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TECHNICAL NOTES

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 814

EFFECT OF SOME PRESENT-DAY AIRPLANE DESIGN TRENDS ON REQUIREMENTS FOR LATERAL STABILITY

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SUMMARY

Computations were made to determine the effect of some airplane design trends on the fin area and the dihedral angle required for lateral stability. The specific factors studied were wing loading, moments of inertia in roll and yaw, wing chord, and tail length. The airplane for which the computations were made was chosen to be representative of pursuit-type airplanes, but the conclusions drawn are applicable to any type of airplane characterized by the parameters and the data employed.

The results of the computations are presented in the form of diagrams of variations of fin area with dihedral angle for neutral stability. The results indicate that increasing the values of wing loading and moments of inertia makes the attainment of lateral stability increasingly difficult and even doubtful in some cases for airplanes with flaps deflected. The fin area required for lateral stability is more dependent on fuselage size than on wing area.

INTRODUCTION

In present airplane design, the trend is to use high-lift devices, high power, and weight distributed along the wings. Since the result of this trend is to increase the wing loading, the radii of gyration in roll and yaw, and the fuselage size with respect to the wing area, the amount of fin area and dihedral angle required for lateral stability will undoubtedly be affected.

The amount of fin area and dihedral angle required for the lateral stability of an airplane may be computed

from the equations of motion for small oscillations, provided the aerodynamic characteristics of the airplane are known. Although few aerodynamic data are available for some of the parameters involved, useful information can be obtained by assuming aerodynamic characteristics and studying the resulting trends. Some investigations of this nature have been reported in references 1 and 2. The data given in both references are limited in scope, and the direct effect of changes in the fin area and the dihedral angle is difficult to visualize. Also, the data given in reference 2 do not include the conditions for high wing loadings and large radii of gyration.

The present report covers the results of an investigation planned to show the general effect of changes in wing loading, radii of gyration, wing chord, and tail length on the amounts of fin area and dihedral angle required for lateral stability. The changes in wing loading and radii of gyration are intended to represent the present trends in pursuit-type airplanes. The results, however, apply to any type of airplane characterized by the parameters and the aerodynamic data used.

Computations were made for neutral spiral and oscillatory divergence, using representative values of the stability derivatives for the airplane as well as for changes in fin area, dihedral angle, and aspect ratio. The results of the computations are given as diagrams of dihedral angle against fin area for neutral spiral and oscillatory divergence.

The airplane parameters used in this investigation were the wing loading, radii of gyration, wing chord, and tail length. The variations in wing loading and radii of gyration were planned to represent present-day trends in pursuit-type airplanes. One parameter was varied at a time while the others were kept at a mean value. In some cases a change in one parameter is either directly or indirectly the result of a change in other parameters. The following table gives, in each group, all changes caused by the change in a particular parameter.

	W/S_w (lb/sq ft)	W (lb)	μ	k_X/b	k_Z/b	l_t/b	c (ft)	A
Mean value	30	6,000	9.70	0.125	0.175	0.4	5.0	8
Wing loading	15	3,000	4.85	0.125	0.175	0.4	5.0	8
	60	12,000	19.40	.125	.175	.4	5.0	8
Radii of gyration	30	6,000	9.70	0.250	0.270	0.4	5.0	8
	30	6,000	9.70	.083	.154	.4	5.0	8
Tail length	30	6,000	9.70	0.125	0.175	0.1	5.0	8
	30	6,000	9.70	.125	.175	.3	5.0	8
	30	6,000	9.70	.125	.175	.5	5.0	8
Wing chord	15	6,000	4.85	0.125	0.175	0.4	10.0	4
	60	6,000	19.40	.125	.175	.4	2.5	16

where

W/S_w wing loading

W weight of airplane

S_w wing area

μ ratio of airplane density to air density computed for wing span of 40 feet and standard air density at sea level $(m/\rho S_w b)$

k_X/b and k_Z/b ratio of radius of gyration about respective airplane axes to wing span (These values have been spotted on a plot (fig. 1) of values of k_X/b against k_Z/b from available design data of pursuit-type airplanes.)

l_t/b ratio of distance from center of gravity of airplane to center of pressure of fin to wing span

c average wing chord

A aspect ratio of wing

m airplane mass

ρ air density

S_f fin area (total vertical tail area)

b wing span

STABILITY DERIVATIVES

Because few data are available for some of the aerodynamic characteristics required, particularly for the interference between component parts of the airplane, two basic assumptions are necessary for computing the lateral-stability derivatives. These basic assumptions are:

1. The stability derivatives for each component part of the airplane are added to give the value for the complete airplane, that is, the interference effects are assumed to be zero. In some cases the interference effects are large and depend upon the arrangement of wing, fuselage, and tail-surface combinations. (See references 3 and 4 for some of the interference effects caused by wing location on the variation of the derivatives that depend upon sideslip.)

2. Neither the propeller nor the slipstream has any effect on the stability derivatives. These effects may be large for some of the derivatives.

The stability derivatives used and their variations with the airplane parameters were computed by the following relations, which are intended to include the effects of the parameters and particularly the changes in fin area and dihedral angle. The stability derivatives are the instantaneous rates of change of the aerodynamic coefficients with attitude or angular velocity when the attitude angle or the angular velocity is zero. For convenience the derivatives are written in the form C_{l_β}, C_{l_p}, C_{l_r},

etc. instead of $dC_l/d\beta$, $\frac{dC_l}{d\left(\frac{pb}{2V}\right)}$, $\frac{dC_l}{d\left(\frac{rb}{2V}\right)}$, etc. where C_l represents the force or moment coefficient and p and r represent the angular rates of rotation, rolling and yawing, respectively.

The relations for the derivatives are:

$$C_{Y_\beta} = C_{Y_{\beta(\text{fuselage})}} + C_{Y_{\beta(\text{wing})}} + C_{Y_{\beta(\text{fin})}}$$

$$= -0.020\,A + 0 - 3.48\,\frac{S_f}{S_w} \qquad (1)$$

where

C_Y force coefficient along Y axis of airplane

β angle of sideslip, radians, positive when the right wing is into the wind

and the subscripts indicate the contribution of the corresponding part of the airplane to C_{Y_β}. (Most wind-tunnel data have been given as $dC_Y'/d\psi'$ where C_Y' is the force along the Y wind axis and ψ' is the angle of yaw, degrees ($57.3\,\beta = -\psi'$); therefore C_{Y_β} $= -\frac{dC_Y'}{d\psi'}\,57.3 - C_D$.)

The constant 0.020 in $C_{Y_{\beta(\text{fuselage})}} = -0.020A$ was computed from data given in reference 3. The fact that, in this investigation, the fuselage size remains constant and the wing area varies inversely as the aspect ratio necessitates the use of this expression.

Although $C_{Y_{\beta(\text{wing})}}$ varies with dihedral (references 3, 5, 6, and 7), this derivative has not been included because it is counteracted, at least partly, by the derivative of the side force due to rolling C_{Y_p}.

The rate of change of normal force on the fin with angle of sideslip, $C_{Y_{\beta(fin)}}$, is equal to $-\left(3.48\ \frac{S_f}{S_w}\right)$ in terms of the wing area. The term S_f/S_w is the ratio of fin area to wing area (a variable for this investigation) and 3.48 is the rate of change of normal force coefficient on the fin with sideslip angle β for an effective aspect ratio of 3. The value obtained with the fin used on the model in reference 3 is about 3.48.

$$C_{l_\beta} = C_{l_{\beta(wing)}} + C_{l_{\beta(fin)}} = K_1\Gamma - \left(\frac{\bar{x}}{b} - \frac{l_t}{b}\sin i\right)\left(3.48\ \frac{S_f}{S_w}\right) \quad (2)$$

where

C_l rolling-moment coefficient

$C_{l_{\beta(wing)}} = K_1\Gamma$

K_1 varies with aspect ratio; it is equal to -0.0175 for $A = 16$, -0.0141 for $A = 8$, and -0.0114 for $A = 4$. These values of K_1 were obtained from data given in reference 6 for a wing of a 2:1 taper ratio.

Γ effective dihedral angle of wing, degrees. Effective dihedral angle is used throughout this report as a fictitious angle that would give the wing the value of C_{l_β}. Wing plan form and elevation, as well as large interference effects, contribute various amounts of C_{l_β}. (See references 3, 5, and 7.)

The contribution of the fin to the value of C_{l_β} is

$$\left(\frac{\bar{x}}{b} - \frac{l_t}{b}\sin i\right)\left(3.48\ \frac{S_f}{S_w}\right)$$

where $\left(\frac{\bar{x}}{b} - \frac{l_t}{b}\sin i\right)$ is the assumed ratio of the distance

of the center of pressure on the fin above the horizontal plane to the wing span. The value of $\bar{x}$ was assumed to be the distance from the fuselage center line to the center of pressure of the fin. Upon the assumption that the distance from the top of the fuselage to the center of pressure of the fin varied with the square root of the fin area and with the use of data from reference 3, the relation $\frac{\bar{x}}{b} = 0.025 + 0.23\sqrt{S_f/S_w}$ was obtained and used for all variations of the fin area. The value of i, the angle of fuselage center line to the horizontal, was zero for all cases when the lift coefficient C_L was equal to 0.2. For other values of C_L, the value of i depended upon both C_L and A. The value of i for $C_L = 2.8$ was the same as for $C_L = 1.4$. The increase in C_L was obtained by merely deflecting the flaps with no change in the angle of attack. The term $3.48\ S_f/S_w$ was used in the expression for $C_{Y_{\beta(fin)}}$.

$$C_{n_\beta} = C_{n_{\beta(wing)}} + C_{n_{\beta(fuselage)}} + C_{n_{\beta(fin)}}$$

$$= K_2 - 0.009A + \frac{l_t}{b}\left(3.48\ \frac{S_f}{S_w}\right) \qquad (3)$$

where C_n is the yawing-moment coefficient.

A value of K_2 of 0.009 was used for plain wings and of 0.030 when the flaps were deflected. Values for K_2 may vary considerably with wing forms. (See references 5 and 7.)

The value $C_{n_\beta} = -0.009A$ was obtained from data in reference 3 and may be expressed as a function of A because the fuselage and the wing span are constant with change of wing area.

The contribution of the fin to the value of C_{n_β} is the computed variation of the normal force with β on the fin $C_{Y_{\beta(fin)}}$ times the nondimensional lever arm l_t/b.

$$C_{l_p} = C_{l_{p(wing)}} = K_3 \tag{4}$$

where the values of K_3 (from reference 6) are -0.6 for $A = 16$, -0.5 for $A = 8$, and -0.4 for $A = 4$. The values of $C_{l_{p(fin)}}$ and $C_{l_{p(fuselage)}}$ are probably very small as compared with $C_{l_{p(wing)}}$ and therefore have not been used.

$$C_{n_p} = C_{n_{p(wing)}} + C_{n_{p(fin)}} = K_4 C_L + 2 \frac{l_t}{b}\left(\frac{\bar{x}}{b} - \frac{l_t}{b} \sin i\right)\left(3.48 \frac{S_f}{S_w}\right) \tag{5}$$

where K_4 varies with the aspect ratio and is (from reference 6) - 0.089 for $A = 16$, - 0.065 for $A = 8$, and - 0.040 for $A = 4$.

The expression for $C_{n_{p(fin)}}$, that is,

$2 \frac{l_t}{b} \left(\frac{\bar{x}}{b} \frac{l_t}{b} \sin i\right) \left(3.48 \frac{S_f}{S_w}\right)$, is the rate of change of the yawing moment with the rolling velocity, produced by assuming that the normal force on the fin is proportional to the angle induced at the center of pressure of the fin by the rolling velocity; actually the induced angle is a variable along the fin.

$$C_{l_r} = C_{l_{r(wing)}} + C_{l_{r(fin)}} = 0.25 C_L + 2 \frac{l_t}{b}\left(\frac{\bar{x}}{b} - \frac{l_t}{b} \sin i\right)\left(3.48 \frac{S_f}{S_w}\right) \tag{6}$$

where the value of 0.25 was obtained from data given in reference 6, and the term for $C_{l_{r(fin)}}$ is of the same

form as for $C_{n_{p(fin)}}$; but, in this case, the angle is induced by the yawing velocity and the value of $C_{l_{r(fin)}}$ therefore is probably more nearly representative of the actual value of the derivative than the value given for $C_{n_{p(fin)}}$.

$$C_{n_r} = C_{n_{r(fuselage)}} + C_{n_{r(wing)}} + C_{n_{r(fin)}}$$

$$= -0.00125A + \left(K_5 C_L{}^2 - K_6\right) - 2\left(\frac{l_t}{b}\right)^2\left(3.48\ \frac{S_f}{S_w}\right) \quad (7)$$

where K_5 (from reference 6) is -0.0113 for $A = 16$, -0.02105 for $A = 8$, and -0.03838 for $A = 4$. The constant K_6 depends on the profile-drag coefficient of the wing (reference 6) and is assumed to be 0.003 for the plain wing and 0.030 for the wing with flaps deflected. The value $-0.00125A$ was assumed for the fuselage and varies inversely with the wing area.

The expression for $C_{n_{r(fin)}}$, that is, $2\left(\frac{l_t}{b}\right)^2\left(3.48\ \frac{S_f}{S_w}\right)$, is the nondimensional form of the rate of change of yawing moment due to the fin with yawing velocity.

THEORETICAL EQUATIONS

The equations used to compute the boundaries of neutral spiral and oscillatory stability were developed from the theory of small oscillations, as given in reference 2, simplified for the level-flight condition. The equations are of the form:

$$A\lambda^4 + B\lambda^3 + C\lambda^2 + D\lambda + E = 0$$

where

$$A = 1$$

$$B = -Y_v - n_r - l_p$$

$$C = l_p n_r - l_r n_p + Y_v \left(l_p + n_r\right) + \mu n_v$$

$$D = Y_v \left(l_r n_p - l_p n_r\right) + \mu l_v \left(n_p - \frac{C_L}{2}\right) - \mu n_v l_p$$

$$E = \frac{C_L}{2} \left(\mu l_v n_r - \mu n_v l_r\right)$$

and

$$Y_v = \frac{1}{2} C_{Y_\beta}$$

$$\mu l_v = \frac{\frac{1}{2} \mu C_{l_\beta}}{\left(k_X/b\right)^2}$$

$$\mu n_v = \frac{\frac{1}{2} \mu C_{n_\beta}}{\left(k_Z/b\right)^2}$$

$$l_p = \frac{\frac{1}{4} C_{l_p}}{\left(k_X/b\right)^2}$$

$$n_p = \frac{\frac{1}{4} C_{n_p}}{\left(k_Z/b\right)^2}$$

$$l_r = \frac{\frac{1}{4} C_{l_r}}{\left(k_X/b\right)^2}$$

$$n_r = \frac{\frac{1}{4} C_{n_r}}{\left(k_Z/b\right)^2}$$

The necessary conditions for stability are that the values of A, B, C, D, E, and Routh's discriminant R shall all be positive. Since the stability derivatives and the airplane parameters used for this investigation are such that the values of A, B, and C are positive and the values of D are positive when R is positive for any practical value of effective dihedral angle, stability depends on the values of E and R. When the value of E becomes negative, the airplane becomes spirally unstable. The lateral oscillations increase in amplitude when the value of R becomes negative. The limits of the stable region are therefore defined by the values of Γ and S_f/S_w when

$$E = 0$$

and

$$R = BCD - D^2 - B^2E = 0$$

RESULTS AND DISCUSSION

The results are presented in the form of diagrams (figs. 2 to 5) showing the variations in the computed boundaries of spiral and oscillatory stability with S_f/S_w (the ratio of the effective fin area to the wing area) and with Γ, the effective dihedral angle. Figure 6 is a replot of data given in figure 4(c) with the ordinate given as $\left(S_f/S_w\right)(8/A)$. Figure 7 is a replot of the data given in figure 5(c) with the ordinate given as $\left(S_f/S_w\right)\left(l_t/b\right)$, (effective tail volume). The value of S_f/S_w where C_{n_β} is zero, the value of C_{n_β} required for neutral weathercock stability, is indicated in figures 2 to 7.

The results, in general, indicate that the value of Γ required for spiral stability increases with C_L and S_f/S_w. The value of S_f/S_w required for oscillatory stability decreases with Γ for several degrees and then increases with continued increase in Γ. For some cases the diagrams do not cover a sufficient range of Γ to

show the increase in S_f/S_w. The rate of change of S_f/S_w with Γ required for oscillatory stability increases with C_L and with some of the airplane parameters.

The stability boundaries for $C_L = 2.8$ may appear to be inconsistent with the other lift coefficients. The differences, however, result from the changes in C_L, C_{n_β}, and C_{n_r} produced by the flaps and from the assumption that the angle of attack is the same as that with no flaps, that is, $C_L = 1.4$.

Effect of wing loading on stability boundaries.- The value of μ, relative density of the airplane to air, has no effect on the boundary of neutral spiral stability. (See fig. 2 and the expression $E = C_L/2\,(\mu l_v n_r - \mu n_v l_t) = 0$.) The change in the boundaries with a variation in C_L results from the variation of the derivatives with C_L and not from the presence of μ in the expression for E.

For oscillatory stability, increasing μ increases the value of S_f/S_w required for a given value of Γ and decreases the value of Γ for a given value of S_f/S_w. This effect is small for small values of Γ but becomes large when Γ is increased more than a certain amount and is particularly important for high values of C_L and μ. (See figs. 2(c) and 2(d).)

The resulting effect of increasing μ is to increase the difficulty of obtaining lateral stability, especially with large values of C_L and with flaps deflected.

Effect of radii of gyration in roll and yaw on stability boundaries.- The effects of k_X/b and k_Z/b (fig. 3) are practically the same as the effects of μ. The values of k_X/b and k_Z/b used have a greater effect on the oscillatory boundaries than have the values of μ. There is no effect of k_X/b and k_Z/b on the spiral boundaries.

The data given in figures 3(c) and 3(d) indicate that, for large values of C_L, it is impossible to satisfy the two conditions both with and without flaps. The

value of C_{l_β} represented in this report as an effective Γ usually increases with C_L if the wing has sweepback or certain tip shapes (references 5 and 7). Although this variation in effective Γ with C_L might be useful in obtaining better stability characteristics, it may cause serious oscillatory instability at the high values of C_L with large values of μ or of k_X/b and k_Z/b.

Effect of changing wing chord on stability boundaries.- Changing the wing chord without changing the span, the fuselage, or the total weight of an airplane changes A, μ, and S_w. Because the fin area, as used in this report, is represented as a ratio of the wing area, S_f/S_w, the actual fin area is inversely proportional to A. These factors must be considered when the effects of changes in wing chord, given in figure 4, are compared. Figure 6, which is included to show the effect of these factors, is a replot of figure 4(c) based on the actual value of the fin area.

Decreasing the wing chord (increasing A and μ), in general, decreases the value of Γ necessary for spiral stability (fig. 4). This effect must be due to A and to the ratio of wing area to fuselage size because, as previously explained, μ has no effect on these changes. When the actual values of S_f are considered (fig. 6), the variation of the area with Γ is irregular for large values of Γ and the wing with $A = 8$ requires the larger values of Γ for spiral stability.

The value of S_f/S_w required for oscillatory and weathercock stability increases with a decrease in the wing chord (figs. 4 and 6). This effect is greater than the change in the actual fin area (fig. 6) and in μ, as shown in figure 2, and is primarily due to the decrease in the values of C_{n_p} and C_{n_r} of the wing as compared with the fuselage size. It appears from these results that changes in fin area should be based on fuselage size and fuselage characteristics rather than on wing area. This conclusion is in agreement with the results from flight tests reported in reference 8.

Effect of tail length on stability boundaries.- Increasing the ratio of the tail length to the wing span l_t/b decreases, almost directly, the value of S_f/S_w required for stability (figs. 5 and 7). This decrease results from the fact that the computed values of C_{n_β} vary directly with $\left(S_f/S_w\right)\left(l_t/b\right)$, but the closeness to a direct variation is also due to the fact that the computed decrease in C_{Y_β} is partly counter-balanced by the computed increase in C_{n_r}.

As the tail length is increased, the value of Γ, and the possibility of obtaining stability, is increased. These effects are indicated by figure 7, where the effects of Γ are small but may be important for airplanes having large values of k_X/b, k_Z/b, and μ.

GENERAL COMMENTS

The aerodynamic characteristics of airplanes vary considerably; so choice of their values, their rates of change with various parameters, and their interference effects may greatly modify the magnitude of these results. The results of this investigation indicate general effects of changes in certain airplane parameters and do not indicate the stability characteristics of a particular airplane.

The general results of this investigation may be summed up by pointing out the factors that make the attainment of lateral stability more critical and difficult:

1. Large values of μ (high wing loading and high altitude) and large values of radii of gyration (weight distributed along the wings)

2. Small wing chords

3. Short tail lengths. (The magnitude of this effect is comparatively small.)

4. High lift coefficients and particularly the necessity of obtaining stability both with and without flaps

Other factors, such as the type and the degree of stability desired and the choice of aerodynamic characteristics used, should have a large effect on the interpretation of these results.

Although the type and the degree of stability that affect the control and the riding qualities of the airplane are outside the scope of this report, some of these factors should be considered.

It is normally considered that, for satisfactory stability characteristics, the oscillatory motion should be highly damped, that is, the airplane should have a large amount of oscillatory stability. This stability can be obtained by proportioning the effective fin area and the effective dihedral angle so that their values will be positive, that is, when plotted on a diagram for a particular airplane, the value will be in a stable area and well above and to the left of the zero oscillatory stability boundary of the airplane. The distance from the boundary, although an indication, is not a quantitative measure of the amount of damping. An airplane should be spirally stable, particularly for flying conditions of poor visibility. For good riding qualities in rough air, however, the amount of spiral stability should be small; spiral instability is generally considered to be preferable to a poorly damped oscillatory motion. The value of C_{n_β} should be positive, weathercock stability; the amount, however, is dependent upon the spiral and the oscillatory stability requirements.

CONCLUSIONS

From the analysis for lateral stability, assumed data for changes in certain airplane parameters being used, the following conclusions may be drawn:

1. Increasing the values of the relative density of the airplane to the air, that is, increasing the wing loading or the altitude, makes the attainment of lateral stability increasingly difficult and critical.

2. Increasing the radii of gyration more than certain amounts makes the attainment of lateral stability very difficult and critical.

3. Increasing the aspect ratio of the wing by decreasing the chord makes the attainment of lateral stability more difficult.

4. Increasing the tail length, with the fin area inversely proportional to the tail length (constant tail volume), makes the attainment of stability slightly less difficult.

5. The fin area required for lateral stability depends more upon the fuselage size than upon the wing area and, for a given fuselage, the required fin area should be increased slightly as the wing area is decreased.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., May 14, 1941.

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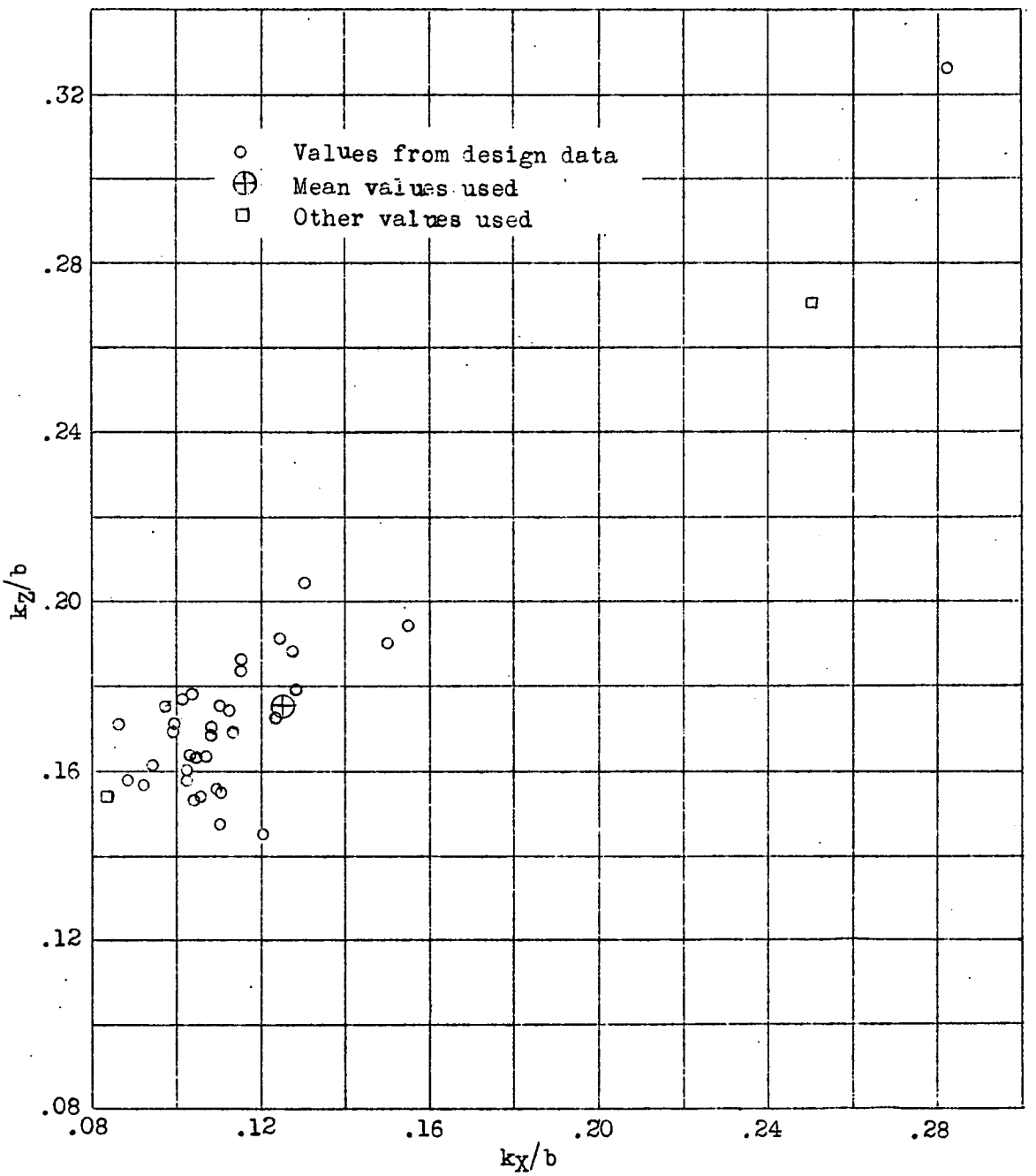

Figure 1.- Variation of k_Z/b with k_X/b from design data available for pursuit-type airplanes and those used in the computations.

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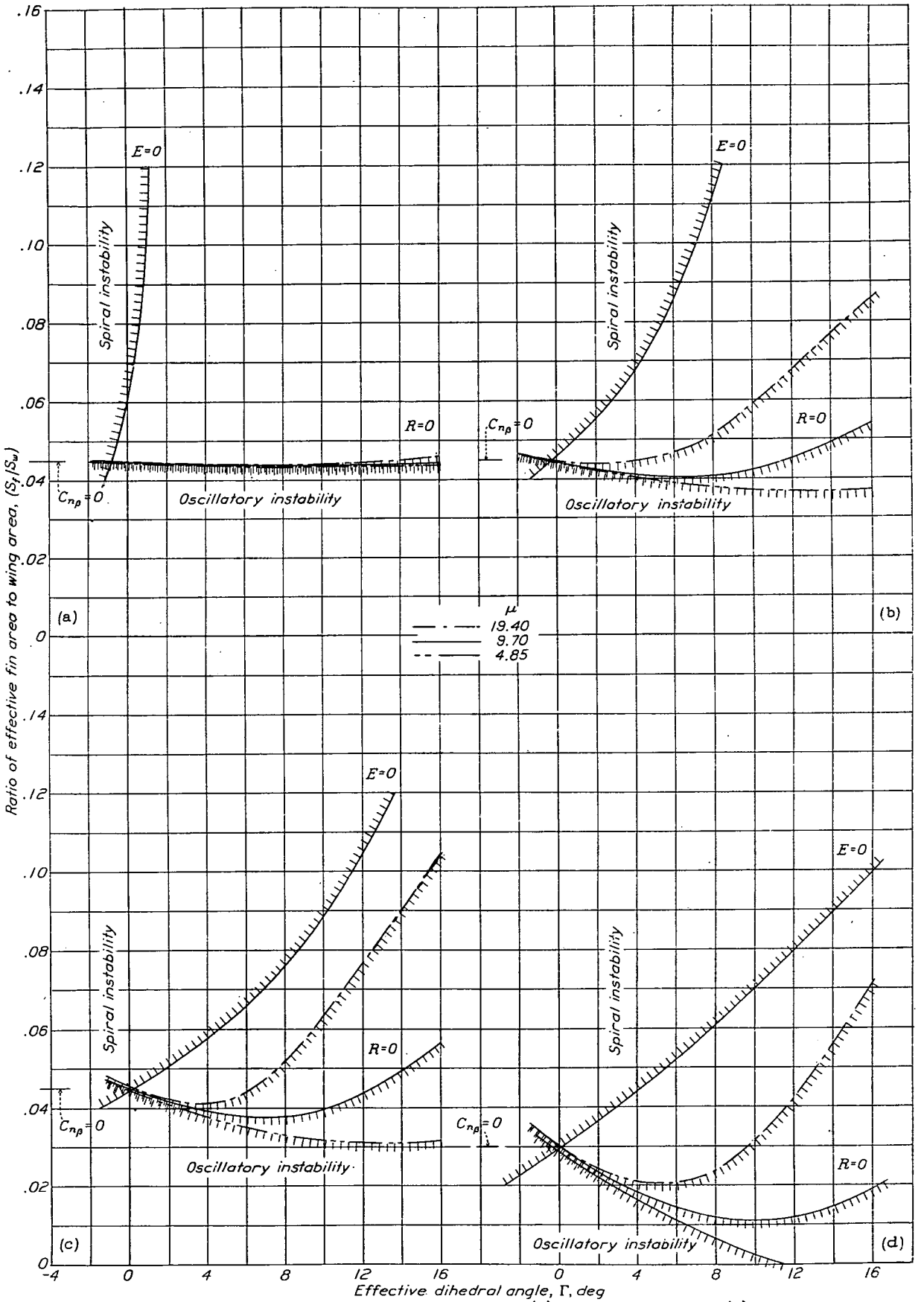

(a) C_L , 0.2 (b) C_L , 0.8 (c) C_L , 1.4 (d) C_L , 2.6

Figure 2. Effect of μ on boundaries of spiral and oscillatory stability. A , 8 ; l_t/b , 0.4 ; k_X/b , 0.125; k_Z/b , 0.175 .

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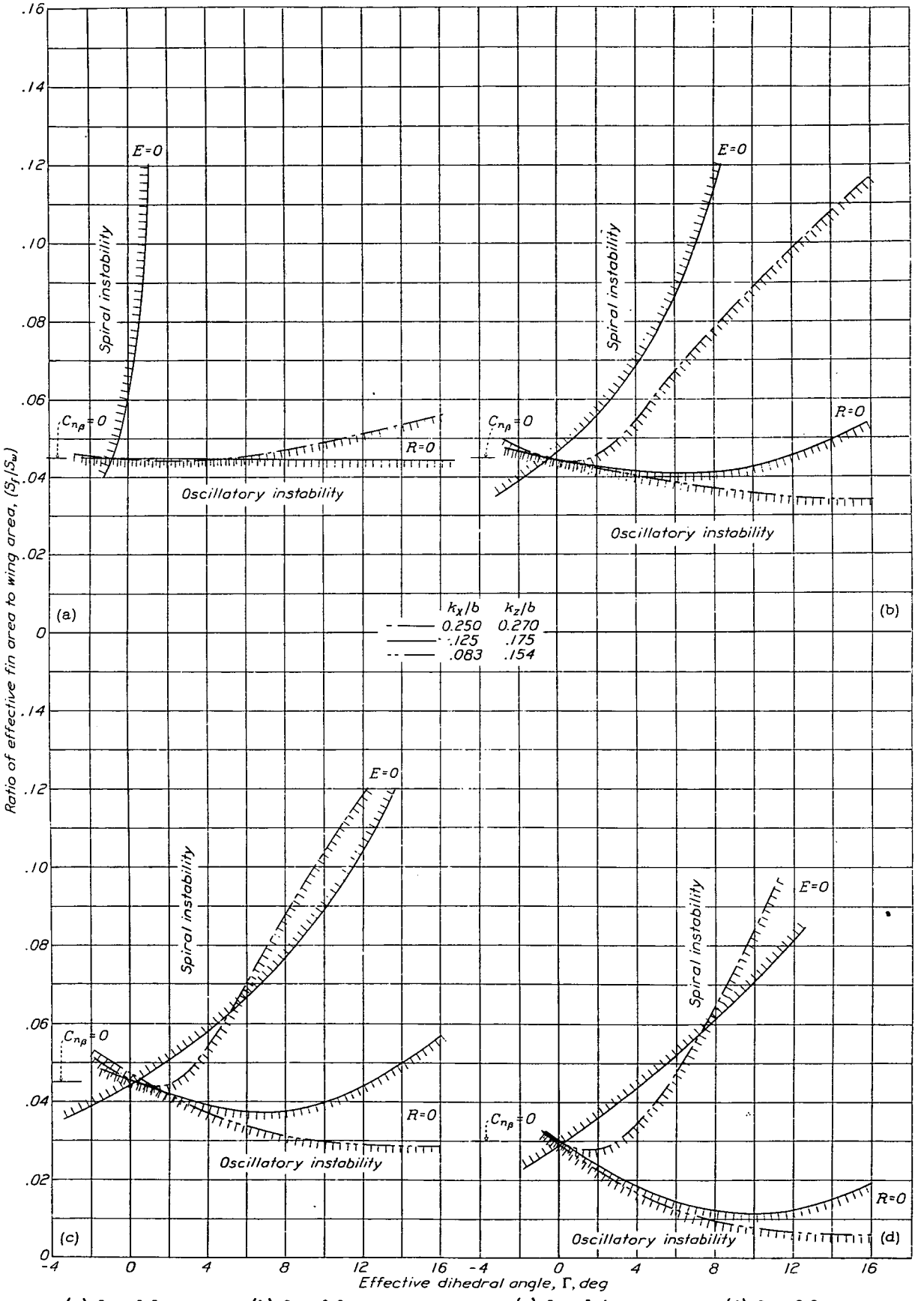

(a) C_L , 0.2 (b) C_L , 0.8 (c) C_L , 1.4 (d) C_L , 2.8

Figure 3.- Effect of radii of gyration on boundaries of spiral and oscillatory stability. A , 8 ; l_t/b , 0.4 ; μ , 9.70 .

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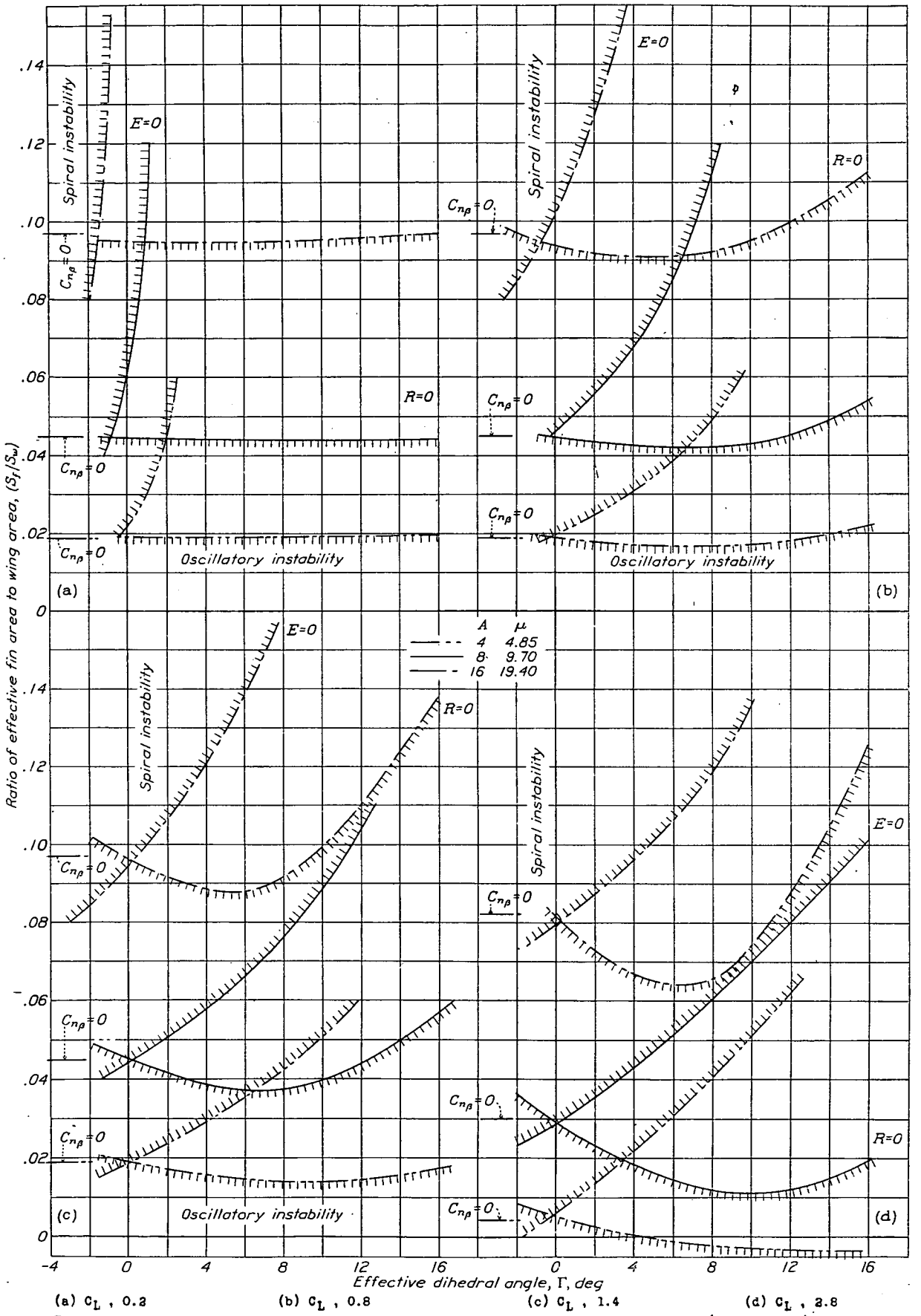

(a) C_L , 0.2 (b) C_L , 0.8 (c) C_L , 1.4 (d) C_L , 2.8

Figure 4. Effect of wing chord on boundaries of spiral and oscillatory stability. l_t/b , 0.4; k_X/b , 0.125; k_Z , 0.175.

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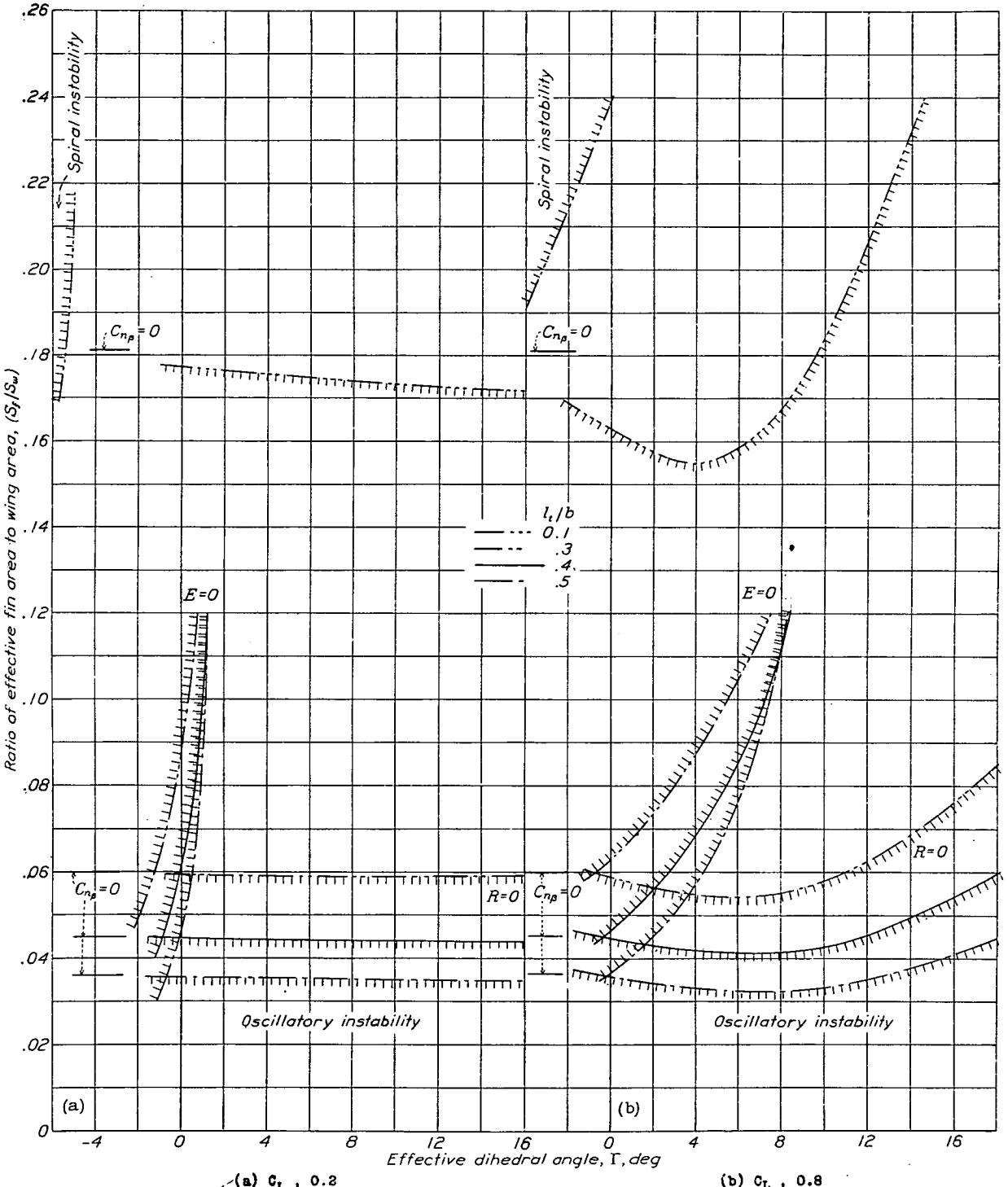

(a) C_L , 0.2 (b) C_L , 0.8

Figure 5a to d.- Effect of tail length on boundaries of spiral and oscillatory stability. A , 8 ; k_X/b , 0.125; k_Z/b , 0.175; μ , 9.70 .

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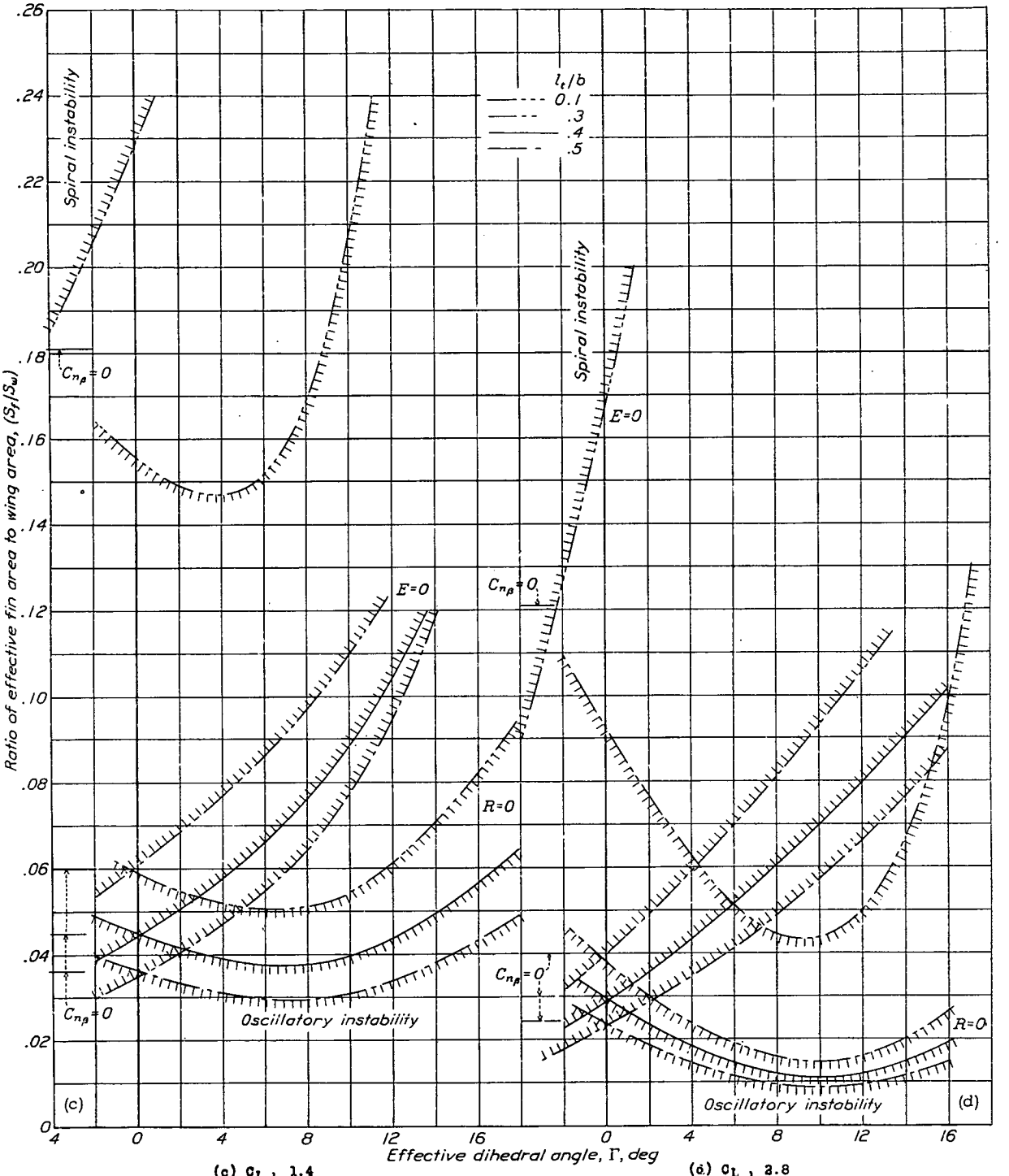

(c) C_L , 1.4 (d) C_L , 2.8

Figure 5. Concluded.

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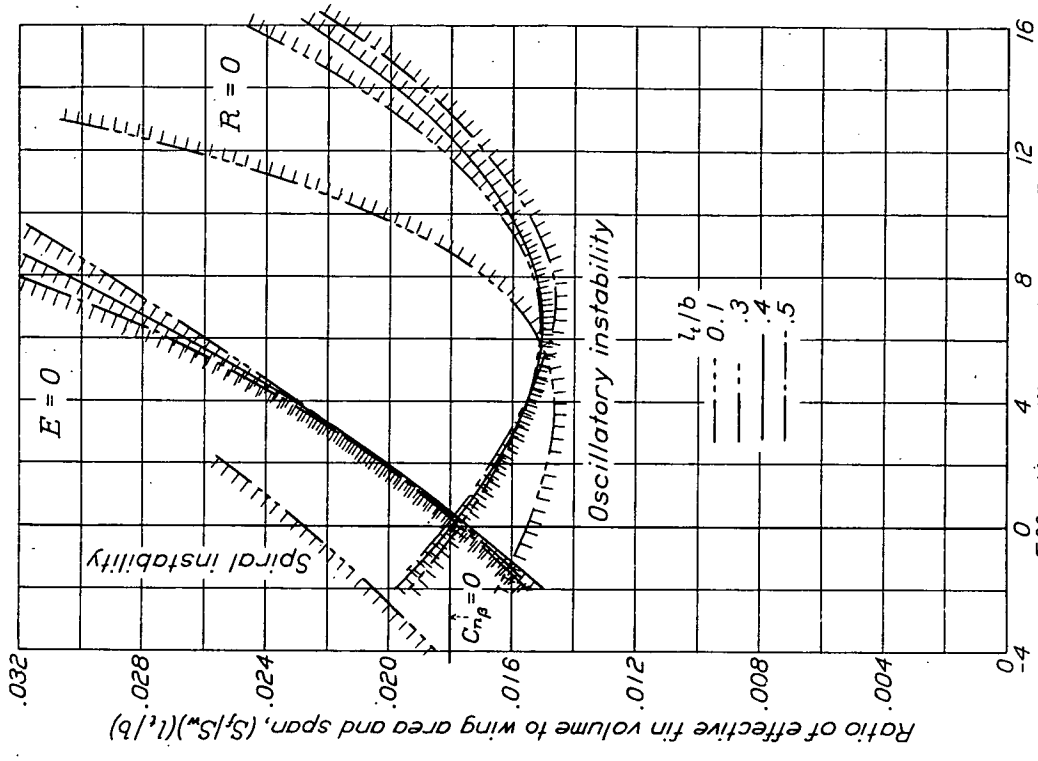

Figure 7. Effect of tail length on boundaries of spiral and oscillatory stability. A , 8 ; k_X/b,0.125; k_Z/b , 0.175 ; μ , 9.70 ; C_L , 1.4.

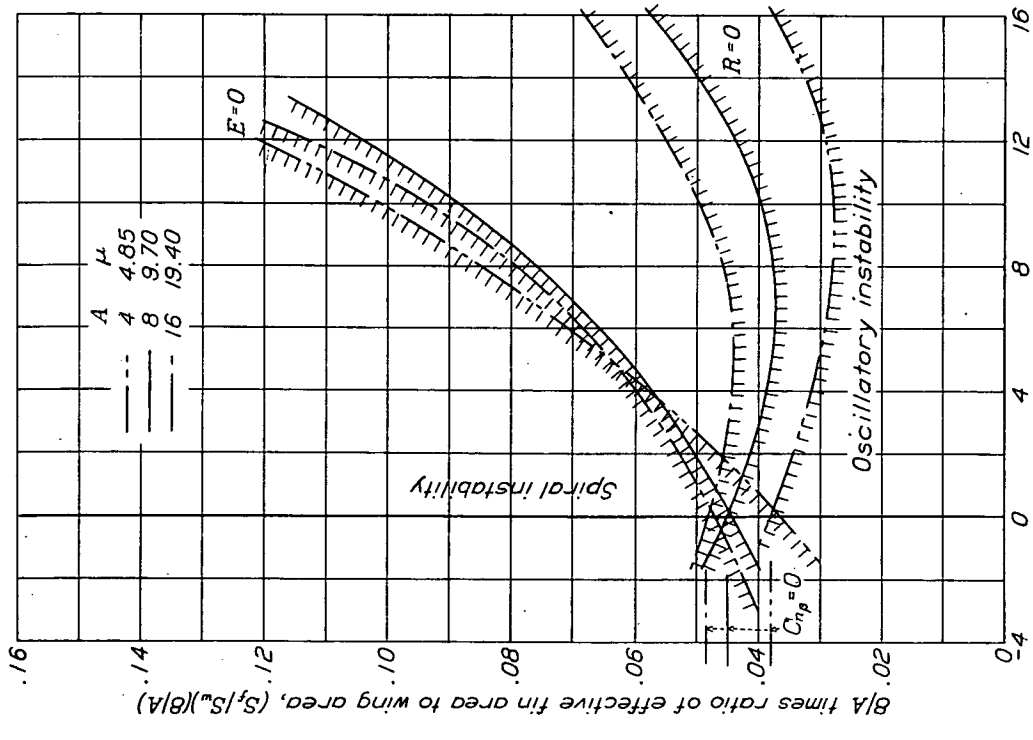

Figure 8. Effect of wing chord on boundaries of spiral and oscillatory stability. l_t/b, 0.4; k_X/b , 0.125 ; k_Z/b , 0.175 ; C_L , 1.4.