TECHNICAL NOTES

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SPINNING CHARACTERISTICS OF WINGS

V - N.A.C.A. 0009, 23018, AND 6718 MONOPLANE WINGS

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SUMMARY

Three rectangular monoplane wings having rounded tips were tested on the N.A.C.A. spinning balance in the 5-foot vertical wind tunnel. The airfoil sections used were the N.A.C.A. 0009, 23018, and 6718.

The aerodynamic characteristics of the models and a prediction of the angles of sideslip for steady spins are given. There is included an estimate of the yawing moment that must be furnished by parts of the airplane to balance the inertia couples and wing yawing moments for spinning equilibrium. The predicted angles of sideslip and yawing moments required for spinning equilibrium for a Clark Y wing with the same plan form are included for comparison.

INTRODUCTION

In order to provide necessary data for predicting airplane spinning characteristics from the design features, the N.A.C.A. is conducting an extensive investigation to determine the aerodynamic characteristics of airplane models and models of airplane parts in spinning attitudes.

The investigation to determine the spinning characteristics of wings, in which the N.A.C.A. spinning balance was used, included variations in airfoil section, plan form, and tip shape of monoplane wings and variations in stagger of biplane cellules. The first and third series of tests reported were made of Clark Y monoplane wings with rectangular plan forms, with square and rounded tips, and with a 5:2 tapered plan form having rounded tips (references 1 and 2). The second and fourth series were made of a rectangular Clark Y biplane cellule with -0.25, 0, and 0.25 stagger and are reported in references 3 and 4.

This report gives the aerodynamic characteristics in spinning attitudes of N.A.C.A. 23018, 6718, and 0009 rectangular monoplane wings with rounded tips. Data for the Clark Y wing previously tested are included for comparison. The discussion of the data is based on the method of analysis given in reference 1.

APPARATUS AND MODELS

The tests were made on the spinning balance in the N.A.C.A. 5-foot vertical wind tunnel. The tunnel is described in reference 5 and the 6-component balance in reference 6.

The wings were made of laminated mahogany to the N.A.C.A. 0009, 23018, and 6718 airfoil sections. They are rectangular in plan form with rounded tips, and have an aspect ratio of 6. The tip plan form is composed of quadrants of similar ellipses. The section profile is maintained to the end of the wings and, in elevation, the maximum upper-surface section points are in one plane. This tip shape, as shown in figure 1, has been designated the "Army" tip. Figure 2 shows the N.A.C.A. 6718 model mounted on the balance.

TESTS

In order to cover the probable spinning range, tests were made at angles of attack of 30°, 40°, 50°, 60°, and 70°. At each angle of attack tests were made with sideslip angles of 10°, 5°, 0°, -5°, and -10° for the N.A.C.A. 0009 and 6718 airfoils and of 5°, 0°, -5°, and -10° for the N.A.C.A. 23018 airfoil. At each angle of attack at each angle of sideslip, tests were made with values of $\Omega b/2V$ of 0.25, 0.50, 0.75, and 1.00. The angles of attack were referred to the chord of the section in the plane of symmetry. The angles of sideslip were measured at the quarter-chord point in the plane of symmetry of the wing, which was the center of rotation for all tests. Because of variations between individual balance readings, at least one repeat test was made for each condition and an average of the individual measurements was used to compute the coefficients.

The tunnel air speed was 70 feet per second for tests with $\Omega b/2V$ = 0.25 and 0.50, and 60 and 45 feet per second for $\Omega b/2V$ = 0.75 and 1.00, respectively. The Reynolds Numbers of the tests were about 210,000 for the highest air speed and 140,000 for the lowest air speed. Previous tests showed no appreciable change in scale effect for this range.

RESULTS AND DISCUSSION

The data were converted to coefficient form by the following relations:

$$C_X = \frac{X}{qS} \qquad C_Y = \frac{Y}{qS} \qquad C_Z = \frac{Z}{qS}$$

$$C_l = \frac{L}{qbS} \qquad C_m = \frac{M}{qbS} \qquad C_n = \frac{N}{qbS}$$

All coefficients are standard N.A.C.A. coefficients except C_m, which is based on the span instead of the chord of the wing, and it may be converted to the standard coefficient by multiplying by 6. All coefficients are given conventional signs for right spins (references 1 and 6). The coefficients and moments are given about the quarter-chord point of the lower surface of the wing.

The coefficients of longitudinal force in the earth system of axes $C_{X''}$ and of all six components of the forces and moments in the body system of axes are given in tables I, II, and III. Sample curves of $C_{X''}$, C_l, C_m, and C_n are given in figures 3 to 6.

The data are believed to be correct to within the following limits:

$C_{X''}$, ±0.02	C_X, ±0.02	C_Y, ±0.01
C_Z, ±0.02	C_m, ±0.002	C_l, ±0.001
C_n, ±0.001		

No corrections have been made for the effects of jet boundary, scale, or interference of the balance.

Variation of the coefficients with airfoil section.- The longitudinal-force coefficients $C_{X''}$ generally decrease in the order N.A.C.A. 6718, Clark Y, N.A.C.A. 0009, and N.A.C.A. 23018 (fig. 3). The values of the rolling-moment coefficients C_l generally are algebraically less for the wings in the following order: N.A.C.A. 6718, Clark Y, N.A.C.A. 23018, and N.A.C.A. 0009 (fig. 4). The variations in C_l with angle of attack, angle of sideslip, and $\Omega b/2V$ are about the same for all the wings. The absolute values of the pitching-moment coefficients C_m generally decrease in the order N.A.C.A. 6718, Clark Y, N.A.C.A. 0009, and N.A.C.A. 23018 (fig. 5). The variations in C_m with angle of attack, angle of sideslip, and $\Omega b/2V$ are about the same for all the wings. The values of the yawing-moment coefficients C_n generally decrease algebraically in the order Clark Y, N.A.C.A. 6718, N.A.C.A. 0009, and N.A.C.A. 23018 except at the low angles of attack and at large values of $\Omega b/2V$, in which cases the values for the N.A.C.A. 6718 wing are the lowest (fig. 6). The variations of C_n for each wing are small with changes in angle of sideslip and are somewhat larger with changes in $\Omega b/2V$.

For the coefficients of all wings to be exactly comparable, the angles of attack for each wing should have been measured from the angle of zero lift and not from the chord line. The following coefficients may be corrected to the absolute angle of attack by the relations

$$C_{X_o''} = C_{X''} \quad \text{at} \quad \alpha - \Delta\alpha$$

$$C_{l_o} = C_l \quad \text{at} \quad \alpha - \Delta\alpha$$

$$C_{m_o} = C_m \quad \text{at} \quad \alpha - \Delta\alpha$$

$$C_{n_o} = C_n - C_l \sin \Delta\alpha$$

where $C_{X''}$, C_l, C_m, and C_n are the values given in the tables and $\Delta\alpha$ is the angle of attack for zero lift measured from the chord line. The values of C_{l_o} and C_{n_o} obtained in this way will be approximate, but the actual errors involved will be negligible. This same method may be used to transfer the data to airplane axes, since the wing is usually set to give some lift when the thrust line of the airplane has zero angle of attack. For this case $\Delta\alpha$ will be the angle between the chord line of the wing and the thrust line when the wing is set to give the required lift.

ANALYSIS

The data were analyzed to show the effects of some of the important parameters on the spinning characteristics of an airplane using wings similar to those tested. The method of analysis with the assumptions used and the errors involved is given in references 1 to 6.

Parameters.- The characteristics of the particular airplane determine the values of wing loading, aspect ratio, radii of gyration, and pitching moments. The characteristics of airplanes have changed appreciably since the investigation reported in reference 1; therefore, in the present analysis the two sets of parameters given in the table (p. 6) were used. The previous values are the same as those used in references 1 to 4 and are included to allow the results in this report to be compared with the earlier investigations. The present values are used to cover the range for present-day airplanes.

The values of μ cover the range for airplanes that are normally spun. The early values of $\frac{b^2}{(k_Z^2 - k_X^2)}$ and $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ cover the range for the 11 airplanes given in reference 7.

In each set, all the parameters were varied, one at a time, while the others were kept at the mean values, except C_L, which was set equal to $C_{X''}$ for all cases.

Discussion of results of analysis.- The angles of sideslip at which the pitching and the rolling moments balance in a steady spin and the yawing moment that must be furnished by the other parts of the airplane to balance the inertia couples and the wing yawing moments are plotted against the first set of parameters in figures 7 to 14 and against the second set of parameters in figures 15 to 22. Negative values of C_n required show the amount of yawing moment opposing the spin that must be supplied to balance the resultant aiding moment given by the wings and the inertia couples. It is obvious that, in order to insure against a dangerous spin, an additional opposing moment must be supplied as a margin of safety.

Parameters	Previous values	Present values
Slope of assumed pitching-moment curve $\frac{-C_m^2}{\alpha - 20^\circ}$	0.0010, 0.0015, $(0.0020)^a$, 0.0025, and 0.0030	0.0010, 0.0020, $(0.0030)^a$, 0.0040, and 0.0050
Pitching-moment inertia parameter $\frac{b^2}{k_Z^2 - k_X^2}$ $\left(\frac{mb^2}{C-A}\right)$	60, $(80)^a$, 100, and 120	40, $(60)^a$, 80, and 100
Relative density of airplane to air, μ $\left(\frac{m}{\rho S b}\right)$	2.5, $(5.0)^a$, 7.5, and 10.0	2.5, 5.0, $(7.5)^a$, 10.0, and 12.5
Rolling- and yawing-moment inertia parameter $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ $\left(\frac{C-B}{C-A}\right)$	0.5, $(1.0)^a$, 1.5, and 2.0	0.15, 0.30, $(0.50)^a$, 1.00, and 1.50
Lift coefficient, C_L	$C_L = C_{X''}$	$C_L = C_{X''}$

[a]Values in parentheses are mean values of the parameters.

$A = mk_X^2$, the moment of inertia about the X axis.

$B = mk_Y^2$, the moment of inertia about the Y axis.

$C = mk_Z^2$, the moment of inertia about the Z axis.

Increasing the pitching-moment parameter $\frac{-C_m}{\alpha - 20^\circ}$ increases the aerodynamic diving moment. An increase may be accomplished by increasing the area of the horizontal tail surfaces, putting the elevators down, or moving the center of gravity of the airplane forward. Increasing the pitching-moment inertia parameter $\frac{b^2}{k_Z^2 - k_X^2}$ decreases the inertia pitching moment, which has about the same effect as increasing the aerodynamic pitching moment. An increase may be accomplished by increasing the aspect ratio of the wing or by so distributing the weights in the airplane that $k_Z^2 - k_X^2$ will be reduced without changing the ratio $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$.

Increasing $\frac{-C_m}{\alpha - 20^\circ}$ or $\frac{b^2}{k_Z^2 - k_X^2}$ algebraically decreases the sideslip (changes it in the direction from inward toward outward) for all wings and all parameters used. (See figs. 7, 9, 15, and 17.) The rate of change of the sideslip depends upon the angle of attack, the airfoil section, and the other parameters used. The effect of increasing $\frac{-C_m}{\alpha - 20^\circ}$ or $\frac{b^2}{k_Z^2 - k_X^2}$ on C_n required is about the same for each wing with each set of parameters. With the first set of parameters the algebraic values of C_n required generally decrease as $\frac{-C_m}{\alpha - 20^\circ}$ and $\frac{b^2}{k_Z^2 - k_X^2}$ increase (figs. 8 and 10). With the second set of parameters, C_n required increases to a maximum at about $\frac{-C_m}{\alpha - 20^\circ} = 0.0030$ and then generally decreases as $\frac{-C_m}{\alpha - 20^\circ}$ increases further (fig. 16).

The parameter μ may be increased by increasing the wing loading or the span loading and by flying at higher altitudes. Increasing μ algebraically increases the sideslip for all wings with both sets of parameters, the rate of change being very large when μ is less than 5.0 (figs. 11 and 19). There is a slight tendency for C_n to increase algebraically with μ (figs. 12 and 20).

The rolling- and yawing-moment inertia parameter $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ may be increased by moving weight from the center of gravity out along the wing. Increasing this parameter algebraically decreases the sideslip except for the N.A.C.A. 0009 and N.A.C.A. 23018 wings at 50°, 60°, and 70° angle of attack. The rate of decrease is greatest for the Clark Y and N.A.C.A. 6718 wings (figs. 13 and 21). The algebraic value of C_n decreases as $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ increases except for the N.A.C.A. 0009 and N.A.C.A. 23018 wings at 50°, 60°, and 70° angle of attack. The rate of change of C_n with $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ is largest for the N.A.C.A. 6718 wing (figs. 14 and 22).

The results show that, generally, the algebraic value of the sideslip will increase for the wings in the following order: N.A.C.A. 0009, N.A.C.A. 23018, Clark Y, and N.A.C.A. 6718. It is interesting to note that the amount of camber of the wing sections increases in this same order (fig. 1). It appears, then, that the sideslip is more dependent upon the camber than upon the thickness of the wing section.

The general indications are that the algebraic values of C_n required decrease in the following order: N.A.C.A. 23018, N.A.C.A. 0009, N.A.C.A. 6718, and Clark Y when $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ is greater than 1.0; and N.A.C.A. 6718, N.A.C.A. 23018, N.A.C.A. 0009, and Clark Y when $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ is less than 1.0 (figs. 8, 10, 12, 14, 16, 18, 20, and 22). The reason for the order to change when $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ is greater or less than 1.0 is that the inertia yawing moment changes sign at that value.

<u>Prediction of the spinning characteristics of an airplane from the analysis</u>.- Prediction of the spinning char-

acteristics of an airplane in which any of these monoplane wings is used depends largely upon the aerodynamic yawing-moment characteristics of the particular airplane. The value of C_n required, as given in this report, is numerically equal and of opposite sign to the sum of the wing yawing moments and the inertia couples. At any angle of attack, when this value of C_n is supplied by the empennage, fuselage, and interference effects a steady spin will result provided that the equilibrium is stable; for any other value of C_n the airplane will not spin at that attitude. In order to insure against a steady spin in any attitude, a value of C_n opposing the spin must be provided that is larger than any attainable value of C_n required for that particular loading condition. The yawing moment supplied by the empennage, fuselage, and interference effects depends upon the sideslip, the size and shape of the fuselage and tail surfaces, the location of the horizontal tail surfaces with respect to the fuselage, fin, and rudder, the amount of fin area ahead of the center of gravity, the interference effects between the wing and the rest of the airplane, and the limits of the control movements. Data on some of these effects are reported in reference 6 and in references 8 to 13. The geometry of the spin indicates that the vertical tail surfaces should become more effective in producing a yawing moment opposing the spin as the sideslip becomes more outward. Fin area ahead of the center of gravity will give yawing moments aiding the spin if the sideslip is outward. (See reference 12, fig. 2.)

If the values of C_n for all parts of the airplane were known, the prediction of the spin would depend on the algebraic sum of the individual values of C_n for each part for each angle of attack. In any estimate, for normal airplanes, it will be found that a change in some factors will change the yawing moment for some parts in a sense to oppose the spin; whereas, for other parts, the effect will be reversed so that the magnitude of the change in C_n for each part must be considered.

The airplane least likely to spin is one with a wing having large algebraic values of C_n required and small algebraic values of sideslip.

The foregoing discussion shows the manner in which the value of C_n given by the fuselage and tail surfaces may be expected to vary with sideslip.

The results of the analysis indicate that the Clark Y wing, if used on a conventional monoplane, would always be more liable to give a dangerous spin than some one of the three wings investigated. The N.A.C.A. 0009 and the N.A.C.A. 23018 wings would generally be equally good. The N.A.C.A. 0009 wing usually gives a lower algebraic value of C_n required than the N.A.C.A. 23018 wing but it also has a lower algebraic value of sideslip (more outward). The N.A.C.A. 6718 wing may be superior to the other wings investigated when the value of $\frac{{k_Z}^2 - {k_Y}^2}{{k_Z}^2 - {k_X}^2}$ is less than about 0.6 and (1) the airplane has large fin area ahead of the center of gravity or (2) the fuselage and tail are so arranged that the change in yawing moment with sideslip is small.

CONCLUSIONS

The following conclusions are indicated by the analysis presented for a conventional monoplane having a rectangular wing with rounded tips:

1. A monoplane having either the N.A.C.A. 0009 wing or the N.A.C.A. 23018 wing appears to be less liable to spin dangerously than with either of the two other wings tested except the N.A.C.A. 6718 wing when $\frac{{k_Z}^2 - {k_Y}^2}{{k_Z}^2 - {k_X}^2} <$ about 0.6 and the yawing moment produced by the fuselage and tail surfaces does not change much with sideslip.

2. For all conditions investigated, one or more of the three wings tested was always superior to the Clark Y.

3. The algebraic value of the sideslip required for equilibrium in a spin increases (changes from an outward to an inward direction) as the camber increases.

Langley Memorial Aeronautical Laboratory,
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Langley Field, Va., December 23, 1937.

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TABLE I.

AERODYNAMIC CHARACTERISTICS OF THE N.A.C.A. 0009 MONOPLANE WING WITH ROUNDED TIPS

[Coefficients and moments given about the quarter-chord point of the lower surface of the wing]

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
$\beta = -10°$								
0.25	30	0.861	-0.021	0.010	-1.006	0.0051	-0.0229	-0.0008
	40	.794	-.023	.009	-1.055	.0029	-.0253	-.0028
	50	.819	.043	.001	-1.223	.0111	-.0307	-.0018
	60	.703	.048	.005	-1.325	.0191	-.0432	-.0025
	70	.592	.122	.003	-1.394	.0279	-.0507	-.0018
.50	30	.927	-.023	.009	-1.083	-.0368	-.0283	.0028
	40	.942	.000	.008	-1.230	-.0134	-.0276	-.0021
	50	.886	.037	.006	-1.335	-.0120	-.0341	-.0049
	60	.759	.045	.007	-1.440	-.0018	-.0488	-.0051
	70	.591	.091	.006	-1.478	.0071	-.0545	-.0045
.75	30	1.014	-.033	.016	-1.190	-.1147	-.0313	.0007
	40	1.150	.025	.011	-1.480	-.0614	-.0380	-.0007
	50	1.174	.105	.002	-1.702	-.0343	-.0561	-.0065
	60	.970	.134	.002	-1.707	-.0297	-.0597	-.0074
	70	.659	.087	-.001	-1.684	-.0161	-.0624	-.0058
1.00	30	1.104	-.004	.024	-1.278	-.2324	-.0340	.0005
	40	1.455	.069	.025	-1.841	-.1436	-.0522	-.0018
	50	1.547	.234	.013	-2.128	-.0940	-.0779	-.0094
	60	1.259	.221	.007	-2.135	-.0649	-.0751	-.0071
	70	.701	.017	.005	-2.003	-.0476	-.0792	-.0068
$\beta = -5°$								
0.25	30	0.855	-0.002	0.004	-0.989	-0.0053	-0.0215	-0.0011
	40	.822	-.002	.004	-1.075	-.0070	-.0260	-.0023
	50	.807	.035	.004	-1.214	.0002	-.0333	-.0031
	60	.732	.073	.003	-1.337	.0058	-.0439	-.0028
	70	.547	.078	.004	-1.384	.0142	-.0509	-.0023
.50	30	.935	-.016	.007	-1.089	-.0413	-.0267	.0027
	40	.953	.012	.004	-1.235	-.0258	-.0322	-.0029
	50	.879	.040	.005	-1.321	-.0224	-.0383	-.0055
	60	.766	.057	.006	-1.434	-.0120	-.0502	-.0049
	70	.565	.067	.005	-1.469	-.0044	-.0565	-.0044
.75	30	1.114	.012	.018	-1.279	-.1170	-.0311	.0009
	40	1.224	.067	.012	-1.542	-.0686	-.0437	-.0017
	50	1.139	.093	.008	-1.662	-.0433	-.0508	-.0058
	60	.962	.124	.007	-1.707	-.0383	-.0594	-.0068
	70	.697	.123	.005	-1.698	-.0242	-.0621	-.0061
1.00	30	1.222	.018	.031	-1.401	-.2338	-.0460	-.0008
	40	1.571	.112	.025	-1.957	-.1427	-.0516	-.0025
	50	1.514	.192	.015	-2.126	-.0964	-.0757	-.0060
	60	1.220	.180	.010	-2.128	-.0708	-.0798	-.0078
	70	.732	.036	.009	-2.042	-.0518	-.0798	-.0078
$\beta = 0°$								
0.25	30	0.825	-0.009	0.004	-0.958	-0.0146	-0.0233	-0.0012
	40	.815	-.002	-.003	-1.065	-.0163	-.0285	-.0022
	50	.756	-.005	.002	-1.182	-.0121	-.0347	-.0029
	60	.683	.029	-.001	-1.316	-.0099	-.0450	-.0027
	70	.522	.052	-.002	-1.381	-.0005	-.0528	-.0028
.50	30	.928	-.015	.006	-1.080	-.0450	-.0266	.0023
	40	.923	.005	.003	-1.200	-.0393	-.0337	-.0043
	50	.848	.017	.002	-1.300	-.0321	-.0393	-.0054
	60	.736	.035	-.002	-1.412	-.0258	-.0488	-.0051
	70	.591	.083	-.002	-1.503	-.0147	-.0578	-.0047
.75	30	1.125	.006	.012	-1.295	-.1160	-.0334	.0005
	40	1.142	.010	.004	-1.483	-.0732	-.0446	-.0025
	50	1.145	.102	.002	-1.661	-.0524	-.0496	-.0067
	60	.856	.044	.003	-1.634	-.0469	-.0566	-.0075
	70	.635	.054	.003	-1.706	-.0347	-.0665	-.0070
1.00	30	1.289	.038	.023	-1.466	-.2309	-.0495	-.0008
	40	1.562	.101	.015	-1.954	-.1411	-.0656	-.0040
	50	1.585	.254	.008	-2.164	-.1010	-.0721	-.0079
	60	1.123	.097	.004	-2.076	-.0782	-.0771	-.0094
	70	.726	.029	.007	-2.042	-.0569	-.0860	-.0081
$\beta = 5°$								
0.25	30	0.812	-0.006	0.005	-0.942	-0.0262	-0.0216	-0.0008
	40	.821	.015	.004	-1.059	-.0292	-.0240	-.0022
	50	.794	.028	.005	-1.202	-.0271	-.0328	-.0030
	60	.649	.007	.004	-1.285	-.0263	-.0446	-.0028
	70	.422	-.038	.005	-1.338	-.0225	-.0534	-.0028
.50	30	.889	-.027	.004	-1.042	-.0492	-.0262	.0020
	40	.858	-.023	.002	-1.139	-.0492	-.0316	-.0043
	50	.855	.035	.005	-1.289	-.0454	-.0372	-.0057
	60	.720	.026	.003	-1.396	-.0367	-.0483	-.0056
	70	.561	.054	.001	-1.490	-.0200	-.0572	-.0050
.75	30	1.090	.002	.008	-1.259	-.1154	-.0316	-.0003
	40	1.151	.041	.005	-1.467	-.0806	-.0437	-.0039
	50	1.047	.041	.004	-1.580	-.0634	-.0480	-.0080
	60	.914	.103	.005	-1.650	-.0591	-.0578	-.0084
	70	.659	.078	.004	-1.710	-.0376	-.0673	-.0074
1.00	30	1.197	-.006	.015	-1.385	-.2262	-.0477	.0000
	40	1.514	.081	.014	-1.908	-.1396	-.0666	-.0046
	50	1.364	.087	.008	-2.020	-.1042	-.0714	-.0081
	60	1.164	.153	.012	-2.064	-.0839	-.0782	-.0094
	70	.725	.030	.006	-2.036	-.0628	-.0826	-.0094
$\beta = 10°$								
0.25	30	0.779	-0.019	-0.002	-0.910	-0.0373	-0.0162	-0.0002
	40	.805	.012	-.002	-1.041	-.0419	-.0275	-.0025
	50	.769	.012	.001	-1.182	-.0423	-.0344	-.0033
	60	.634	-.003	.000	-1.272	-.0434	-.0393	-.0035
	70	.466	.008	.004	-1.342	-.0408	-.0491	-.0035
.50	30	.893	-.017	-.001	-1.040	-.0539	-.0255	.0012
	40	.871	.008	-.003	-1.131	-.0574	-.0310	-.0051
	50	.834	.030	-.003	-1.262	-.0543	-.0396	-.0060
	60	.730	.043	-.004	-1.385	-.0418	-.0473	-.0058
	70	.555	.058	-.004	-1.463	-.0255	-.0570	-.0052
.75	30	1.096	-.001	.002	-1.267	-.1099	-.0314	-.0005
	40	1.125	.056	.001	-1.422	-.0877	-.0402	-.0055
	50	.994	.039	-.004	-1.499	-.0766	-.0451	-.0098
	60	.833	.040	-.004	-1.597	-.0669	-.0552	-.0098
	70	.621	.047	-.002	-1.687	-.0391	-.0664	-.0083
1.00	30	1.290	.015	.012	-1.482	-.2138	-.0475	.0005
	40	1.431	.062	.005	-1.815	-.1408	-.0622	-.0065
	50	1.295	.080	-.001	-1.921	-.1129	-.0663	-.0106
	60	1.028	.048	.002	-1.972	-.0917	-.0758	-.0115
	70	.725	.037	.000	-2.020	-.0621	-.0850	-.0096

TABLE II. AERODYNAMIC CHARACTERISTICS OF THE N.A.C.A. 23018 MONOPLANE WING WITH ROUNDED TIPS.

[Coefficients and moments given about the quarter-chord point of the lower surface of the wing]

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
$\beta = -10°$								
0.25	30	0.704	-0.037	0.008	-0.806	0.0108	-0.0151	0.0036
	40	.704	-.012	.007	-.929	.0008	-.0207	-.0030
	50	.688	.017	.009	-1.050	.0082	-.0270	-.0040
	60	.612	.046	.004	-1.145	.0167	-.0341	-.0034
	70	.401	-.002	.003	-1.179	.0244	-.0435	-.0028
.50	30	.877	.011	.005	-1.006	-.0287	-.0201	.0053
	40	.843	.013	.003	-1.089	-.0131	-.0255	-.0011
	50	.770	.023	.002	-1.171	-.0133	-.0329	-.0081
	60	.678	.047	.006	-1.274	-.0014	-.0423	-.0084
	70	.501	.059	.006	-1.302	.0080	-.0489	-.0076
.75	30	.994	.009	.011	-1.143	-.1047	-.0253	.0008
	40	1.124	.069	.012	-1.410	-.0528	-.0364	.0010
	50	1.010	.061	.007	-1.499	-.0243	-.0461	-.0049
	60	.754	.004	.004	-1.501	-.0217	-.0547	-.0104
	70	.536	.021	.006	-1.512	-.0135	-.0609	-.0101
1.00	30	1.145	.048	.012	-1.295	-.2168	-.0312	-.0005
	40	1.375	.097	.023	-1.712	-.1381	-.0469	-.0009
	30	1.213	.069	.016	-1.806	-.0868	-.0601	-.0034
	40	.848	-.061	.011	-1.801	-.0562	-.0742	-.0104
	50	.596	-.028	.012	-1.820	-.0405	-.0802	-.0112
$\beta = -5°$								
0.25	30	0.692	-0.038	0.001	-0.821	0.0008	-0.0169	0.0016
	40	.700	-.016	.002	-.926	-.0078	-.0238	-.0041
	50	.687	.011	.002	-1.056	-.0017	-.0319	-.0050
	60	.628	.054	.001	-1.162	.0044	-.0406	-.0046
	70	.502	.088	.000	-1.227	.0115	-.0482	-.0042
.50	30	.846	-.017	.005	-.987	-.0343	-.0205	.0039
	40	.819	-.008	-.001	-1.074	-.0254	-.0286	-.0038
	50	.762	.013	-.001	-1.170	-.0232	-.0353	-.0088
	60	.673	.045	-.001	-1.269	-.0123	-.0438	-.0095
	70	.495	.051	-.001	-1.308	-.0030	-.0515	-.0075
.75	30	1.102	.041	.007	-1.248	-.1079	-.0320	.0003
	40	1.162	.080	.006	-1.450	-.0587	-.0377	-.0018
	50	1.010	.074	.002	-1.483	-.0398	-.0430	-.0081
	60	.731	-.002	.005	-1.465	-.0360	-.0535	-.0112
	70	.467	-.072	.012	-1.564	-.0197	-.0617	-.0095
1.00	30	1.343	.116	.006	-1.484	-.2158	-.0393	-.0024
	40	1.449	.106	.014	-1.802	-.1374	-.0535	-.0034
	50	1.259	.084	.007	-1.858	-.0905	-.0605	-.0067
	60	.837	-.072	.005	-1.799	-.0638	-.0715	-.0113
	70	.516	-.109	.014	-1.807	-.0444	-.0786	-.0105

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
$\beta = 0°$								
0.25	30	0.677	-0.039	-0.006	-0.804	-0.0099	-0.0160	0.0007
	40	.701	-.012	-.007	-.926	-.0175	-.0208	-.0035
	50	.700	.020	-.006	-1.064	-.0135	-.0297	-.0045
	60	.608	.041	-.005	-1.145	-.0101	-.0364	-.0044
	70	.468	.059	-.002	-1.206	-.0037	-.0460	-.0041
.50	30	.869	.001	-.002	-1.004	-.0325	-.0214	.0024
	40	.825	.001	-.008	-1.076	-.0374	-.0272	-.0045
	50	.753	.007	-.005	-1.163	-.0330	-.0345	-.0086
	60	.667	.051	-.003	-1.247	-.0236	-.0424	-.0086
	70	.496	.051	-.002	-1.311	-.0156	-.0512	-.0076
.75	30	1.046	.016	.005	-1.199	-.1107	-.0316	.0000
	40	1.151	.082	.002	-1.433	-.0684	-.0443	-.0040
	50	.936	.030	-.004	-1.420	-.0575	-.0458	-.0097
	60	.764	.025	.000	-1.485	-.0435	-.0524	-.0120
	70	.469	-.042	.002	-1.486	-.0305	-.0624	-.0097
1.00	30	1.283	.077	.004	-1.438	-.2145	-.0451	-.0001
	40	1.425	.085	.007	-1.789	-.1385	-.0528	-.0037
	50	1.175	.009	.005	-1.817	-.0965	-.0682	-.0076
	60	.812	-.087	.002	-1.773	-.0801	-.0709	-.0123
	70	.536	-.096	.005	-1.832	-.0479	-.0809	-.0094
$\beta = 5°$								
0.25	30	0.680	-0.030	-0.007	-0.802	-0.0226	-0.0167	0.0002
	40	.675	-.016	-.005	-.895	-.0298	-.0188	-.0035
	50	.693	.021	-.006	-1.052	-.0278	-.0301	-.0045
	60	.608	.043	-.002	-1.141	-.0273	-.0356	-.0046
	70	.469	.064	-.001	-1.197	-.0226	-.0447	-.0043
.50	30	.856	.000	-.002	-.989	-.0495	-.0214	.0014
	40	.809	.006	-.006	-1.051	-.0483	-.0267	-.0051
	50	.760	.025	-.004	-1.153	-.0440	-.0338	-.0082
	60	.661	.049	-.003	-1.236	-.0382	-.0438	-.0086
	70	.508	.062	-.003	-1.315	-.0227	-.0533	-.0074
.75	30	1.003	-.002	.004	-1.159	-.1139	-.0301	-.0003
	40	1.056	.037	.000	-1.348	-.0797	-.0389	-.0053
	50	.914	.032	.000	-1.384	-.0712	-.0425	-.0106
	60	.722	.012	.000	-1.422	-.0558	-.0530	-.0119
	70	.501	-.010	.003	-1.491	-.0383	-.0635	-.0107
1.00	30	1.216	.029	.005	-1.387	-.2149	-.0428	.0007
	40	1.408	.085	.009	-1.767	-.1416	-.0594	-.0070
	50	1.128	-.009	.004	-1.765	-.1054	-.0635	-.0103
	60	.862	-.023	.001	-1.763	-.0862	-.0690	-.0133
	70	.564	-.055	.006	-1.800	-.0617	-.0783	-.0111

TABLE III.

AERODYNAMIC CHARACTERISTICS OF THE N.A.C.A. 6718 MONOPLANE WING WITH ROUNDED TIPS

[Coefficients and moments given about the quarter-chord point of the lower surface of the wing]

$\frac{\Omega b}{2V}$	α (deg.)	$C_{x''}$	C_x	C_Y	C_Z	C_l	C_m	C_n
				β = -10°				
0.25	30	1.070	0.036	-0.006	-1.215	0.0270	-0.0392	0.0077
	40	1.020	.053	.000	-1.286	.0189	- .0450	-.0022
	50	.942	.092	.002	-1.356	.0207	- .0497	-.0028
	60	.791	.111	.000	-1.391	.0267	- .0547	-.0016
	70	.586	.109	.002	-1.417	.0339	- .0596	-.0040
.50	30	1.172	.045	- .005	-1.328	-.0171	- .0424	.0026
	40	1.121	.056	- .005	-1.416	.0040	- .0509	-.0003
	50	1.018	.075	- .006	-1.495	.0123	- .0563	-.0052
	60	.848	.099	- .001	-1.524	.0138	- .0607	-.0050
	70	.609	.094	.000	-1.523	.0216	- .0650	-.0360
.75	30	1.300	.040	- .012	-1.478	-.0896	- .0538	-.0063
	40	1.341	.070	- .006	-1.692	-.0390	- .0607	-.0031
	50	1.264	.130	- .010	-1.812	-.0082	- .0653	-.0023
	60	1.057	.146	- .004	-1.861	.0082	- .0735	-.0035
	70	.671	.064	.003	-1.784	.0106	- .0772	-.0026
1.00	30	1.449	.062	- .020	-1.637	-.2005	- .0629	-.0128
	40	1.625	.095	- .006	-2.043	-.1178	- .0728	-.0076
	50	1.522	.145	- .005	-2.194	-.0608	- .0832	-.0018
	60	1.242	.145	- .011	-2.233	-.0282	- .0912	.0013
	70	.772	.020	.001	-2.205	.0039	- .0979	.0035
				β = -5°				
0.25	30	1.033	0.030	-0.010	-1.175	0.0167	-0.0385	0.0050
	40	.978	.048	- .005	-1.237	.0022	- .0434	-.0029
	50	.938	.094	- .002	-1.347	.0083	- .0482	-.0031
	60	.807	.121	- .004	-1.405	.0127	- .0551	-.0027
	70	.610	.131	- .001	-1.425	.0198	- .0608	-.0016
.50	30	1.139	.027	- .006	-1.300	-.0186	- .0441	.0018
	40	1.104	.061	- .011	-1.390	-.0058	- .0475	-.0011
	50	1.033	.103	- .004	-1.484	-.0009	- .0539	-.0048
	60	.857	.110	- .002	-1.523	.0034	- .0618	-.0053
	70	.624	.108	.002	-1.527	.0097	- .0660	-.0031
.75	30	1.304	.043	- .005	-1.481	-.0869	- .0584	-.0052
	40	1.353	.084	- .007	-1.696	-.0434	- .0625	-.0039
	50	1.251	.133	- .015	-1.788	-.0201	- .0707	-.0037
	60	1.022	.126	- .008	-1.825	-.0041	- .0745	-.0031
	70	.662	.059	- .002	-1.774	.0013	- .0792	-.0023
1.00	30	1.473	.059	- .010	-1.667	-.1977	- .0683	-.0102
	40	1.716	.140	- .011	-2.122	-.1175	- .0835	-.0084
	50	1.572	.186	- .013	-2.225	-.0662	- .0892	-.0033
	60	1.259	.161	- .017	-2.239	-.0366	- .0992	.0013
	70	.747	-.011	- .012	-2.213	-.0035	- .1028	.0026
				β = 0°				
0.25	30	1.008	0.021	-0.008	-1.152	0.0069	-0.0370	0.0036
	40	.964	.039	- .001	-1.226	-.0074	- .0428	-.0031
	50	.923	.088	- .001	-1.331	-.0058	- .0483	-.0040
	60	.795	.112	- .001	-1.396	-.0036	- .0572	-.0031
	70	.597	.118	- .001	-1.421	.0042	- .0597	-.0024
.50	30	1.132	.025	- .005	-1.293	-.0213	- .0410	.0018
	40	1.088	.051	- .008	-1.377	-.0161	- .0491	-.0023
	50	1.010	.087	- .005	-1.467	-.0129	- .0551	-.0058
	60	.844	.104	.000	-1.508	-.0123	- .0593	-.0055
	70	.641	.115	.002	-1.659	-.0012	- .0661	-.0030
.75	30	1.344	.052	- .002	-1.522	-.0855	- .0589	-.0043
	40	1.330	.073	- .006	-1.675	-.0495	- .0624	-.0048
	50	1.205	.114	- .009	-1.740	-.0325	- .0661	-.0047
	60	1.036	.142	- .006	-1.825	-.0142	- .0738	-.0043
	70	.686	.078	.000	-1.792	-.0110	- .0784	-.0036
1.00	30	1.525	.070	- .007	-1.721	-.1964	- .0666	-.0088
	40	1.653	.100	- .005	-2.074	-.1172	- .0799	-.0090
	50	1.484	.126	- .006	-2.158	-.0746	- .0865	-.0058
	60	1.232	.144	- .011	-2.214	-.0443	- .0930	-.0005
	70	.809	.054	- .001	-2.218	-.0127	- .0990	.0017

$\frac{\Omega b}{2V}$	α (deg.)	$C_{x''}$	C_x	C_Y	C_Z	C_l	C_m	C_n
				β = 5°				
0.25	30	0.975	0.023	-0.002	-1.113	-0.0048	-0.0358	0.0036
	40	.947	.044	.002	-1.200	-.0213	- .0412	-.0030
	50	.913	.085	.004	-1.319	-.0217	- .0457	-.0044
	60	.794	.109	.003	-1.400	-.0217	- .0544	-.0033
	70	.594	.116	.007	-1.416	-.0182	- .0597	-.0026
.50	30	1.122	.035	.000	-1.274	-.0254	- .0426	.0014
	40	1.062	.057	- .004	-1.338	-.0279	- .0444	-.0029
	50	.976	.086	.001	-1.415	-.0288	- .0498	-.0065
	60	.842	.106	.003	-1.500	-.0231	- .0601	-.0056
	70	.637	.113	.004	-1.553	-.0073	- .0655	-.0029
.75	30	1.381	.072	- .004	-1.553	-.0844	- .0543	-.0040
	40	1.316	.085	.000	-1.647	-.0572	- .0608	-.0062
	50	1.160	.099	- .002	-1.685	-.0436	- .0649	-.0058
	60	.969	.111	.001	-1.747	-.0314	- .0709	-.0070
	70	.673	.068	.006	-1.784	-.0154	- .0794	-.0034
1.00	30	1.595	.095	- .010	-1.786	-.1910	- .0683	-.0091
	40	1.691	.133	.005	-2.095	-.1162	- .0816	-.0096
	50	1.434	.108	.006	-2.102	-.0844	- .0873	-.0084
	60	1.176	.124	.003	-2.139	-.0544	- .0902	-.0035
	70	.753	.021	.011	-2.143	-.0238	- .0967	.0008
				β = 10°				
0.25	30	0.950	0.023	-0.010	-1.083	-0.0163	-0.0355	0.0043
	40	.911	.025	.000	-1.169	-.0355	- .0412	-.0034
	50	.891	.061	.005	-1.313	-.0369	- .0470	-.0040
	60	.767	.094	.011	-1.372	-.0383	- .0530	-.0037
	70	.584	.107	.010	-1.413	-.0369	- .0593	.0031
.50	30	1.103	.040	- .005	-1.250	-.0297	- .0430	.0017
	40	1.014	.041	- .004	-1.289	-.0386	- .0457	-.0034
	50	.941	.063	.005	-1.388	-.0394	- .0551	-.0068
	60	.801	.080	.007	-1.464	-.0290	- .0563	-.0062
	70	.616	.095	.004	-1.540	-.0128	- .0647	-.0030
.75	30	1.328	.067	- .001	-1.495	-.0851	- .0529	-.0045
	40	1.257	.079	.001	-1.575	-.0645	- .0596	-.0069
	50	1.128	.106	.000	-1.628	-.0580	- .0621	-.0075
	60	.948	.106	.007	-1.713	-.0433	- .0698	-.0077
	70	.707	.105	.005	-1.779	-.0181	- .0780	-.0035
1.00	30	1.550	.100	- .020	-1.732	-.1911	- .0688	-.0086
	40	1.618	.117	.011	-2.014	-.1164	- .0602	-.0082
	50	1.375	.103	.004	-2.017	-.0448	- .0811	-.0106
	60	1.168	.122	.014	-2.125	-.0634	- .0903	-.0061
	70	.825	.097	.013	-2.146	-.0313	- .0959	-.0012

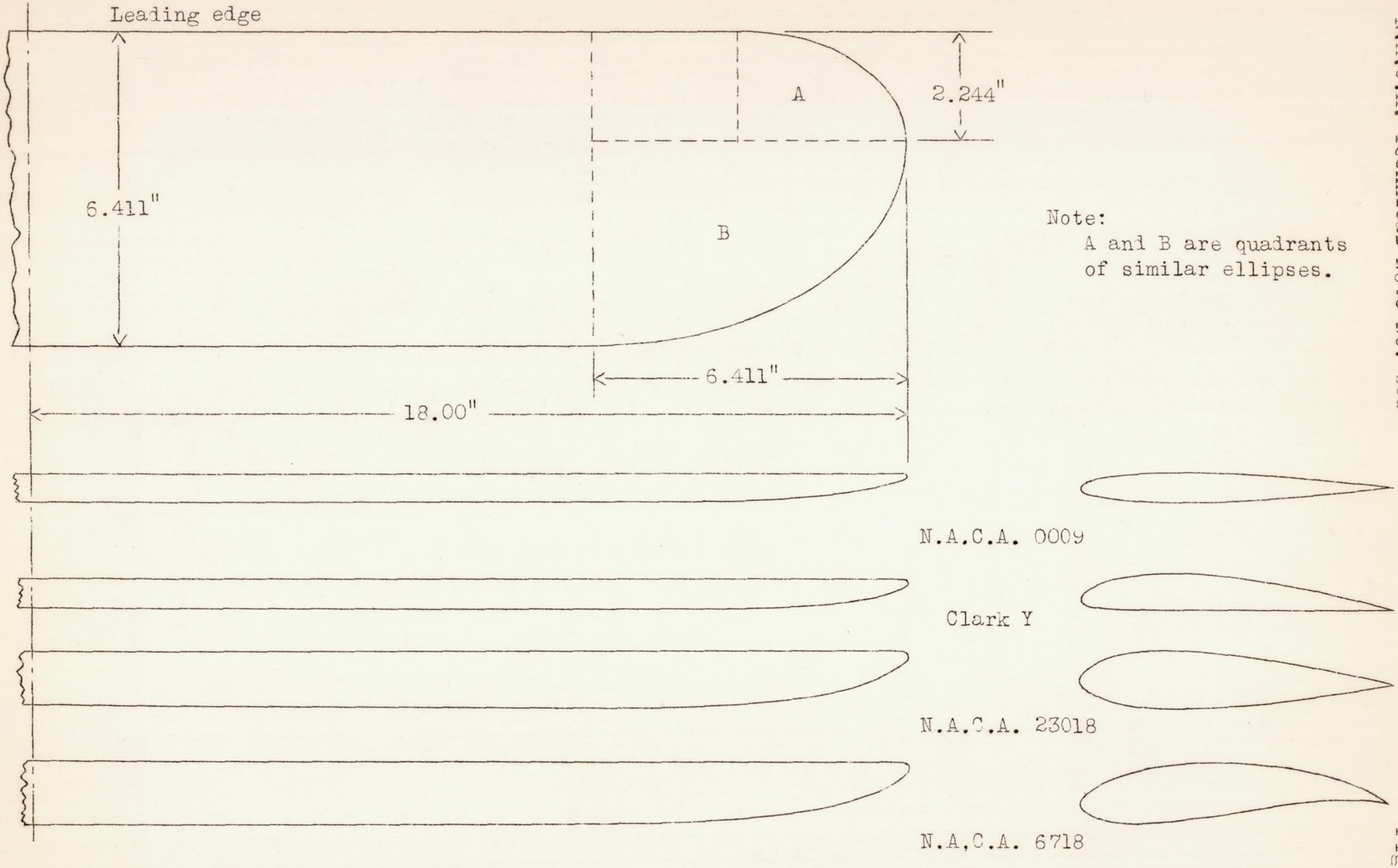

Figure 1.- The Clark Y, N.A.C.A. 0009, N.A.C.A. 23018, and N.A.C.A. 6718 wings with rounded tips.

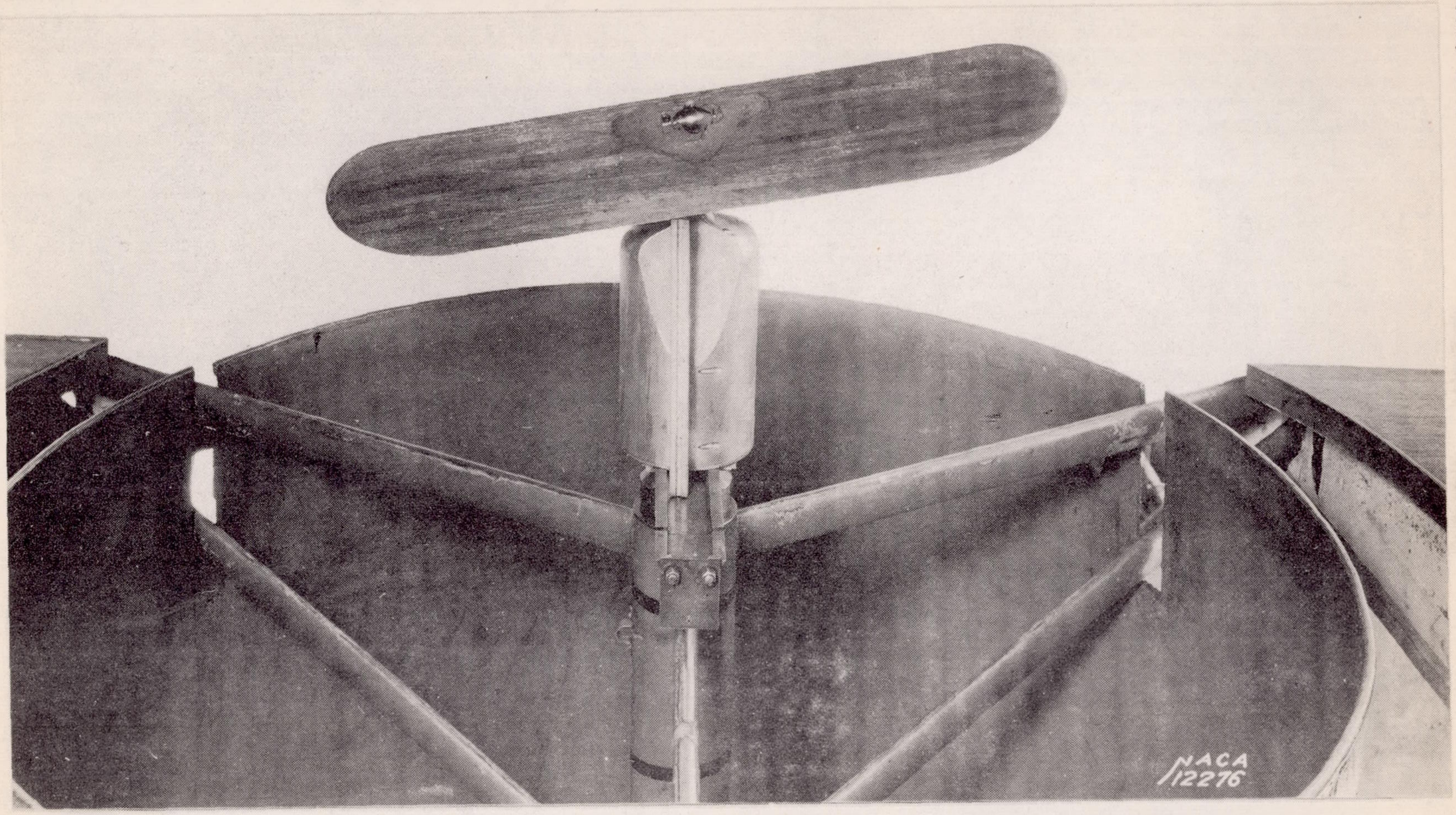

Figure 2.- The rectangular N.A.C.A. 6718 wing with rounded tips mounted on the spinning balance.

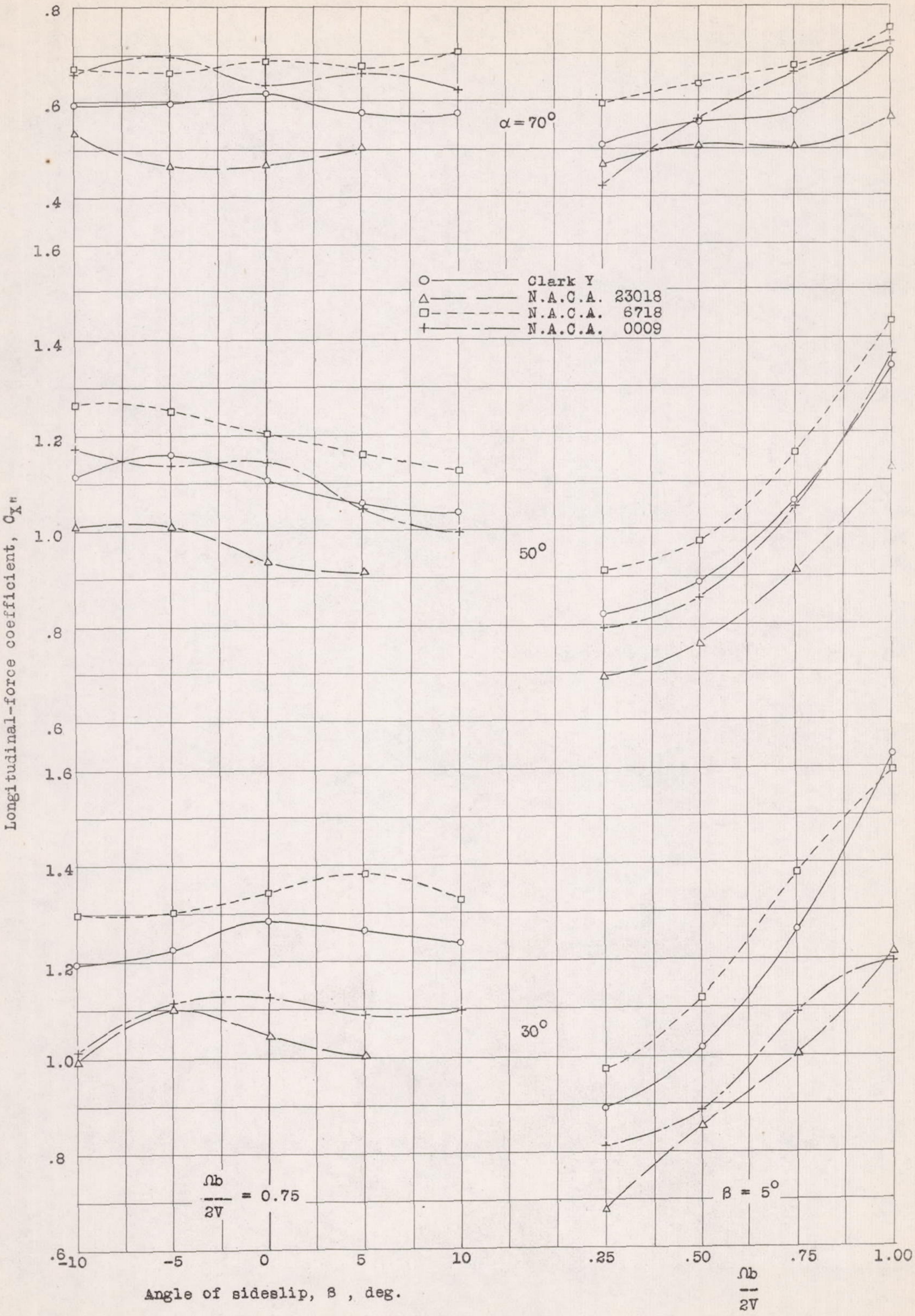

Figure 3. - Variation of longitudinal-force coefficient $C_{X''}$ (earth axes) with angle of sideslip and $\frac{\Omega b}{2V}$.

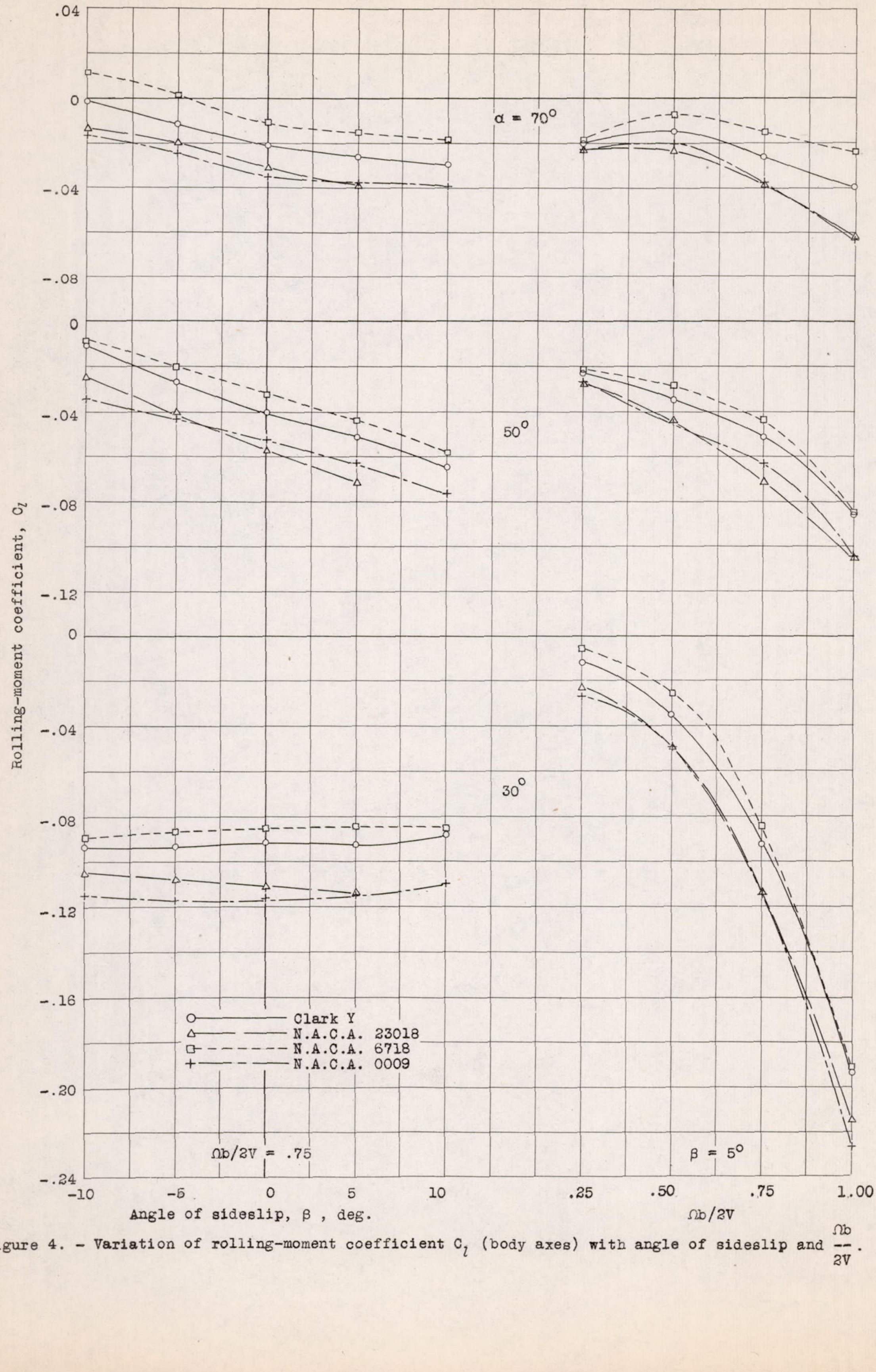

Figure 4. - Variation of rolling-moment coefficient C_l (body axes) with angle of sideslip and $\frac{\Omega b}{2V}$.

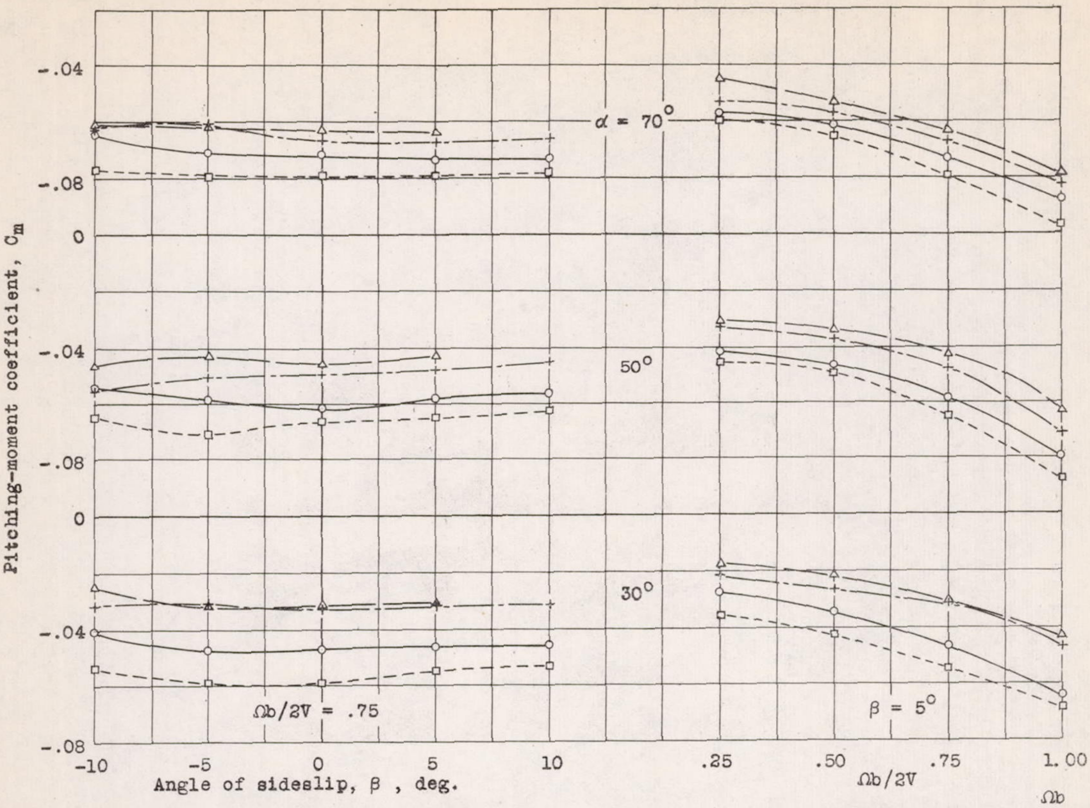

Figure 5. - Variation of pitching-moment coefficient C_m (body axes) with angle of sideslip and $\frac{\Omega b}{2V}$.

○——— Clark Y △—— —— N.A.C.A.23018 □- - - - - N.A.C.A. 6718 +———— N.A.C.A. 0009

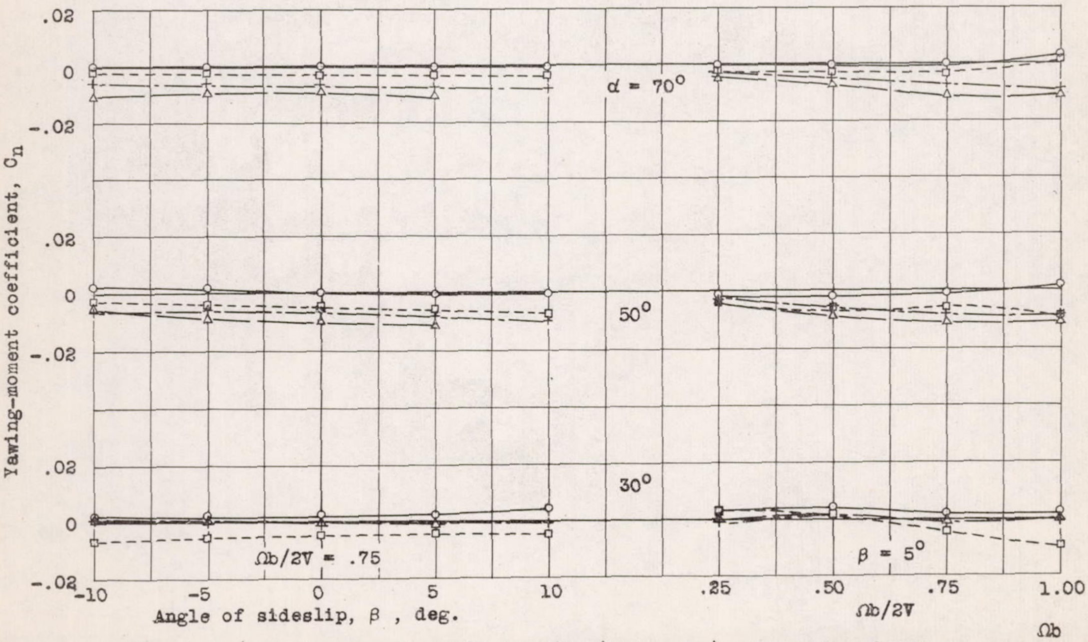

Figure 6. - Variation of yawing-moment coefficient C_n (body axes) with angle of sideslip and $\frac{\Omega b}{2V}$

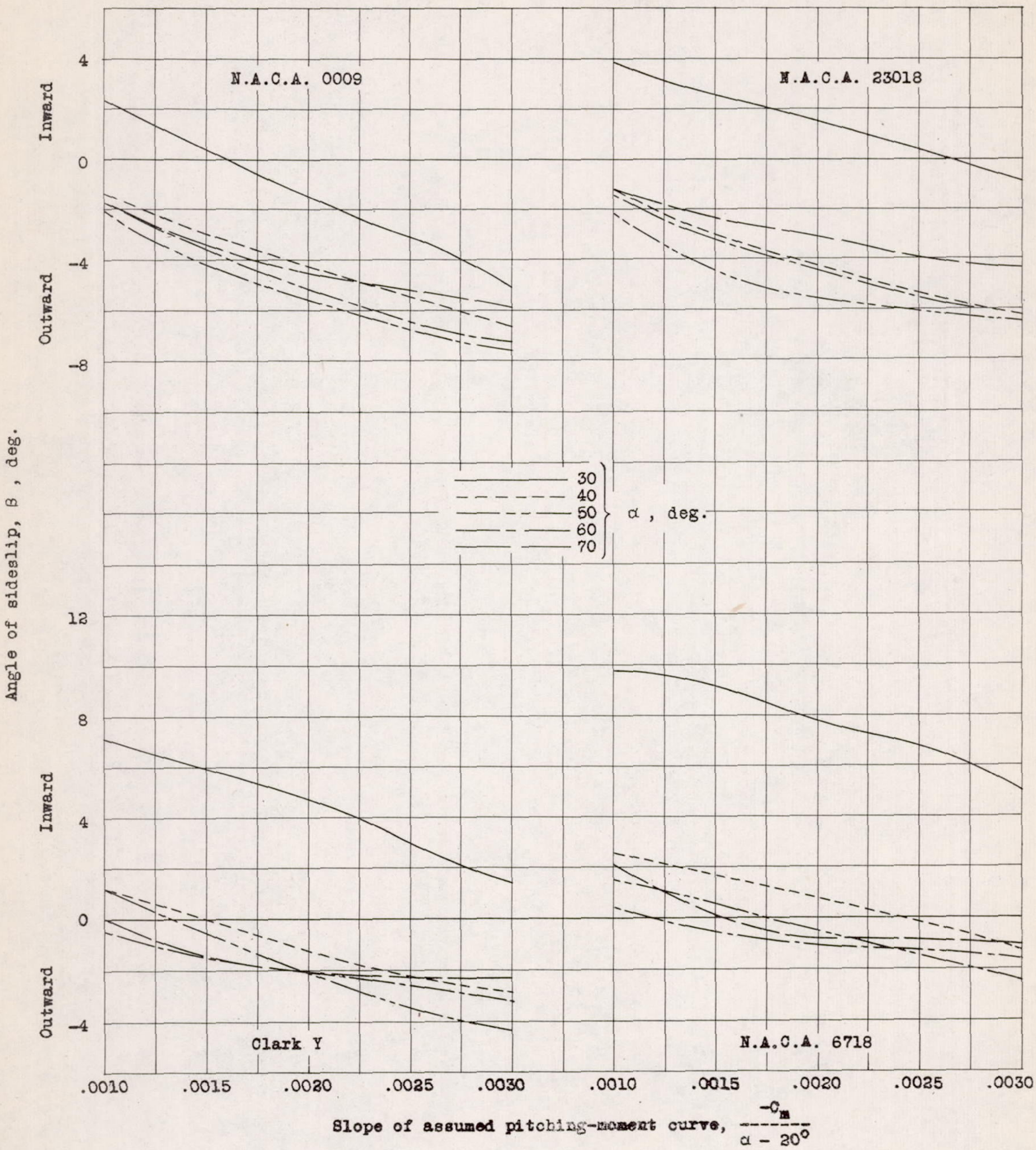

Figure 7. - Effect of pitching-moment coefficient upon sideslip necessary for equilibrium in a spin.

$\mu = 5 \qquad C_L = C_{X'} \qquad \frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0 \qquad \frac{b^2}{k_Z^2 - k_X^2} = 80$

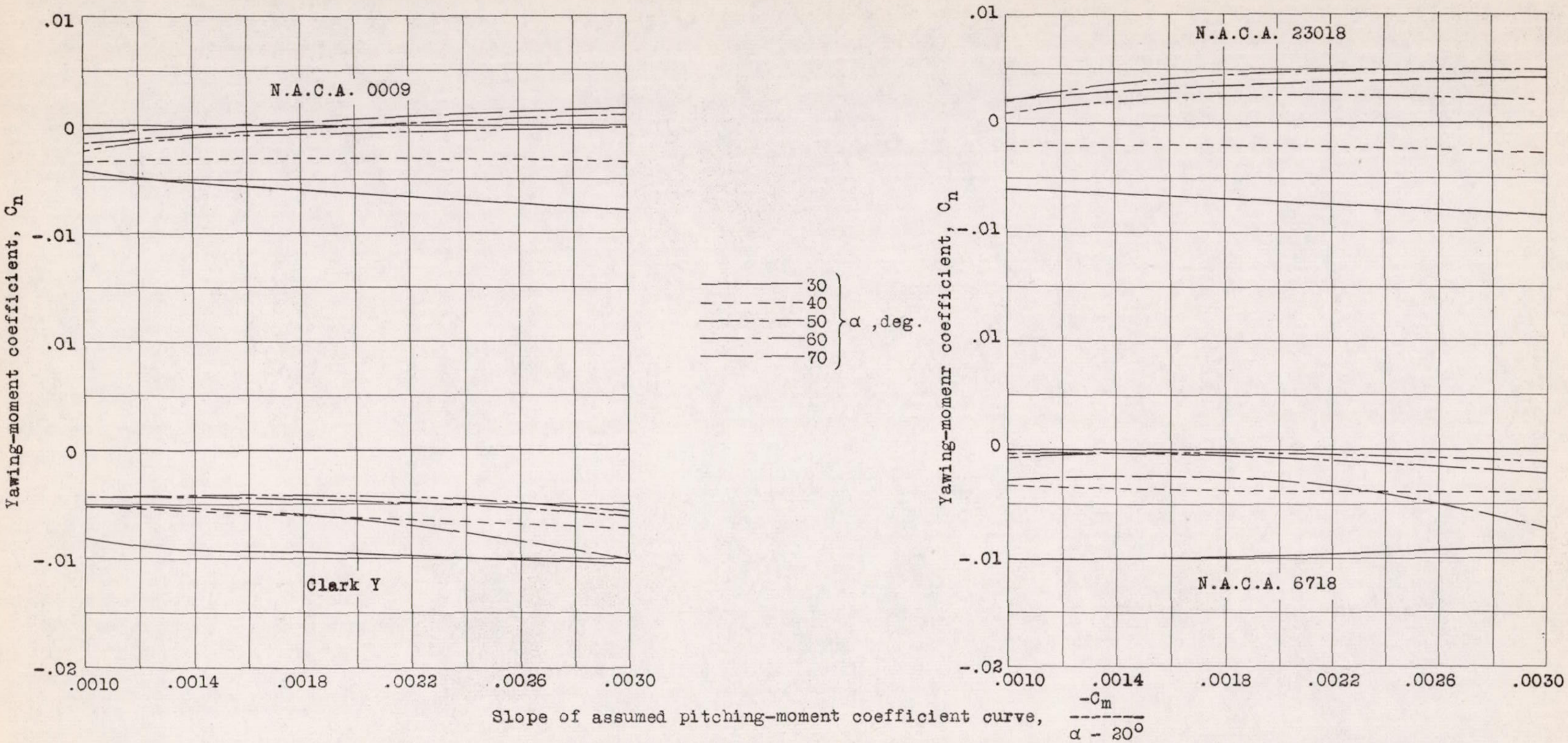

Figure 8. - Effect of pitching-moment coefficient upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin. $\mu = 5$ $C_L = C_{X''}$ $\dfrac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$ $\dfrac{b^2}{k_Z^2 - k_X^2} = 80$

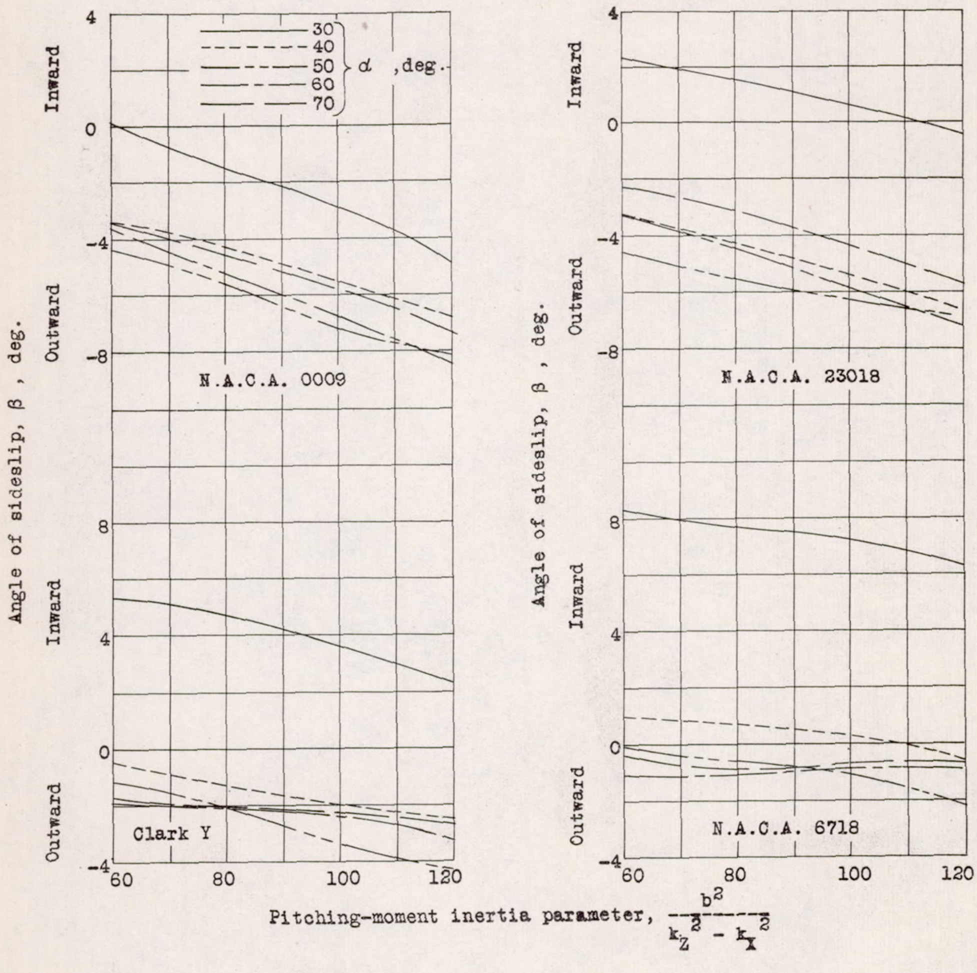

Figure 9. - Effect of pitching-moment inertia parameter upon sideslip necessary for equilibrium in a spin. $\mu = 5$ $C_L = C_X$; $C_m = -0.0020 (\alpha - 20^\circ)$

$$\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$$

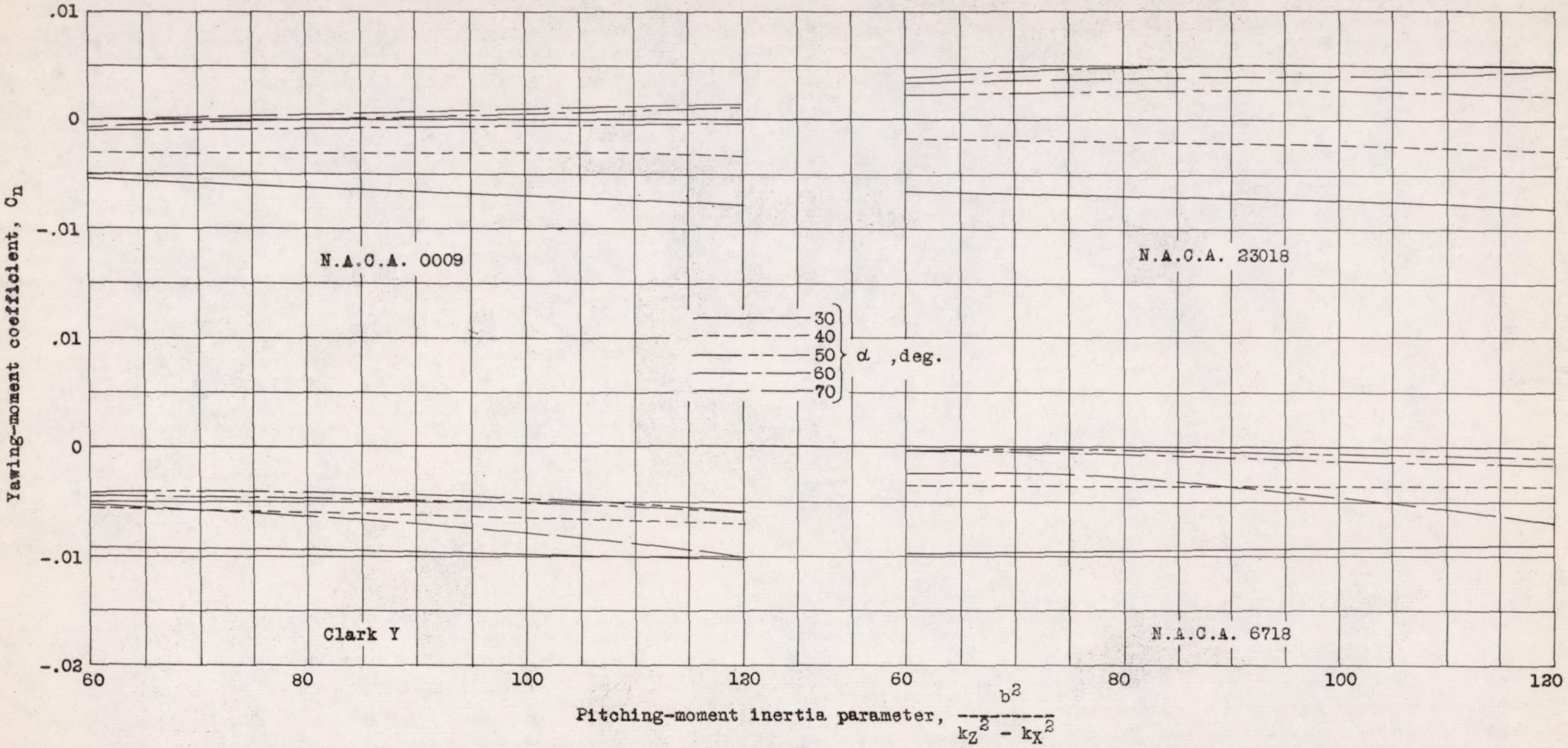

Figure 10. - Effect of pitching-moment inertia parameter upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin. $\mu = 5$ $C_L = C_{X''}$ $C_m = -0.0020\ (\alpha - 20^\circ)$ $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$

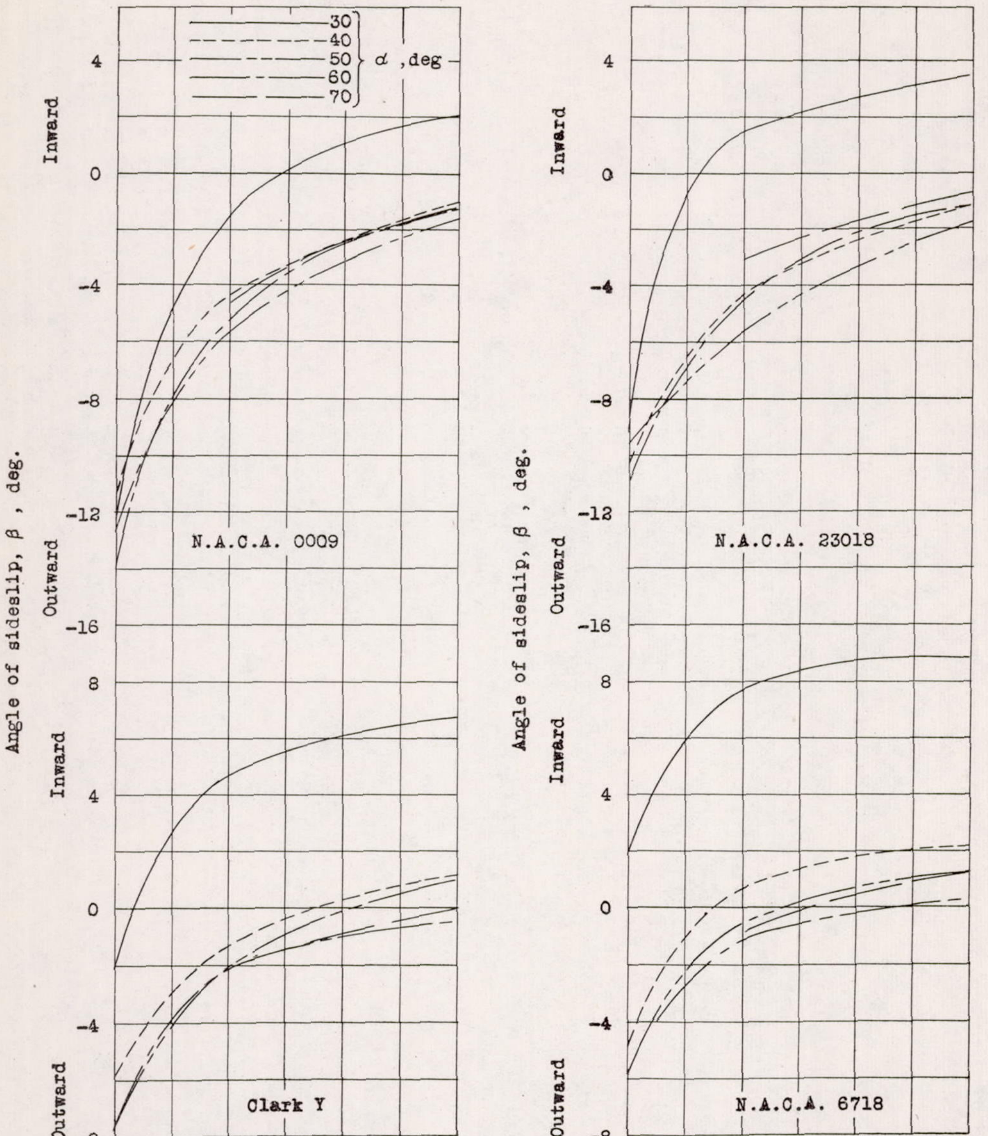

Figure 11. - Effect of relative density of airplane upon sideslip necessary for equilibrium in a spin. $C_m = -0.0020 (\alpha - 20^\circ)$ $C_L = C_X$.

$\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$ $\frac{b^2}{k_Z^2 - k_X^2} = 80$

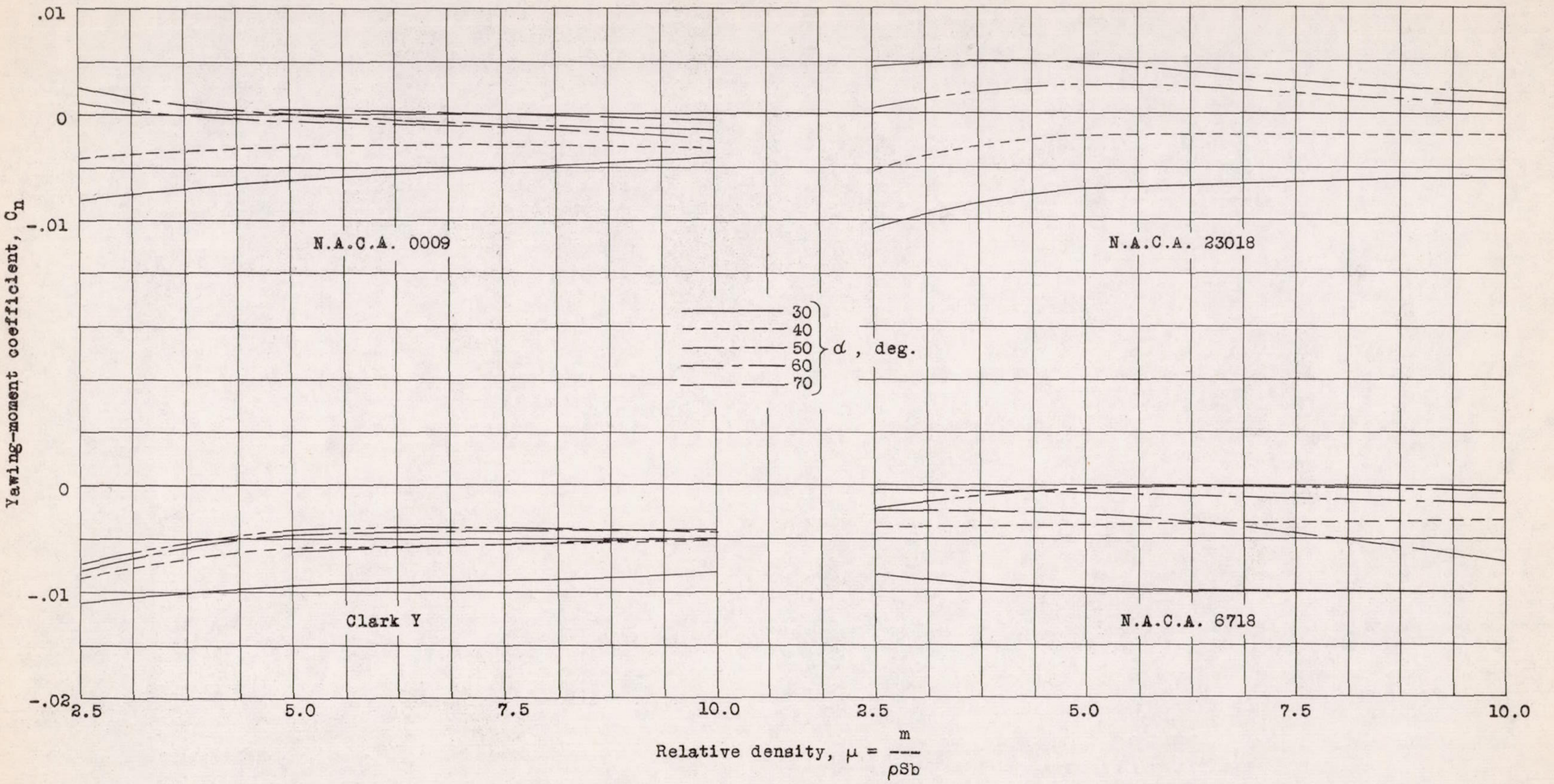

Figure 12.- Effect of relative density of airplane upon yawing-moment coefficient that must be supplied by parts of the airplane other than the wing for equilibrium in a spin. $C_L = C_{X''}$ $C_m = -0.0020\ (\alpha - 20^\circ)$ $\frac{k_Z{}^2 - k_Y{}^2}{k_Z{}^2 - k_X{}^2} = 1.0$ $\frac{b^2}{k_Z{}^2 - k_X{}^2} = 80$

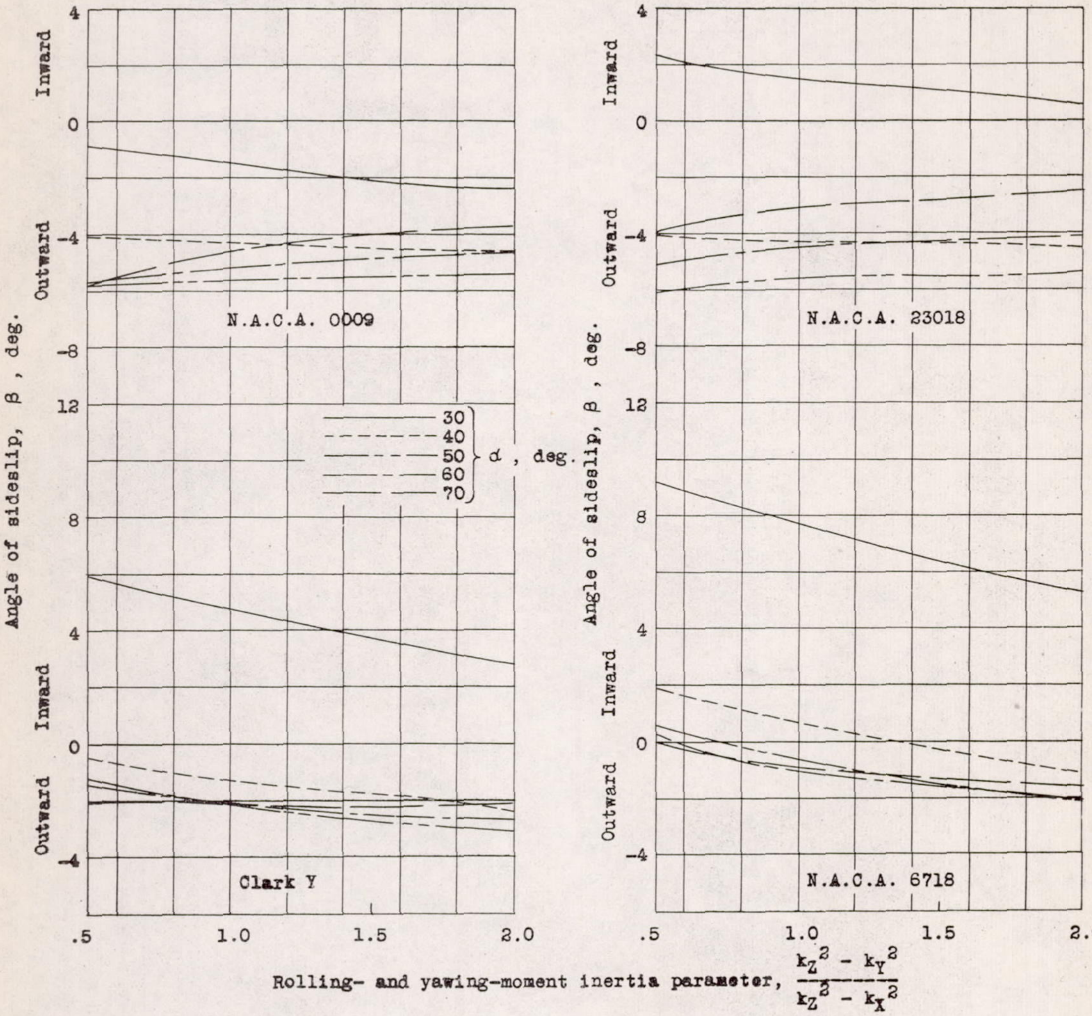

Figure 13. - Effect of rolling-moment and yawing-moment inertia parameter upon sideslip necessary for equilibrium in a spin.

$\mu = 5$ $\quad C_L = C_X$ $\quad C_m = -0.0020\ (\alpha - 20^\circ)$ $\quad \frac{b^2}{k_Z^2 - k_X^2} = 80$

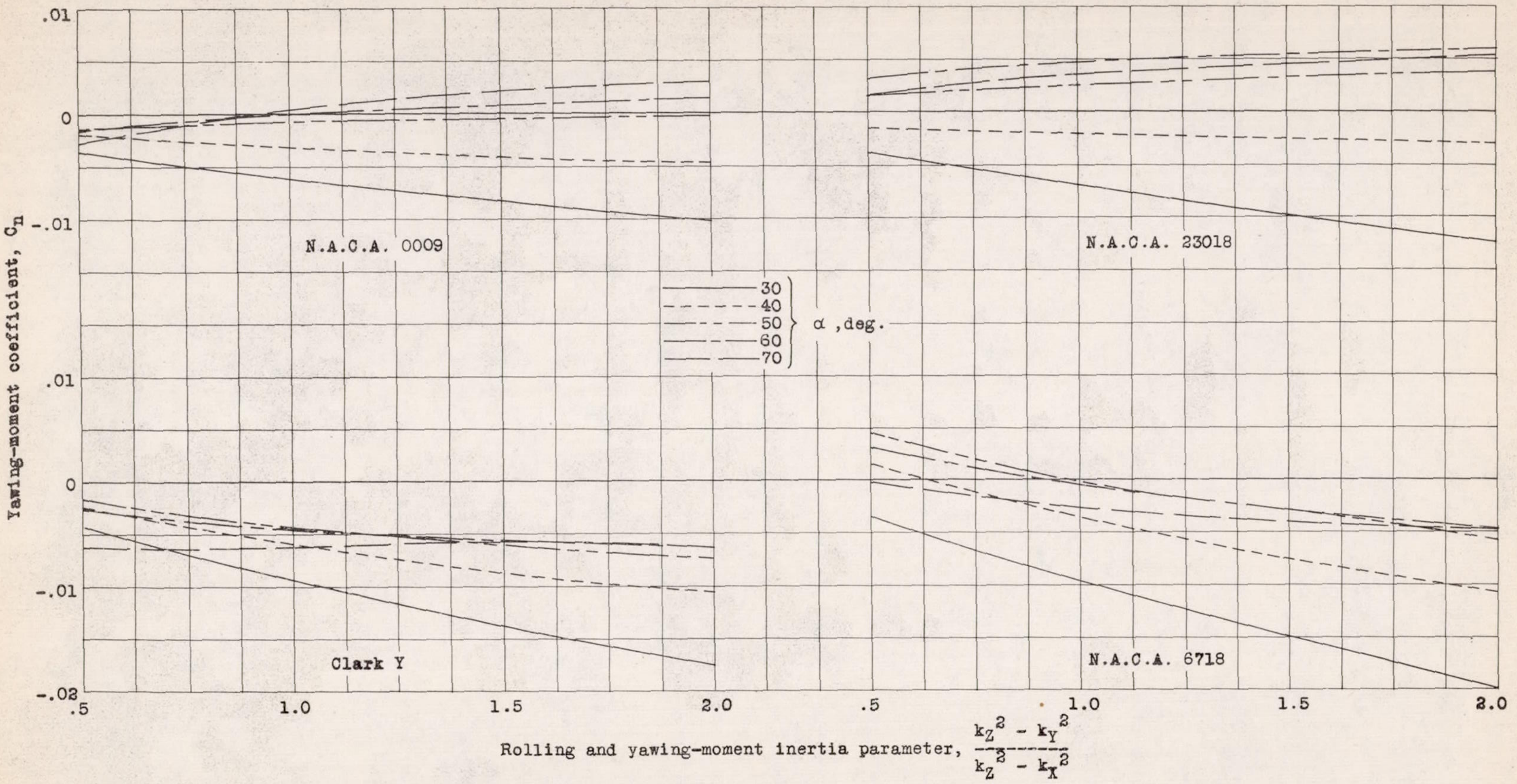

Figure 14. - Effect of rolling- and yawing-moment inertia parameter upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin. $\mu = 5$ $C_m = -0.0020 (\alpha - 20^\circ)$ $C_L = C_{X''}$ $\frac{b^2}{k_Z{}^2 - k_X{}^2} = 80$

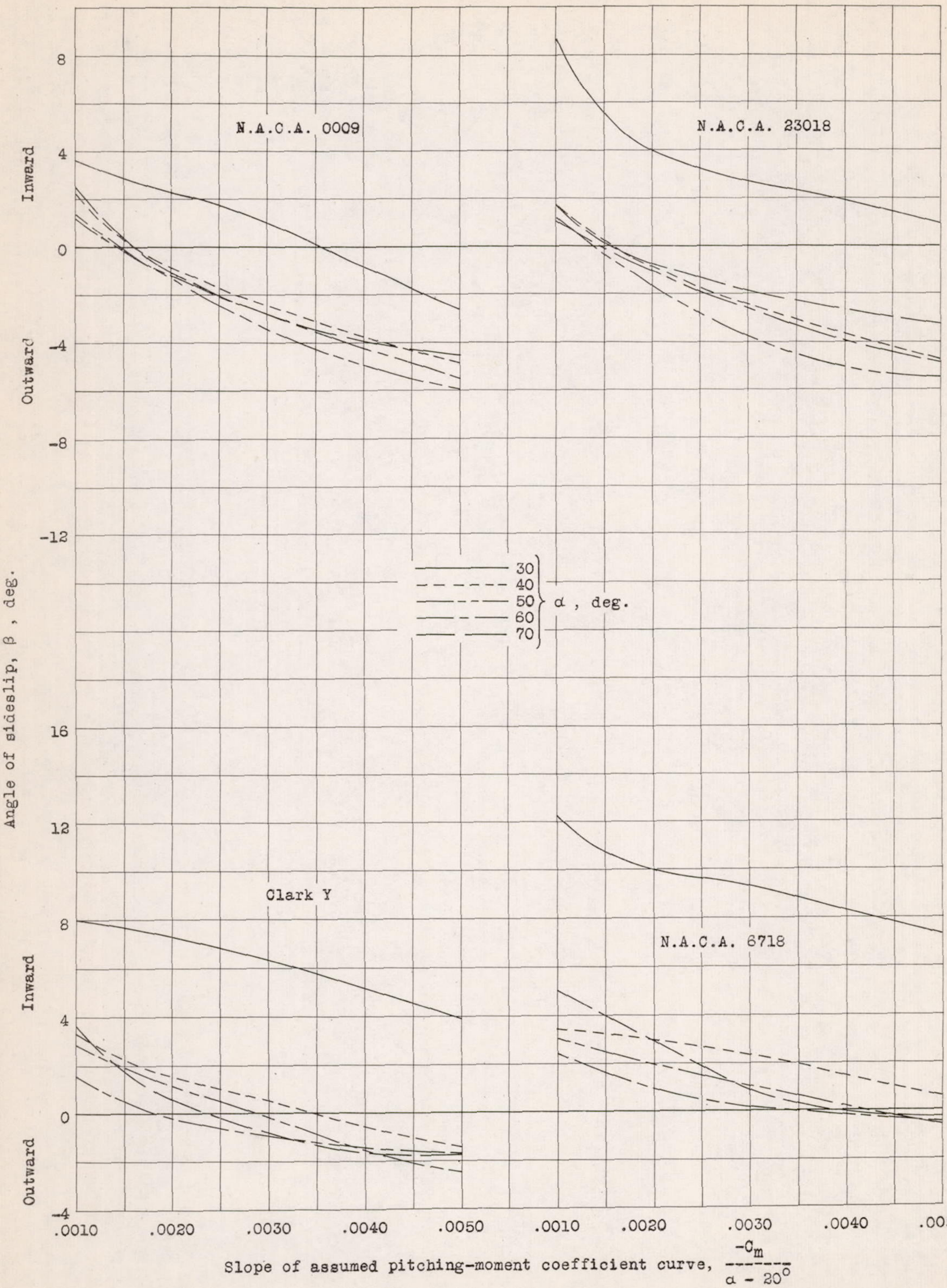

Figure 15. - Effect of pitching-moment coefficient upon sideslip necessary for equilibrium in a spin.

$\mu = 7.5$ $C_L = C_{X''}$ $\frac{k_Z{}^2 - k_Y{}^2}{k_Z{}^2 - k_X{}^2} = 0.5$ $\frac{b^2}{k_Z{}^2 - k_X{}^2} = 60$

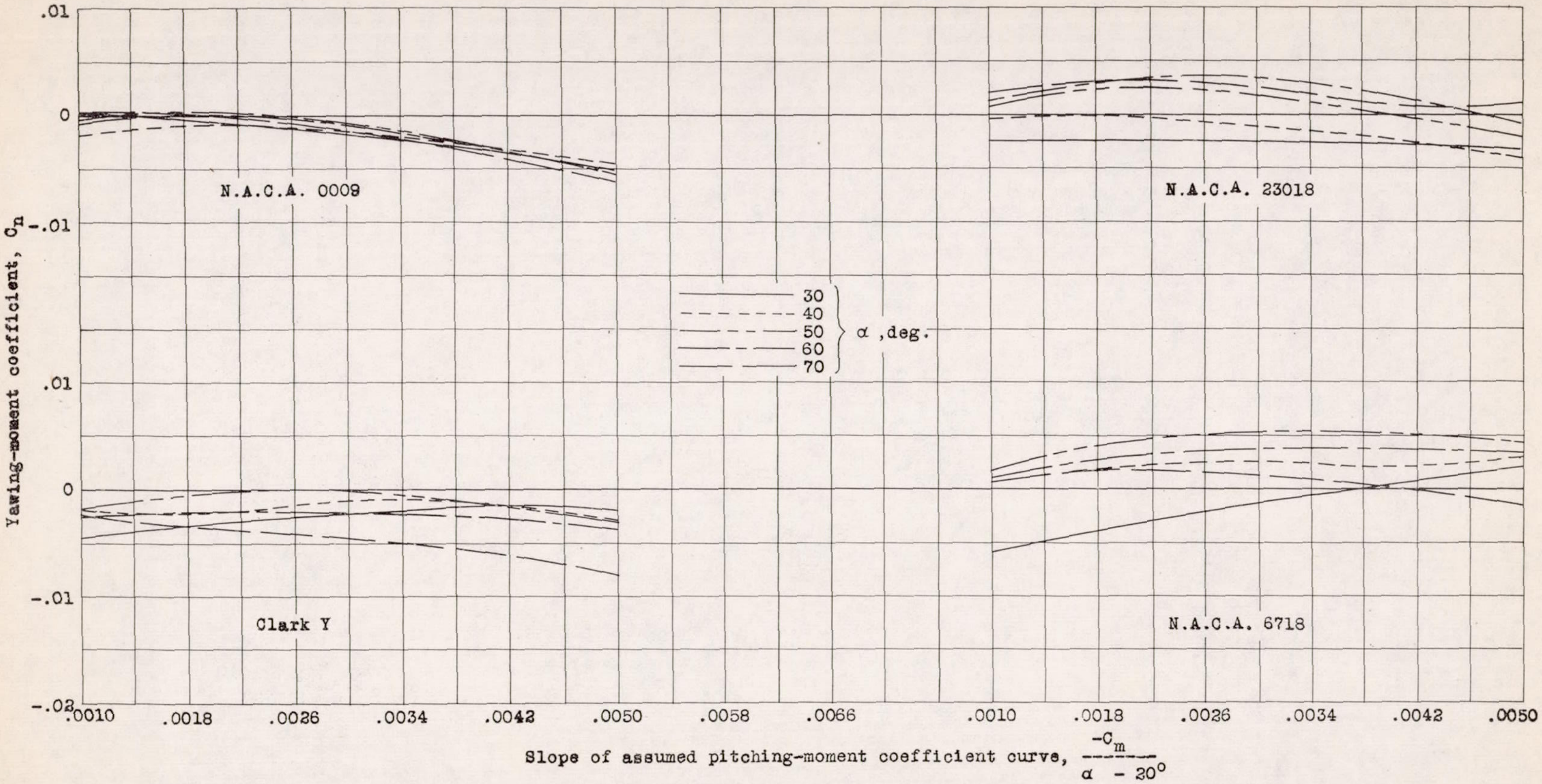

Figure 16. - Effect of pitching-moment coefficient upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin. $\mu = 7.5$ $C_L = C_{X''}$ $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 0.5$ $\frac{b^2}{k_Z^2 - k_X^2} = 60$

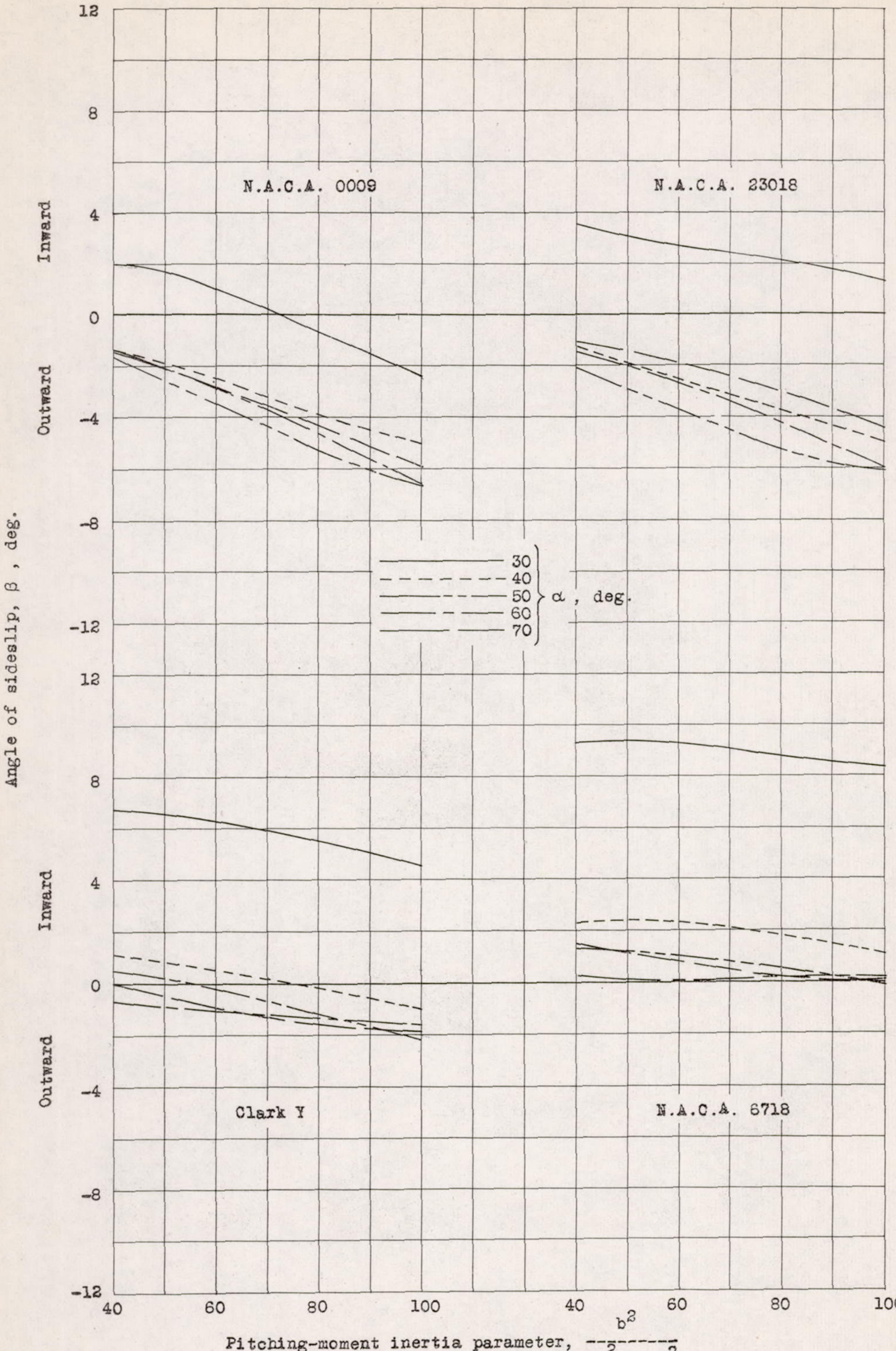

Figure 17. - Effect of pitching-moment inertia parameter upon sideslip necessary for equilibrium in a spin.

$\mu = 7.5$ $C_L = C_{X''}$ $C_m = -0.0030\ (\alpha - 20^o)$ $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 0.5$

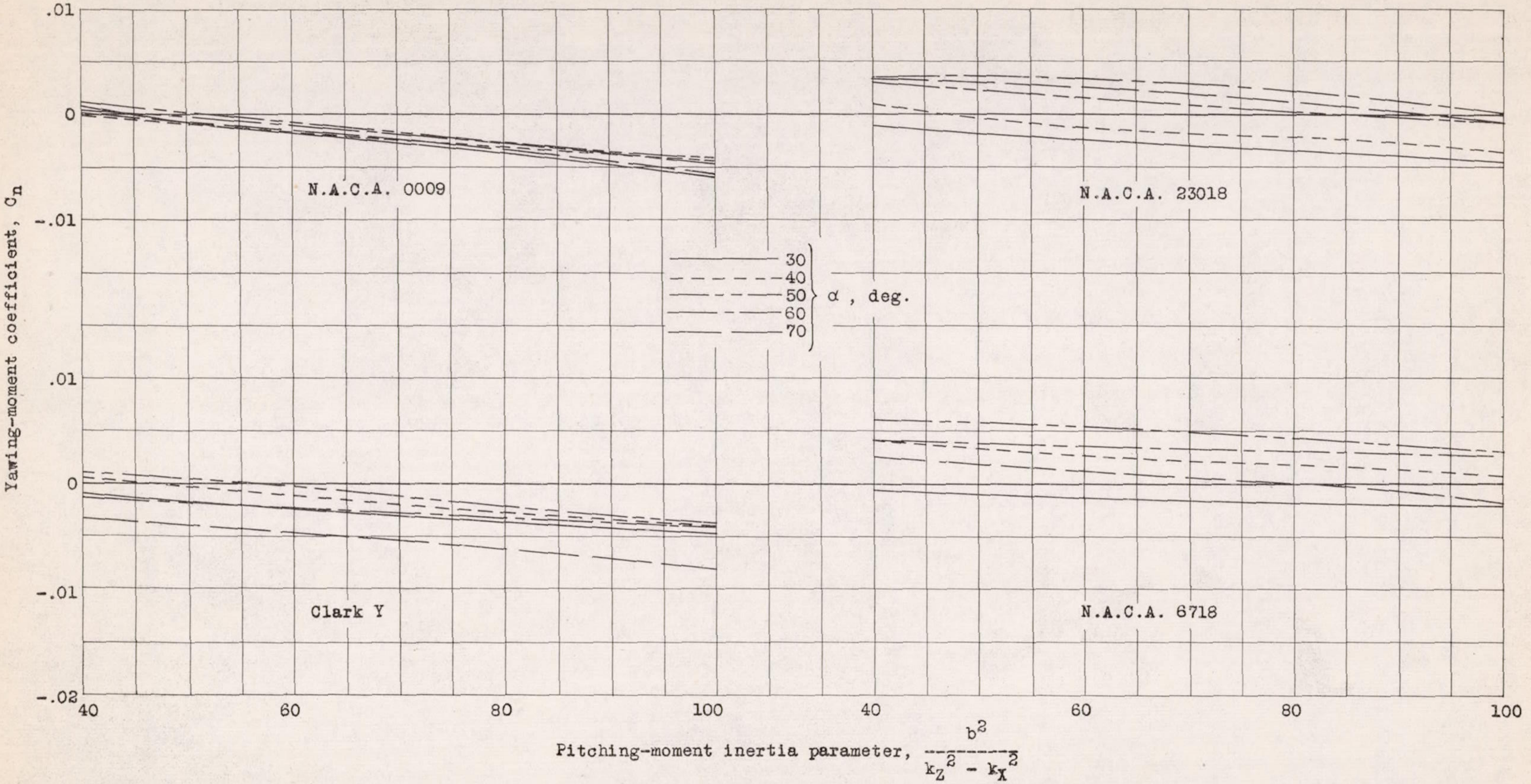

Figure 18. - Effect of pitching-moment inertia parameter upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin. $\mu = 7.5$ $C_L = C_X$" $C_m = -0.0030\ (\alpha - 20^\circ)$ $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 0.5$

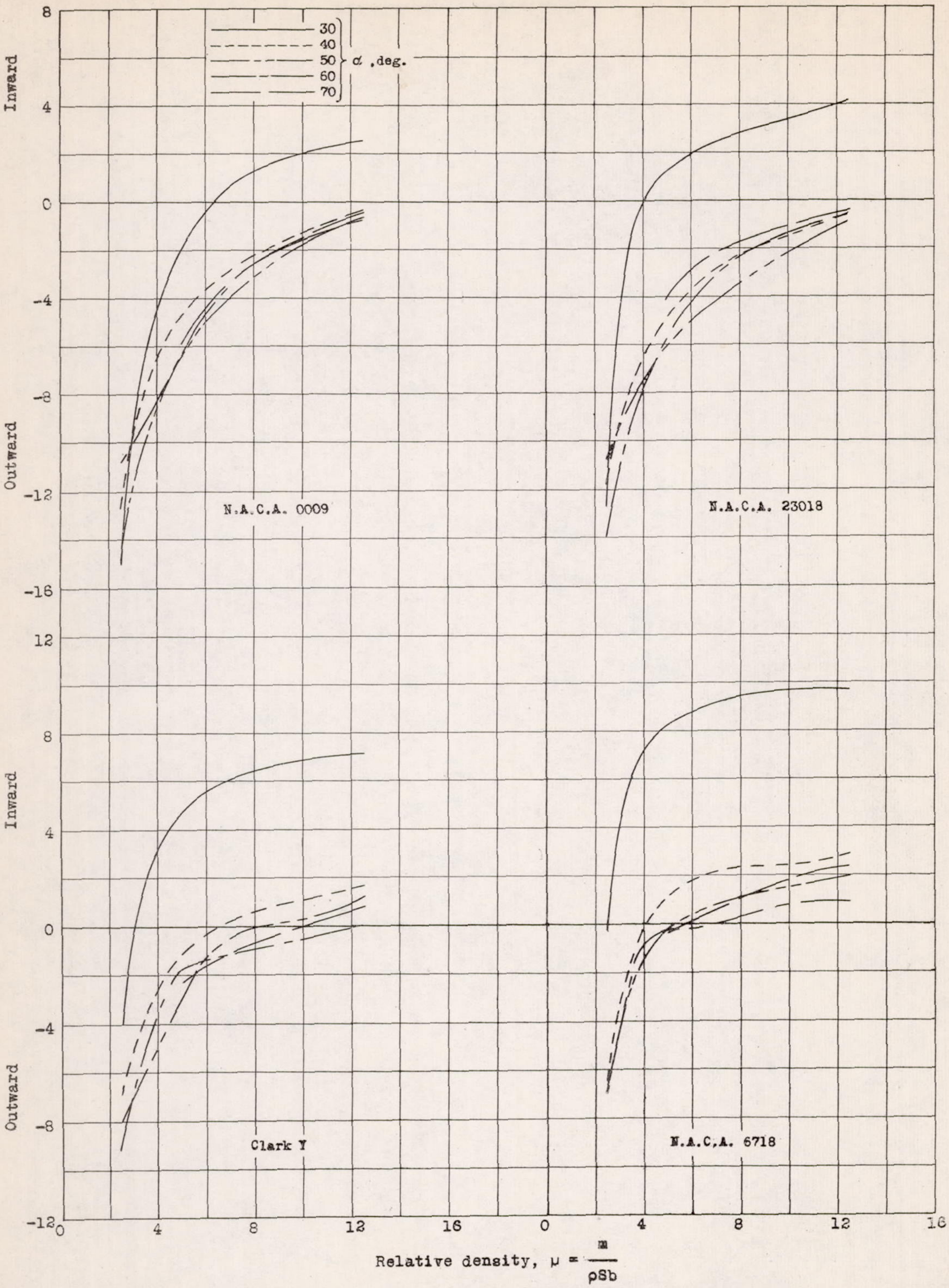

Figure 19. - Effect of relative density of airplane upon sideslip necessary for equilibrium in a spin.

$C_m = -0.0030\ (\alpha - 20^\circ)$ $\quad C_L = C_X$ $\quad \frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 0.5$ $\quad \frac{b^2}{k_Z^2 - k_X^2} = 60$

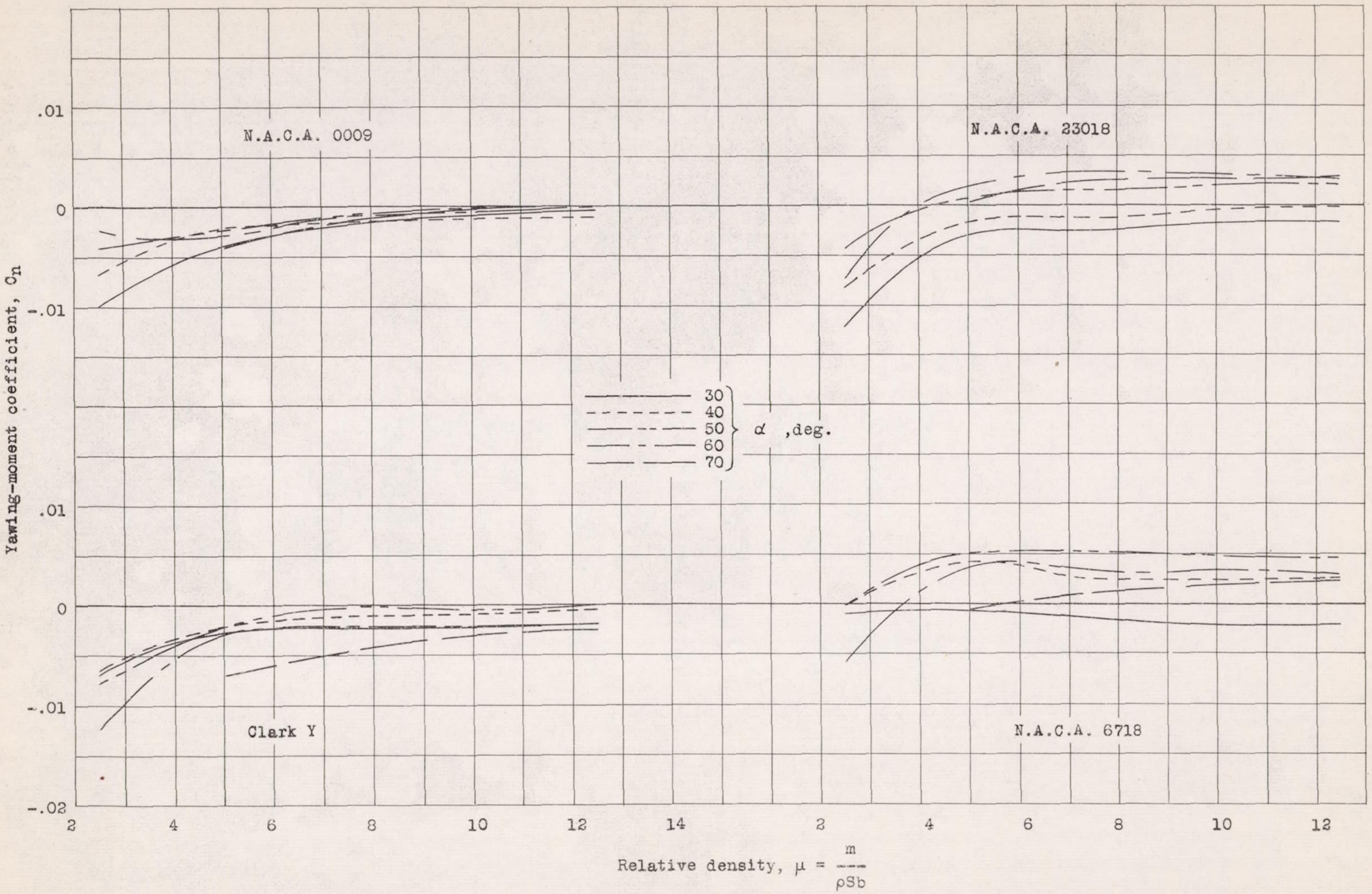

Figure 20.- Effect of relative density of airplane upon yawing- moment coefficient that must be supplied by parts of the airplane other than the wing for equilibrium in a spin. $C_L = C_{X''}$ $C_m = -0.0030\ (\alpha - 20^\circ)$. $\frac{k_Z{}^2 - k_Y{}^2}{k_Z{}^2 - k_X{}^2} = 0.5$ $\frac{b^2}{k_Z{}^2 - k_X{}^2} = 60$

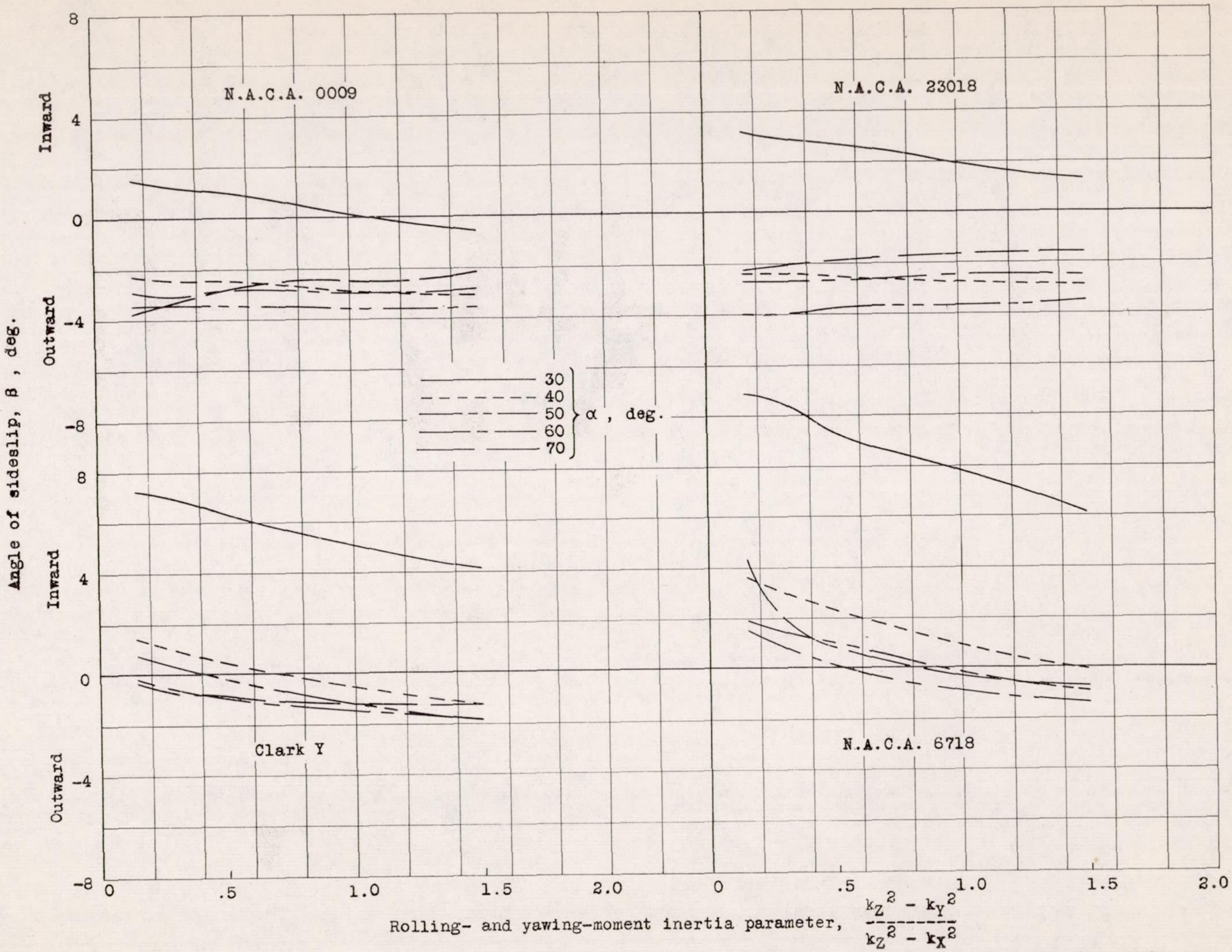

Figure 21. - Effect of rolling- and yawing-moment inertia parameter upon sideslip necessary for equilibrium in a spin.

$\mu = 7.5$ $C_L = C_{X''}$ $C_m = -0.0030\ (\alpha - 20^\circ)$ $\frac{b^2}{k_Z^2 - k_X^2} = 60$

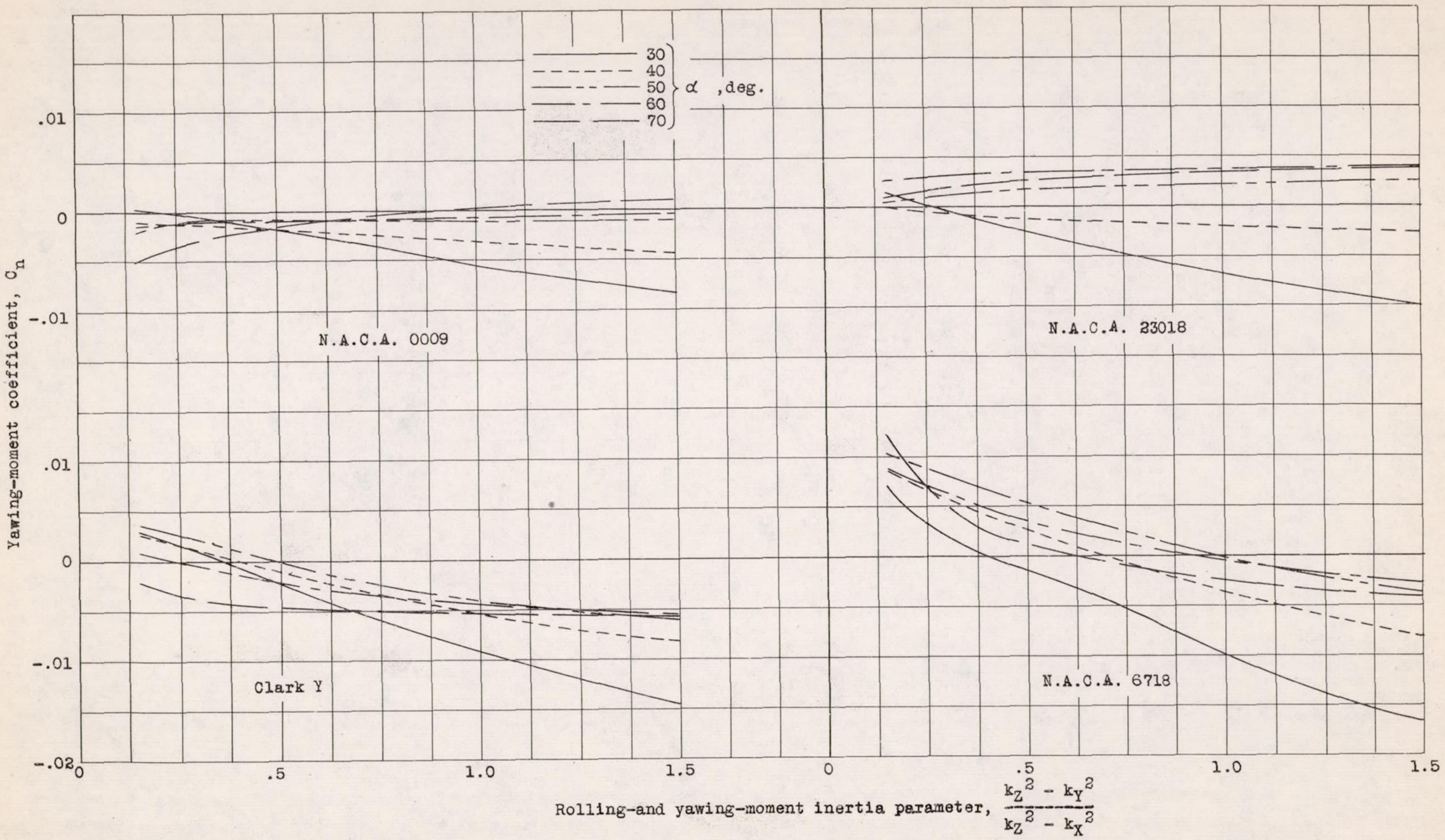

Figure 22. - Effect of rolling-and yawing-moment inertia parameter upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin. $\mu = 7.5$ $C_m = -0.0030\ (\alpha - 20^{\circ})$ $C_L = C_{X''}$ $\frac{b^2}{k_Z^2 - k_X^2} = 60$