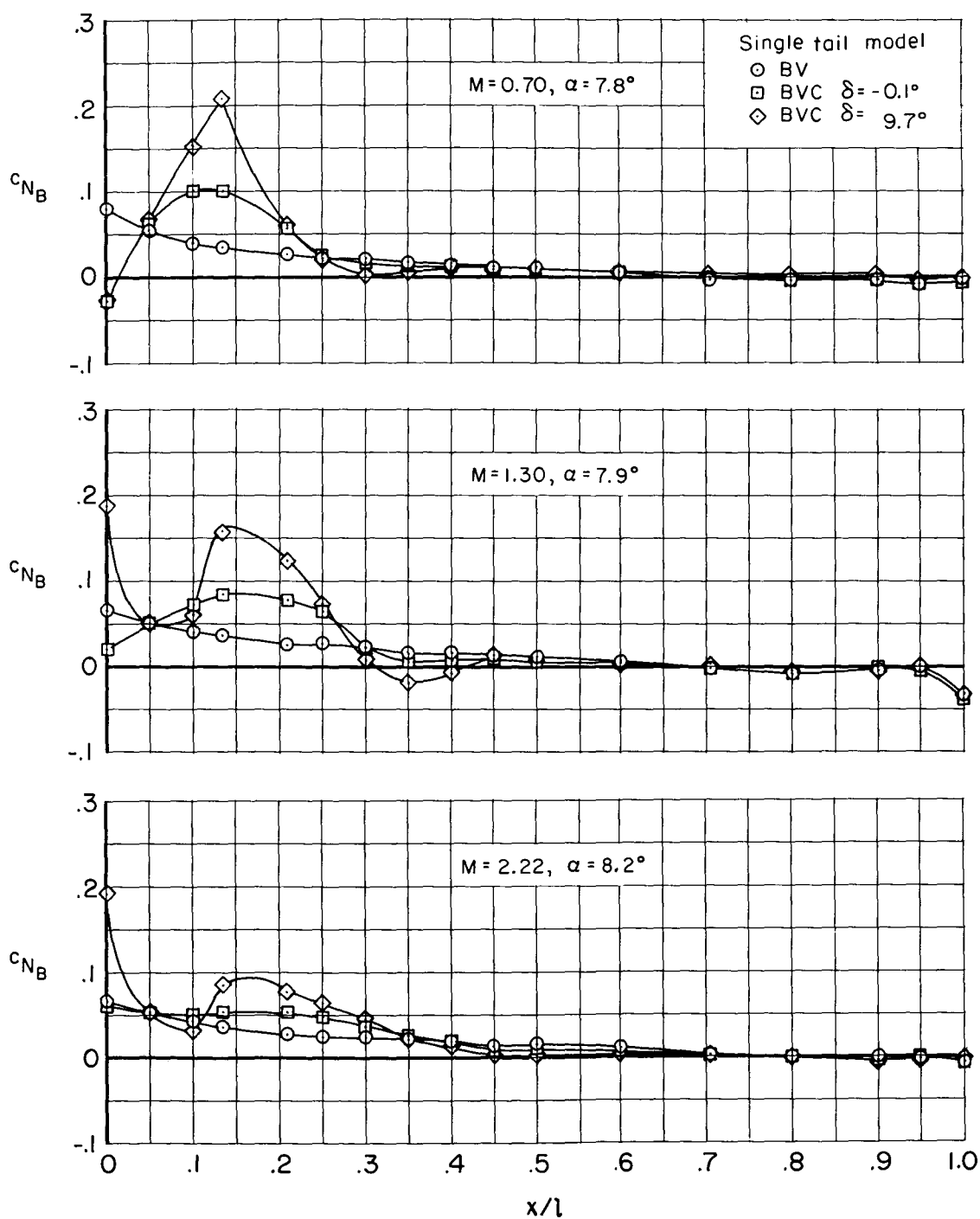
(b) $M = 1.30$

Figure 6.- Continued.



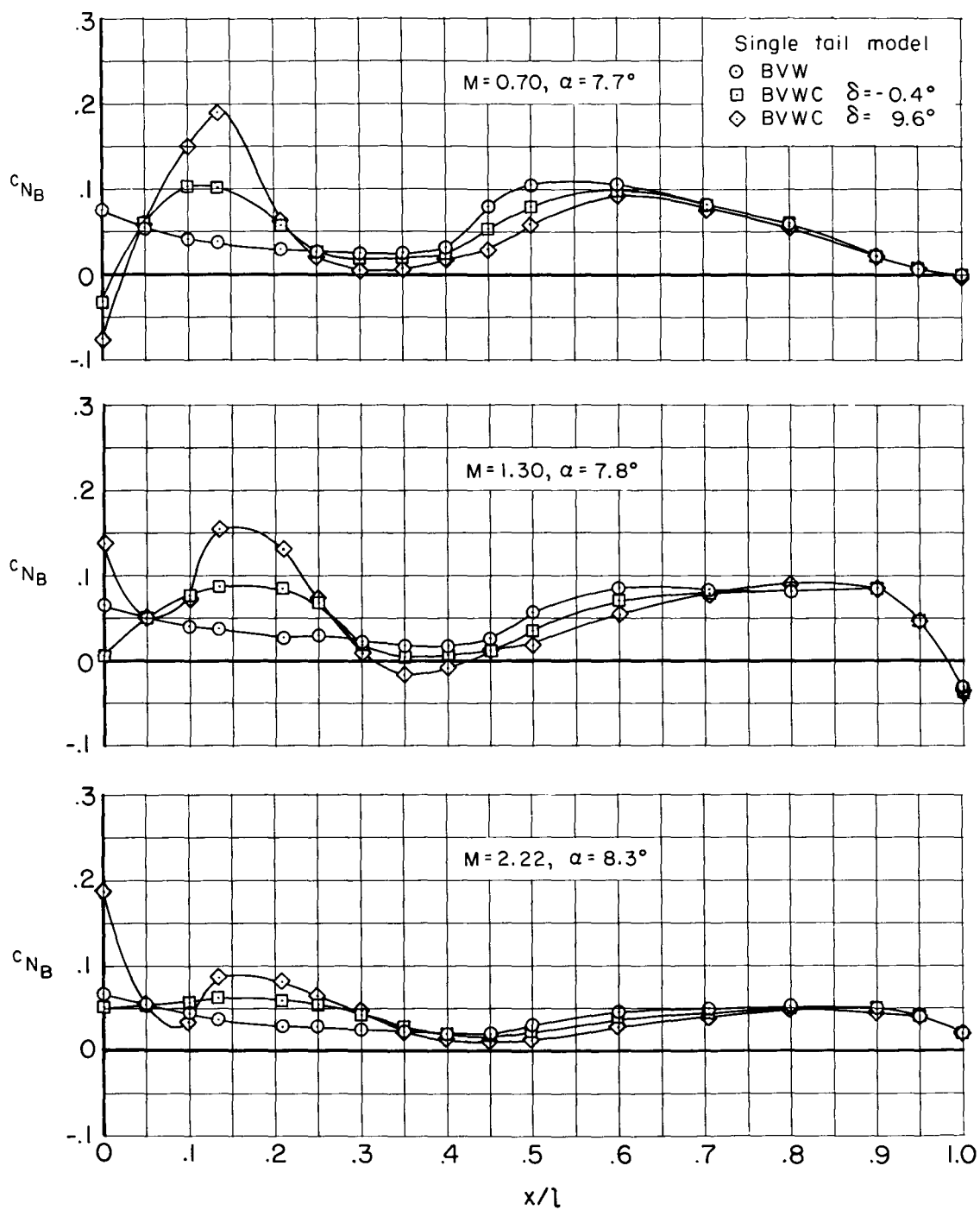
(c) $M = 2.22$

Figure 6.- Concluded.



(a) Wing off.

Figure 7.- Coefficients of force normal to the body surface in the vertical plane of symmetry as a function of body length for the single-tail model with and without the wing; $\beta = 0^\circ$. Values at $x/l = 0$ and 1.0 were obtained by extrapolation.



(b) Wing on.

Figure 7.- Concluded.

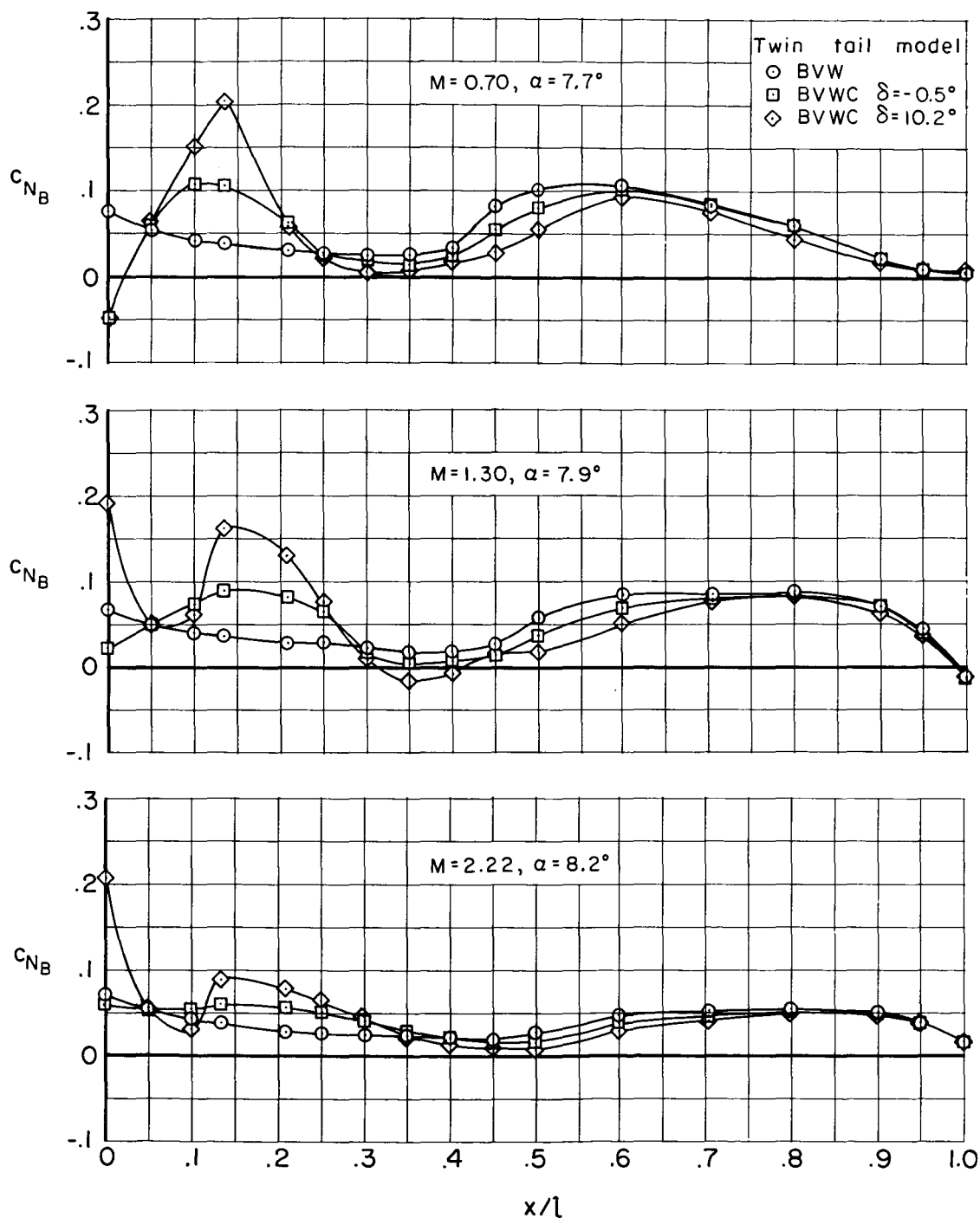


Figure 8.- Coefficients of force normal to the body surface in the vertical plane of symmetry as a function of body length for the twin-tail model with the wing on; $\beta = 0^\circ$. Values at $x/l = 0$ and 1.0 were obtained by extrapolation.

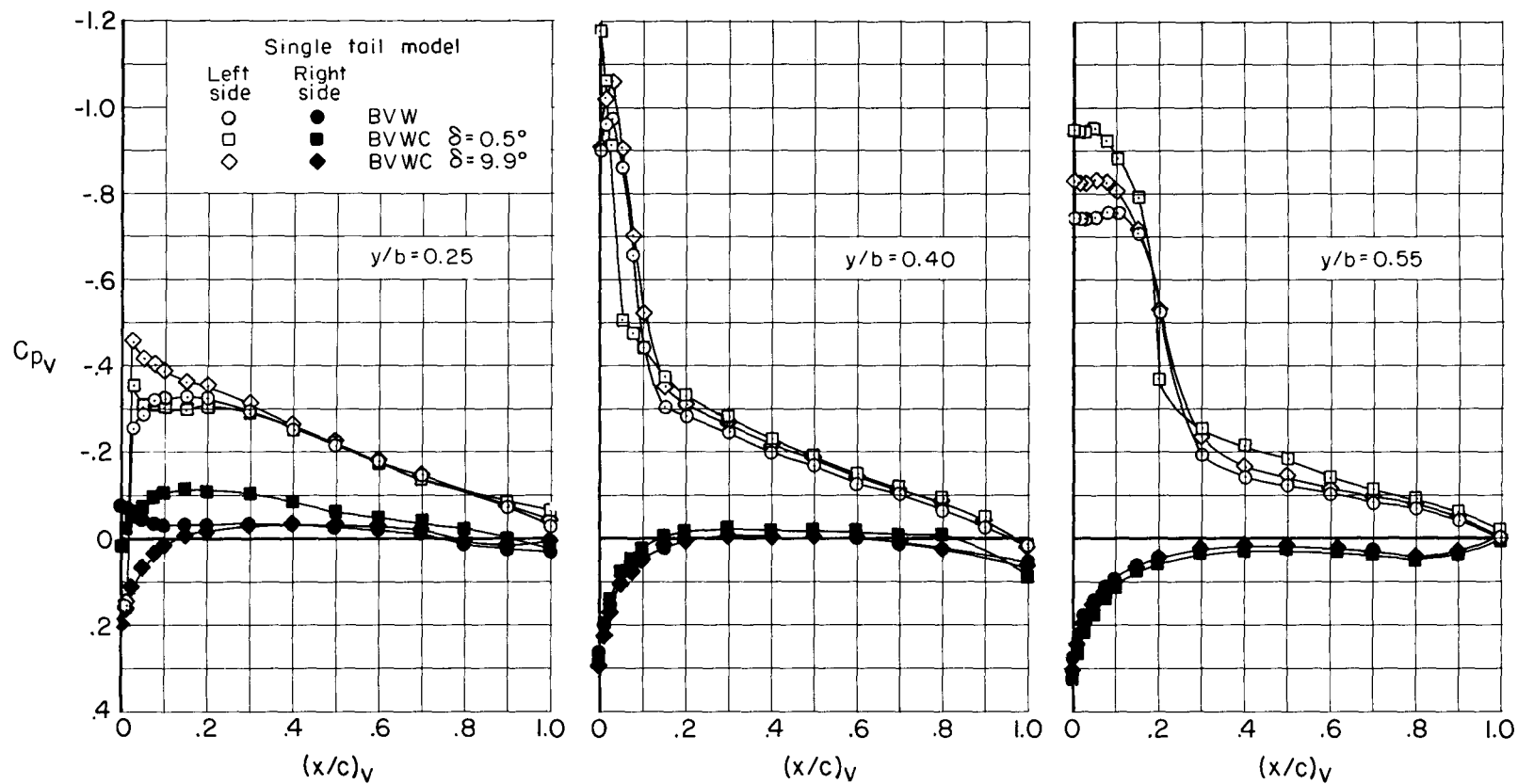
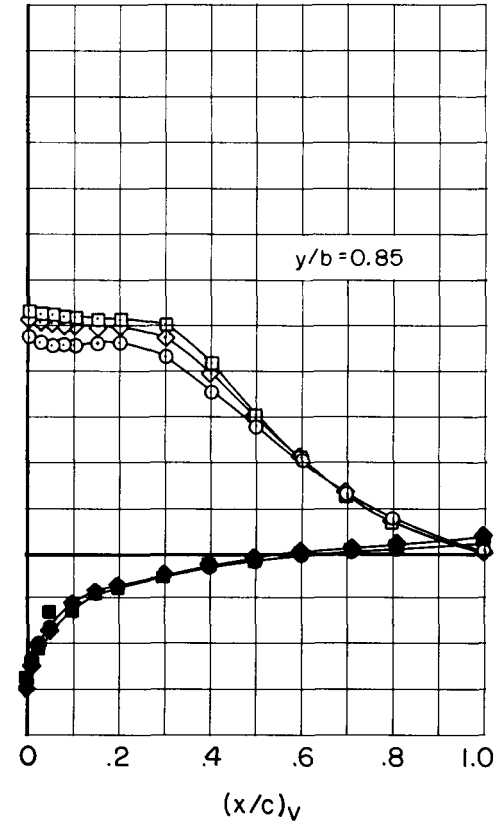
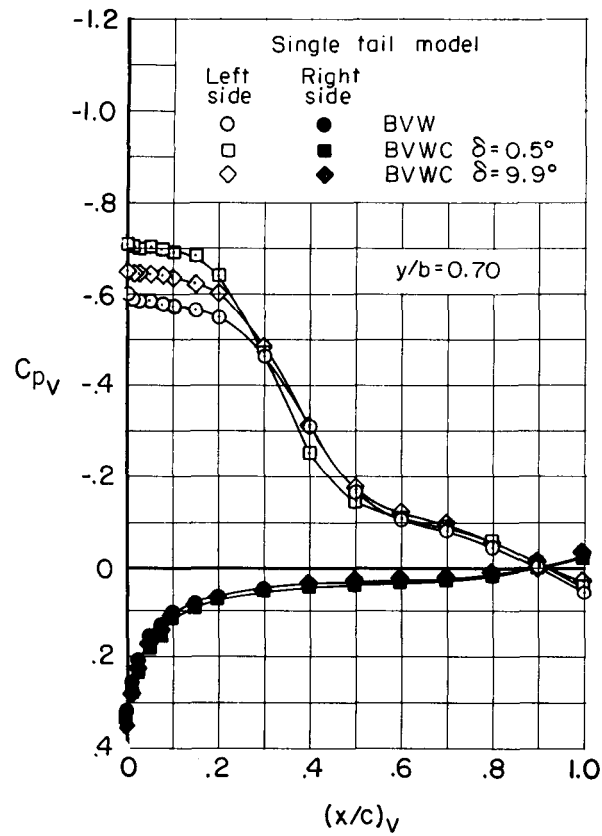
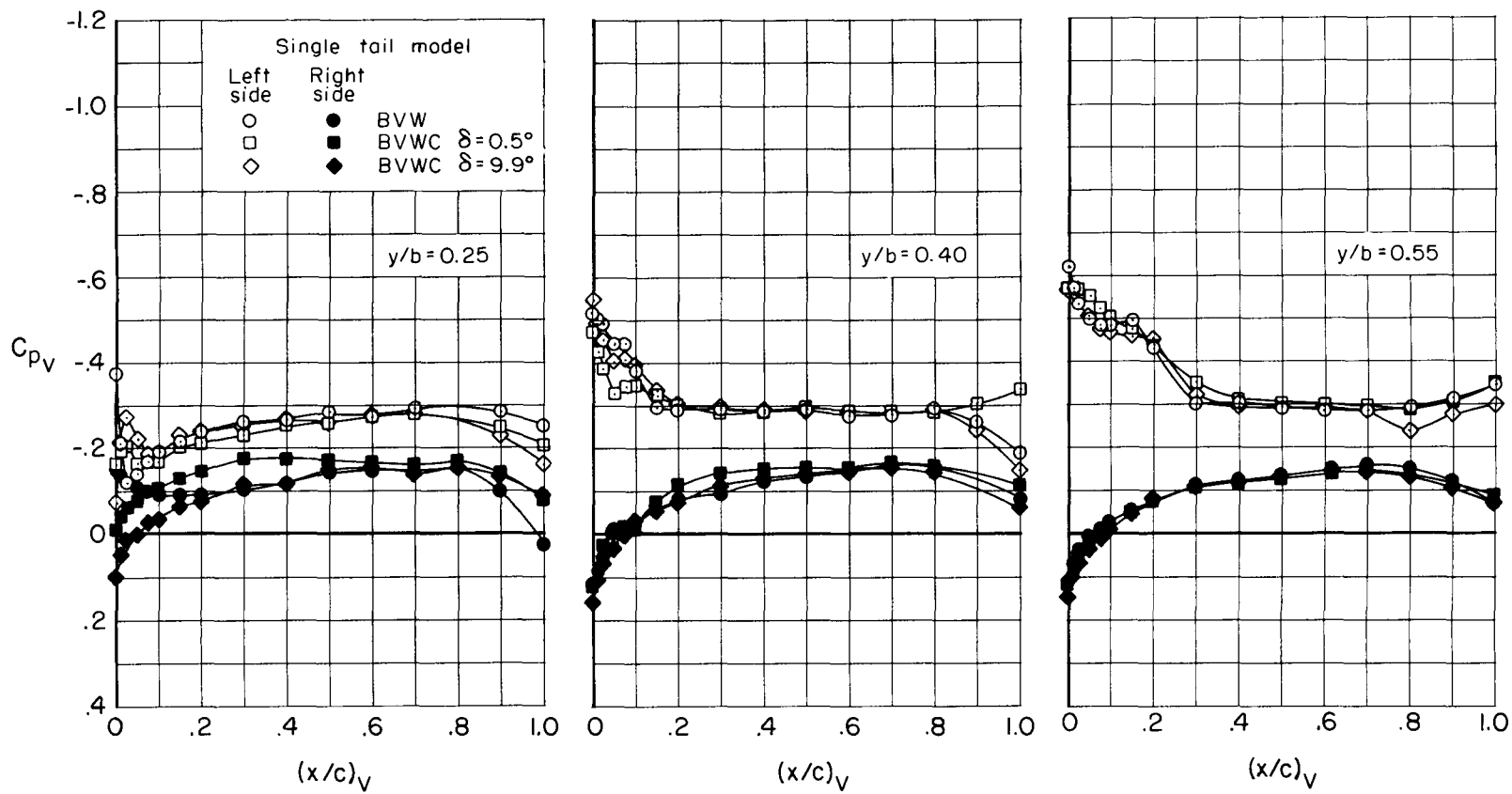
(a) $M = 0.70$, $\alpha = 7.9^\circ$

Figure 9.- Chordwise distributions of pressure on the single vertical tail; $\beta = 5.3^\circ$. Values at $(x/c)_V$ of 0 and 1.0 were obtained by extrapolation.



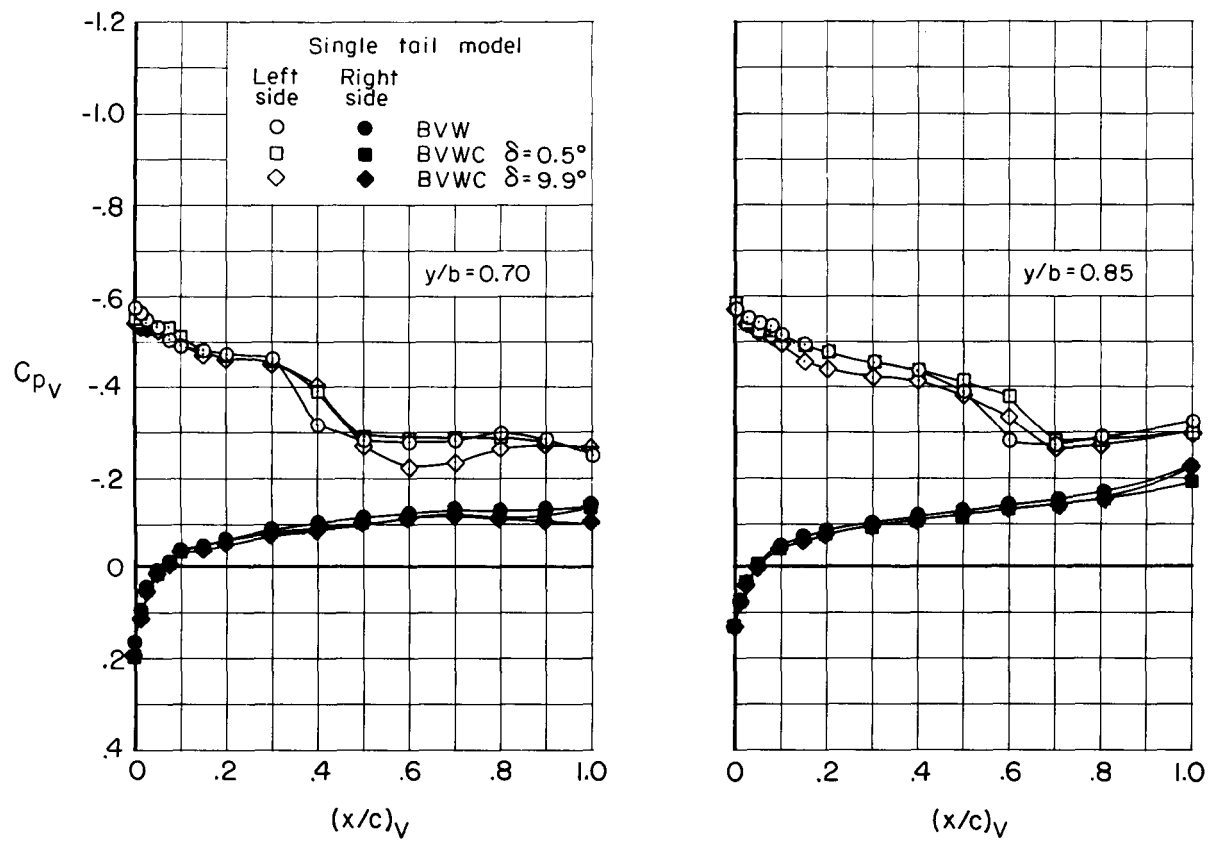
(a) $M = 0.70$, $\alpha = 7.9^\circ$ - Concluded.

Figure 9.- Continued.



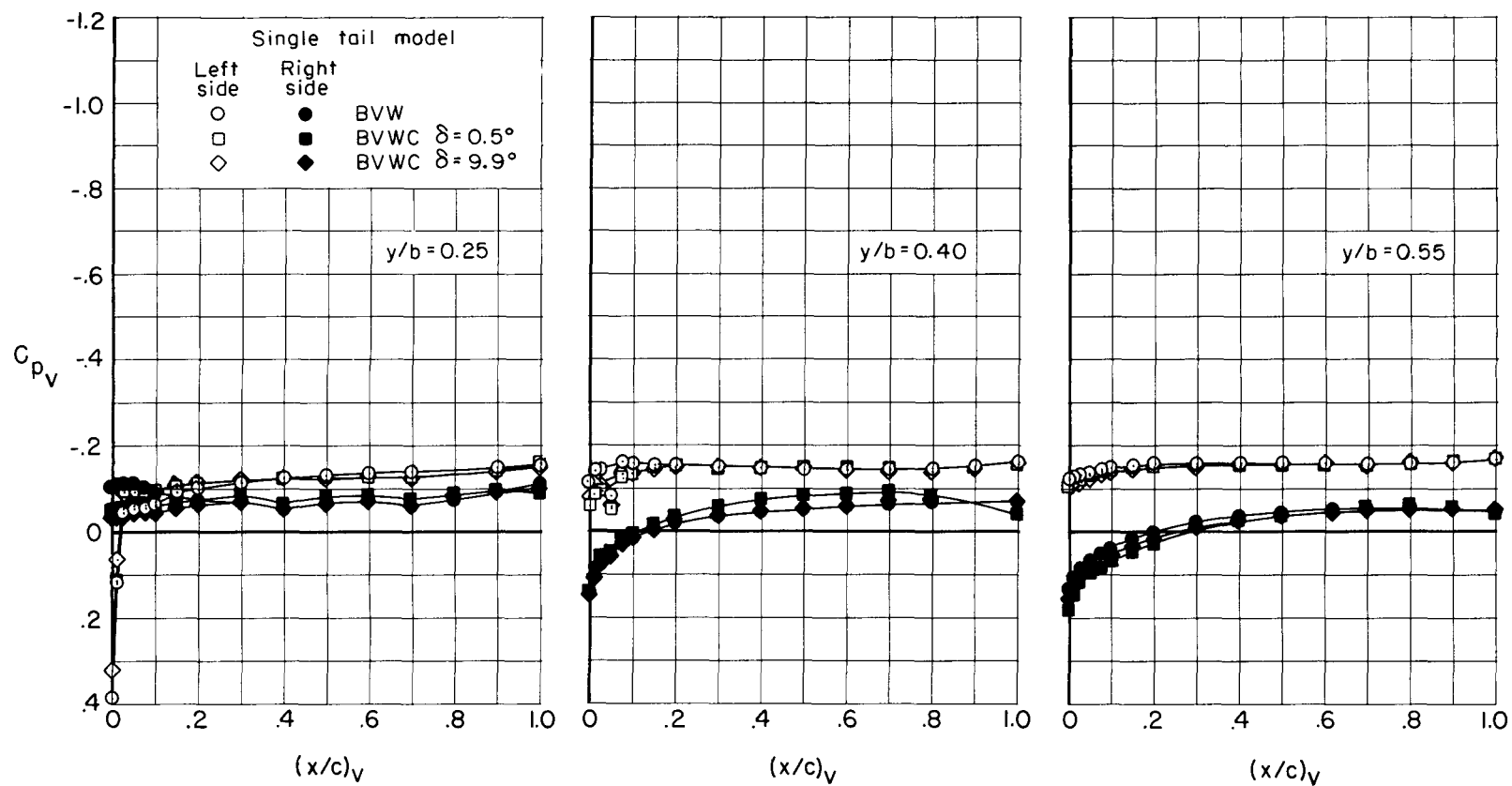
(b) $M = 1.30$, $\alpha = 8.1^\circ$

Figure 9.- Continued.



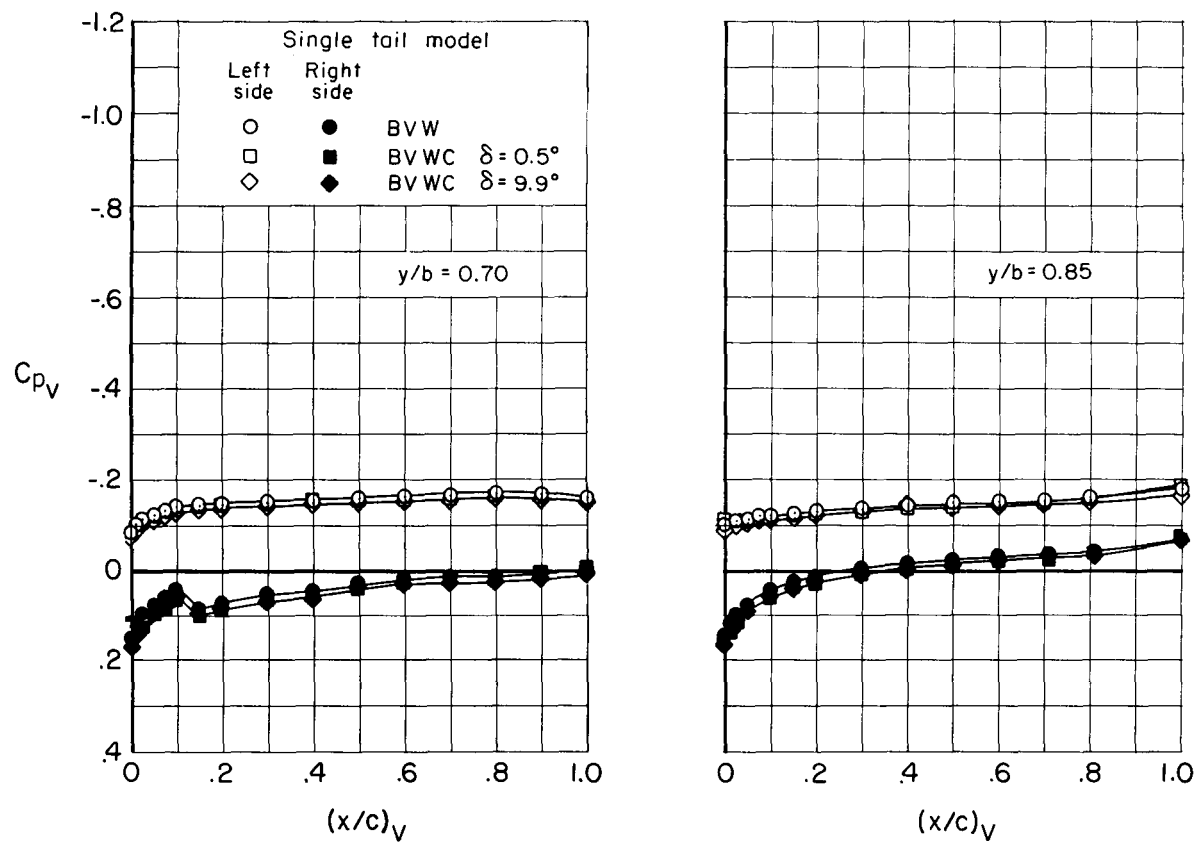
(b) $M = 1.30$, $\alpha = 8.1^\circ$ - Concluded.

Figure 9.- Continued.



(c) $M = 2.22$, $\alpha = 8.3^\circ$

Figure 9.- Continued.



(c) $M = 2.22$, $\alpha = 8.3^\circ$ - Concluded.

Figure 9.- Concluded.

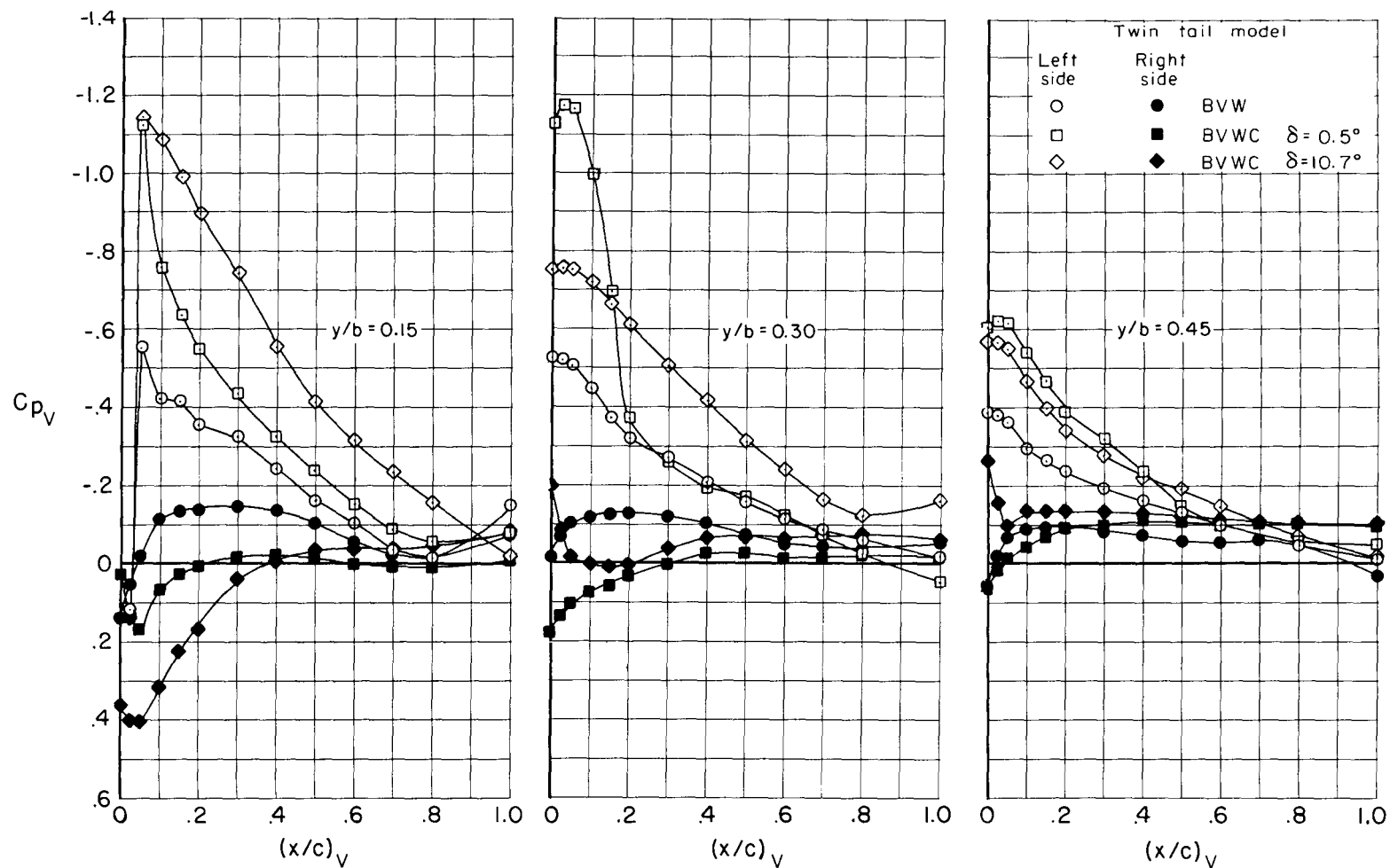
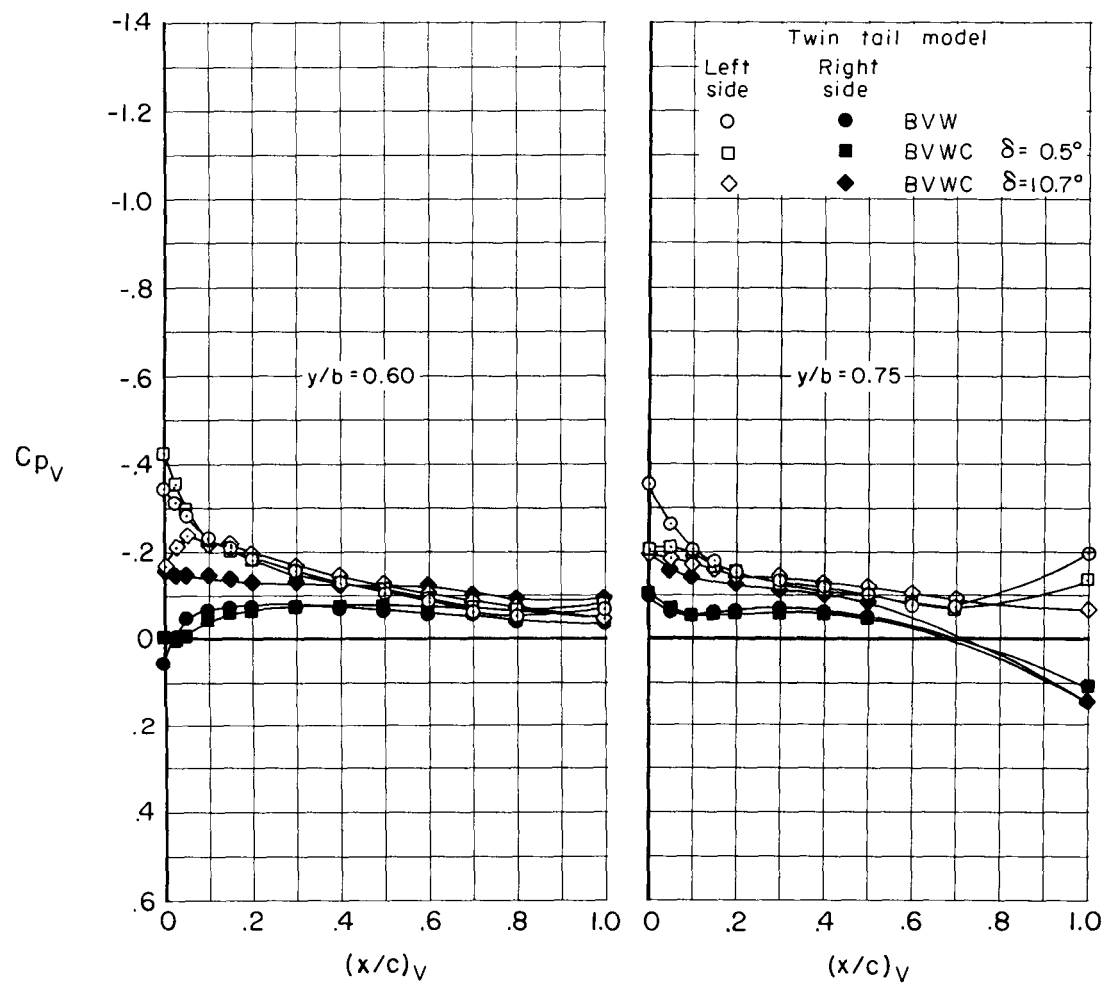
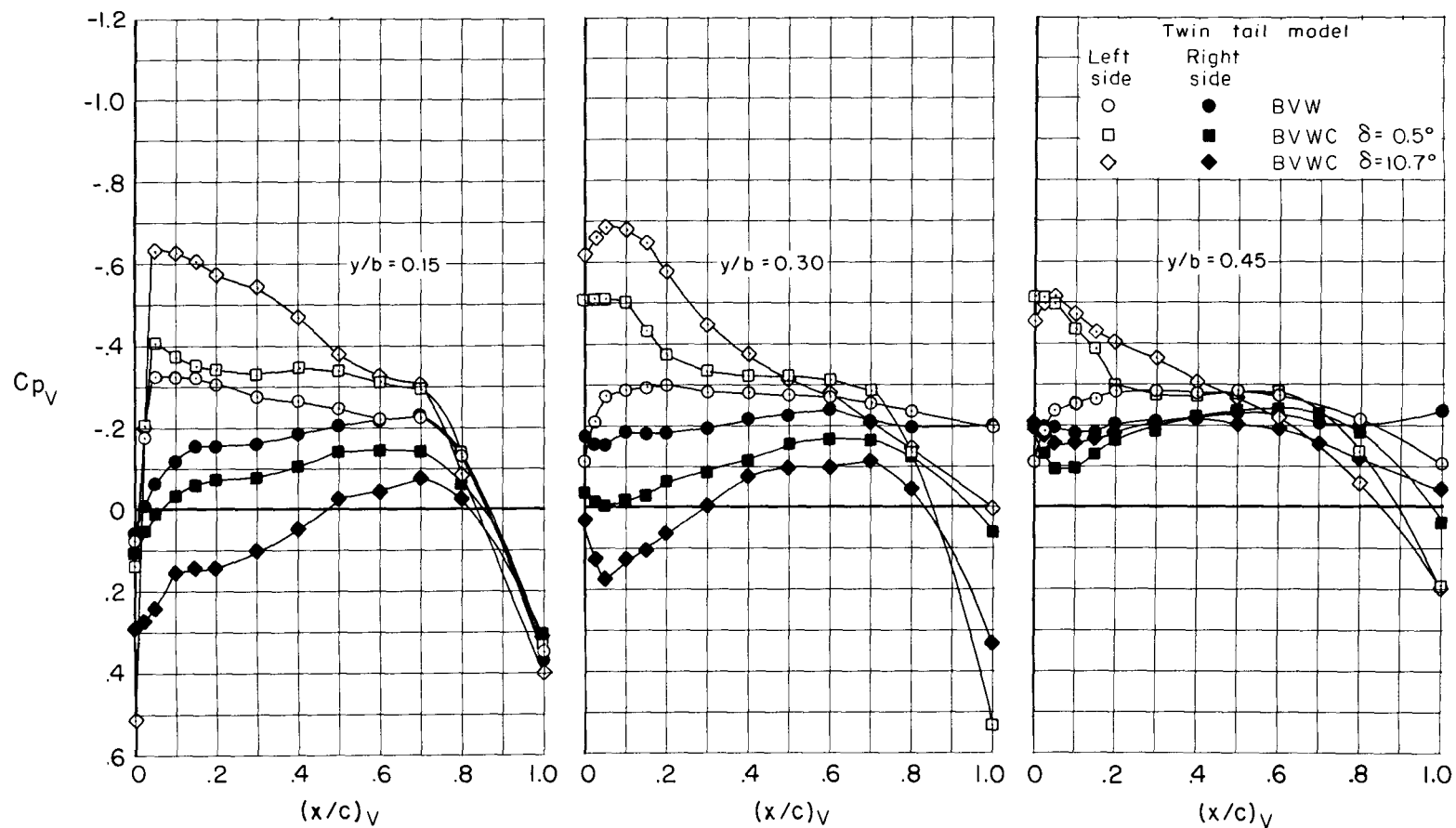
(a) $M = 0.70$, $\alpha = 8.0^\circ$

Figure 10.- Chordwise distributions of pressure on the left-side twin vertical tail; $\beta = 5.3^\circ$.
 Values at $(x/c)_v$ of 0 and 1.0 were obtained by extrapolation.



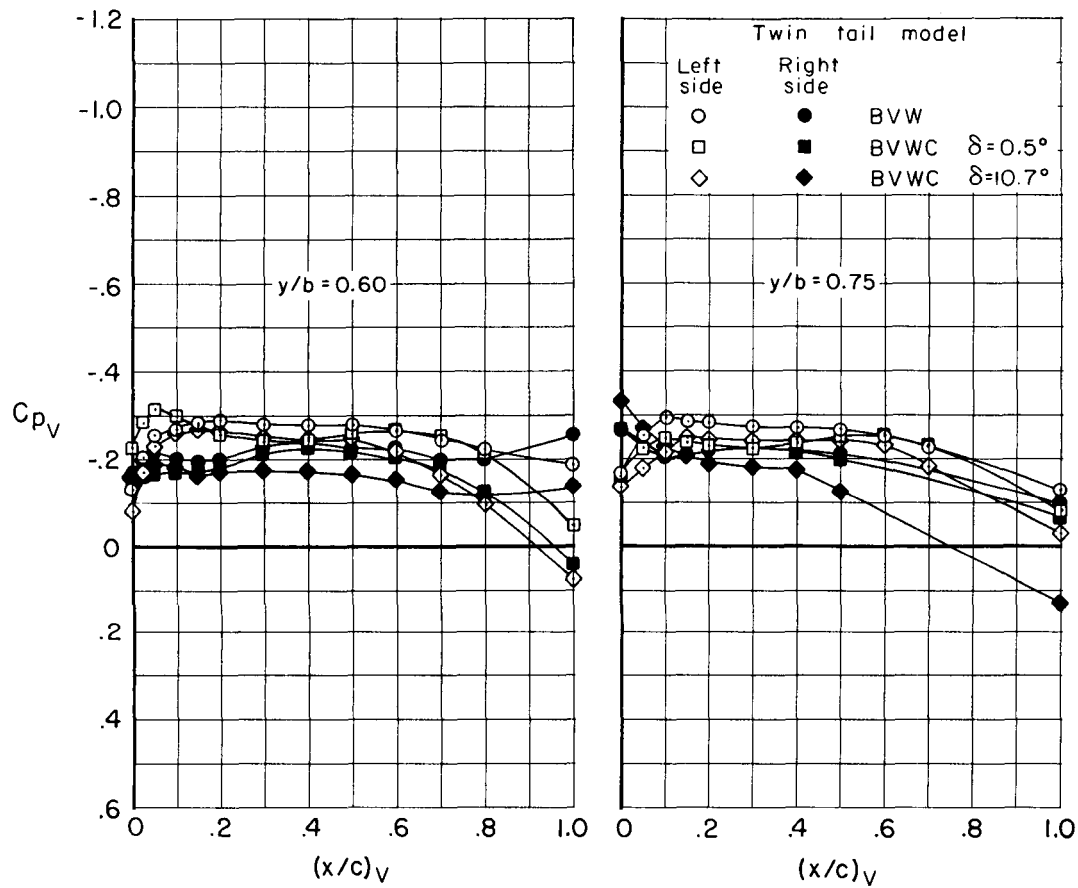
(a) $M = 0.70$, $\alpha = 8.0^\circ$ - Concluded.

Figure 10.- Continued.



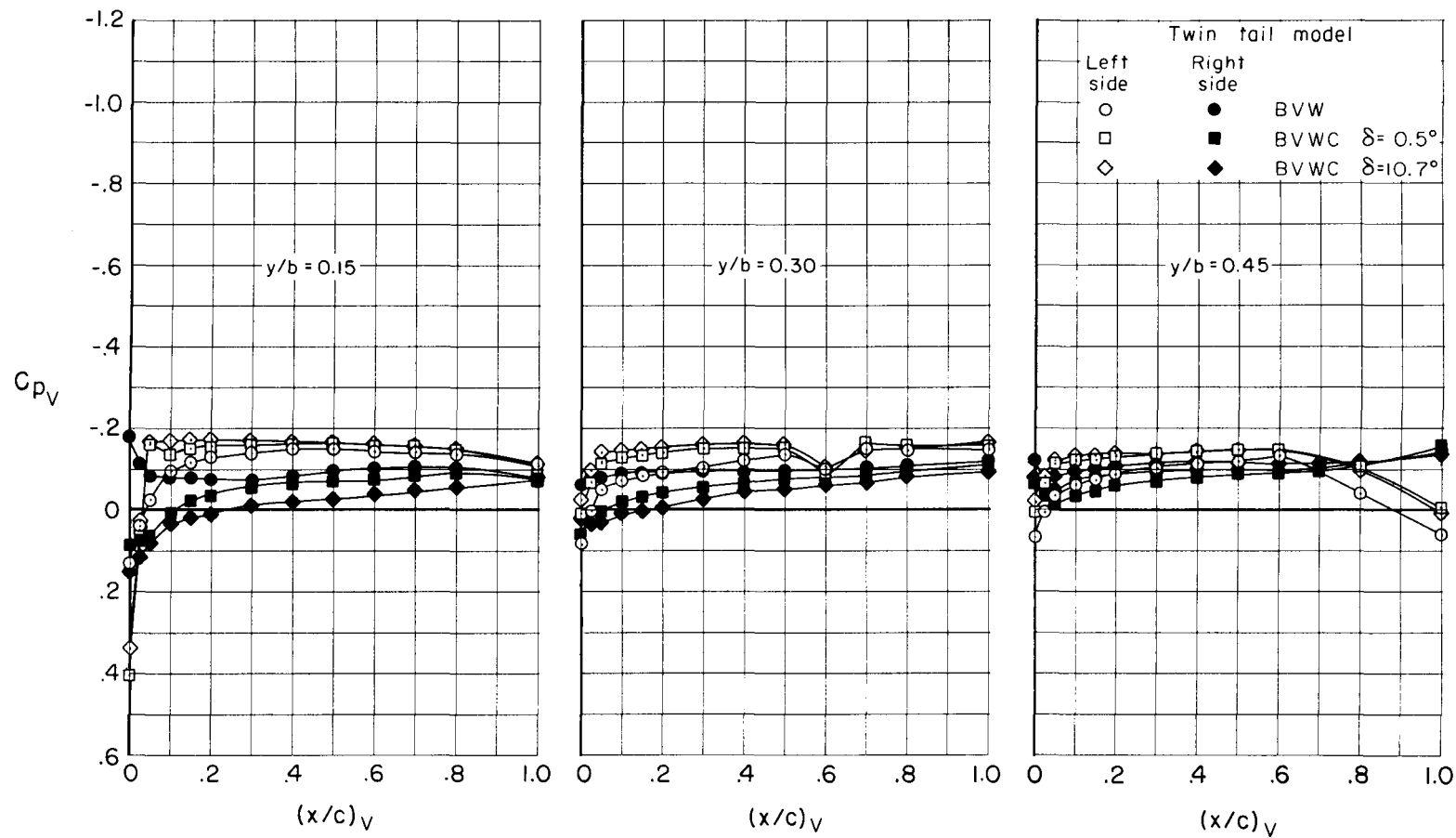
(b) $M = 1.30$, $\alpha = 8.1^\circ$

Figure 10.- Continued.



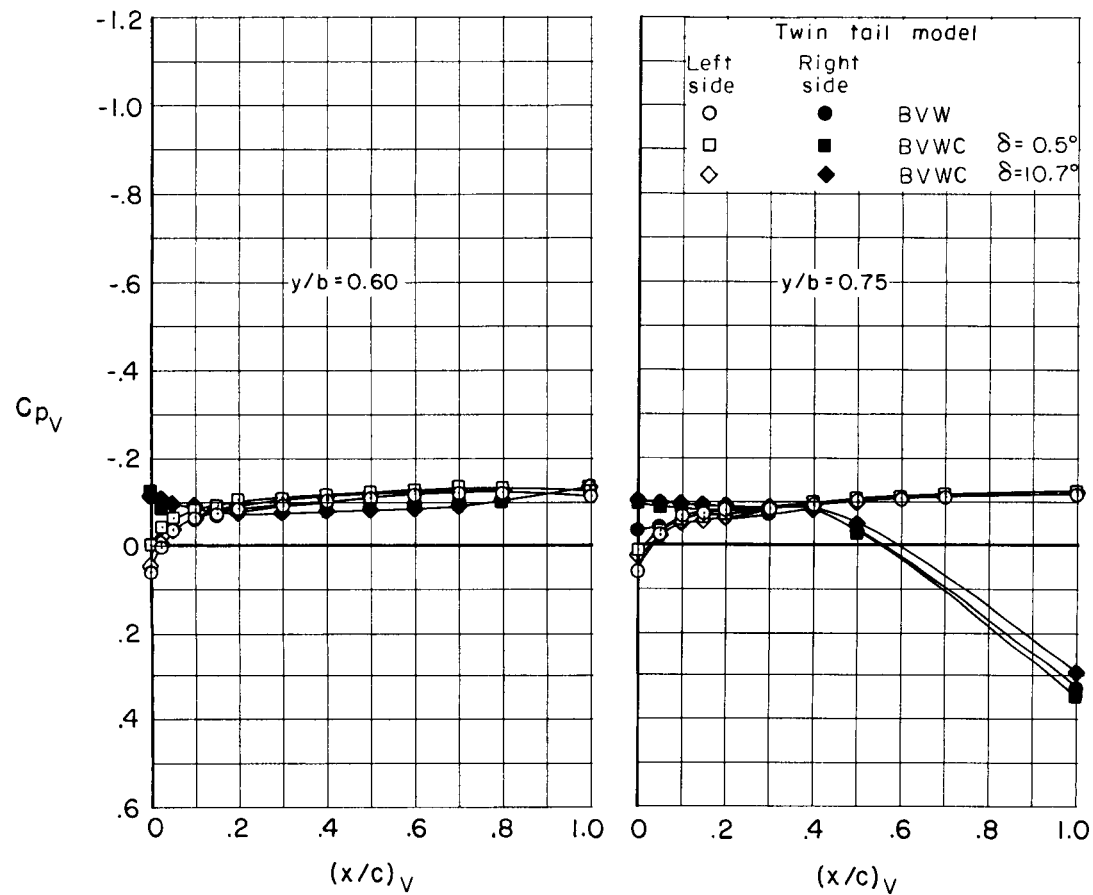
(b) $M = 1.30$, $\alpha = 8.1^\circ$ - Concluded.

Figure 10.- Continued.



(c) $M = 2.22$, $\alpha = 8.3^\circ$

Figure 10.- Continued.



(c) $M = 2.22$, $\alpha = 8.3^\circ$ - Concluded.

Figure 10.- Concluded.

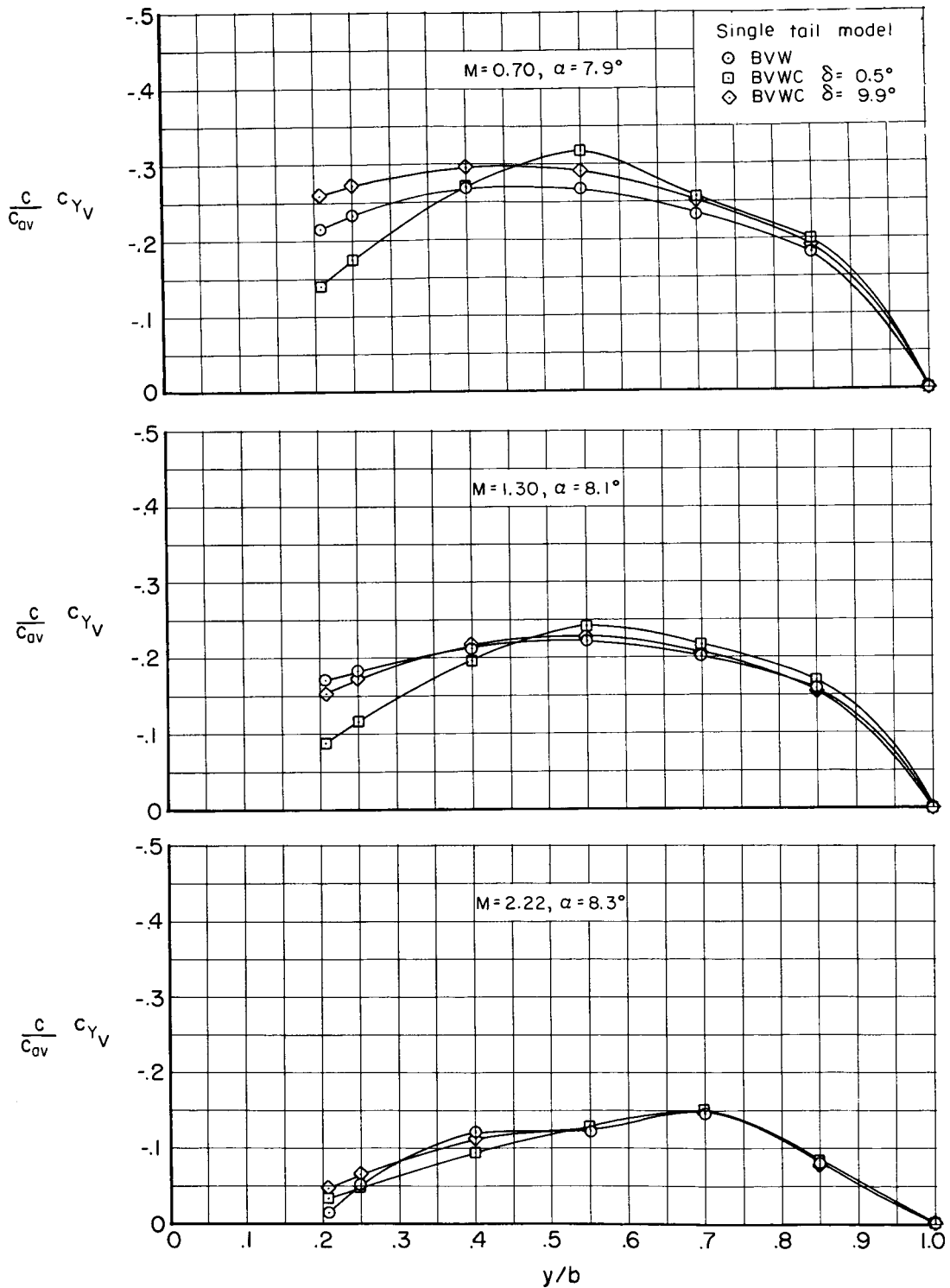


Figure 11.- Span loads on the single vertical tail; $\beta = 5.3^\circ$. Values at $y/b = 0.208$ and 1.0 were obtained by extrapolation.

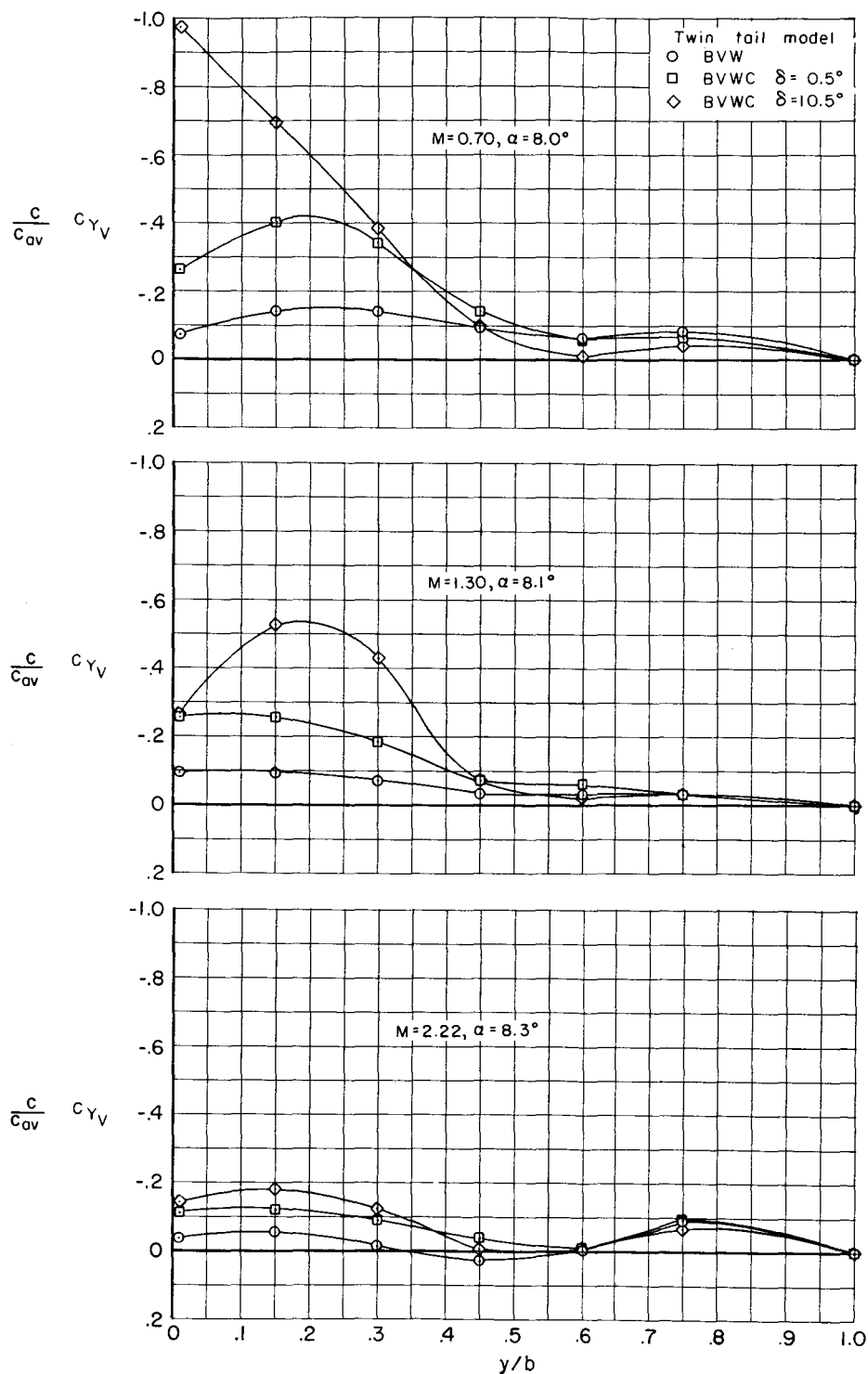


Figure 12.- Span loads on the left-side twin vertical tail; $\beta = 5.3^\circ$.
 Values at $y/b = 0.011$ and 1.0 were obtained by extrapolation.