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RESEARCH MEMORANDUM

COMPRESSIBILITY EFFECTS ON A HOVERING HELICOPTER ROTOR HAVING AN NACA 0018 ROOT AIRFOIL TAPERING TO AN NACA 0012 TIP AIRFOIL

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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COMPRESSIBILITY EFFECTS ON A HOVERING HELICOPTER ROTOR HAVING AN NACA 0018 ROOT AIRFOIL TAPERING TO AN NACA 0012 TIP AIRFOIL

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SUMMARY

An investigation has been conducted on the Langley helicopter test tower to determine experimentally the effects of compressibility on the hovering performance and the blade pitching moments of a helicopter rotor having an NACA 0012 tip airfoil, an NACA 0018 root airfoil, and -7.5° of twist. Data are presented for blade-tip Mach numbers from 0.33 to 0.88. The results show that the primary effect of compressibility on the performance was a rapid increase in the profile-drag torque coefficient once a critical combination of tip speed and tip angle of attack was exceeded.

The airfoil section lift-curve slope required for prediction of the rotor thrust varied from a value of 5.73 at a tip Mach number M_t of 0.33 to a value of 6.33 at $M_t = 0.74$. The variation in lift-curve slope with Mach number had a characteristic shape similar to that shown by two-dimensional data.

Although the tip of the blade encountered compressibility losses there was very little variation in the pitching moments as the rotor thrust was increased well beyond the onset of compressibility losses.

At calculated tip angles of attack from 0° to about 2.5°, the rotor-blade drag-divergence Mach number measured was about 0.01 to 0.02 higher that that indicated by the two-dimensional data where Mach number for drag divergence was defined as $\frac{\Delta c_d}{\Delta M_t} = 0.1$ (where c_d is the section drag coefficient) and about 0.1 higher where Mach number for drag divergence was defined as the start of drag rise.

The efficiency of the rotor, as defined by its figure of merit, decreased with increases in tip Mach number. At $M_t = 0.33$ the maximum figure of merit was about 75 percent whereas at $M_t = 0.83$ the maximum figure of merit dropped to 50 percent.

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INTRODUCTION

One method of meeting the requirements of increased helicopter speeds and higher disk loadings is by increasing the rotor-blade tip speed. Design studies of helicopters with rotor tip speeds in the high subsonic Mach number range and with high blade loadings have emphasized the need for experimental rotor performance and blade pitching-moment data.

This investigation, which is a continuation of a general research program (refs. 1 and 2) to determine the effects of compressibility on rotors having various NACA airfoil sections as the primary variable, extends the program to include tests of a rotor having an NACA 0012 tip airfoil section, an NACA 0018 root airfoil section, and -7.5° of twist. The Mach numbers for drag divergence have been compared with unpublished two-dimensional data for the NACA 0012 airfoil section.

The rotor blades were tested on the Langley helicopter test tower over a tip Mach number range from 0.33 to 0.88 (disk loading up to 5 pounds per square foot) with a corresponding blade tip Reynolds number range from 1.95×10^6 to 5.19×10^6. Inasmuch as the rotor blade leading edge may become rough in field service because of abrasion or the accumulation of foreign particles, such as bug spatters, the tests were repeated for three tip Mach numbers (0.47, 0.74, and 0.83) with leading-edge roughness.

The major portion of the data presented has been obtained with the blade in the factory production finish condition. In this condition the blades had a blunter leading-edge radius than the true airfoil. Repeat test runs were made at tip Mach numbers of 0.31, 0.71, and 0.80 with 4 feet of the outboard leading edge of the rotor blade built up essentially to the nominal airfoil contour (NACA 0012 at the tip and NACA 0014 at the 3/4 radius) for comparison with the data obtained with the blade in its original condition.

SYMBOLS

a	slope of curve of section lift coefficient as a function of section angle of attack, per radian (assumed to be 5.73 for incompressible calculations)
b	number of blades
c	blade-section chord at radius r, ft

c_e equivalent blade chord, $\dfrac{\int_0^R cr^2 dr}{\int_0^R r^2 dr}$, ft

r radial distance to a blade element, ft

R rotor-blade radius, ft

T rotor thrust, lb

Q rotor torque, lb-ft

M rotor-blade pitching moment, lb-ft

M_t rotor-blade tip Mach number

σ rotor solidity, $bc_e/\pi R$, 0.0260

μ coefficient of viscosity, slugs/ft-sec

ρ mass density of air, slugs/cu ft

N_{Re} Reynolds number at blade tip, $\dfrac{\rho \Omega R c_t}{\mu}$

Ω rotor angular velocity, radians/sec

α_r blade-section angle of attack, deg

θ blade-section pitch angle measured from line of zero lift, radians

φ inflow angle at blade element, $\theta - \alpha_r$, radians

$c_{d,o}$ airfoil-section profile-drag coefficient

$\bar{c}_l$ mean rotor-blade lift coefficient, $\dfrac{6C_T}{\sigma}$

C_T rotor-thrust coefficient, $\dfrac{T}{\pi R^2 \rho (\Omega R)^2}$

C_Q rotor-torque coefficient, $\dfrac{Q}{\pi R^2 \rho (\Omega R)^2 R}$

C_M rotor-blade pitching-moment coefficient, $\dfrac{M}{R\frac{\rho}{2}(\Omega R)^2 c_e^{\,2}}$

Q_o rotor profile-drag torque, lb-ft

$C_{Q,o}$ rotor profile-drag torque coefficient, $\dfrac{Q_o}{\pi R^2 \rho (\Omega R)^2 R}$

Subscripts:

o profile

t at blade tip

Definition:

Figure of merit minimum possible power required to produce a given thrust with an ideal rotor (minimum induced power (constant inflow), zero profile drag power, and zero rotational and tip losses) compared to the actual power required to produce that thrust,

$$\left(\text{Figure of merit} = 0.707\ \frac{C_T^{3/2}}{C_Q}\right)$$

APPARATUS AND TEST METHODS

Rotor Blades

The rotor used for this investigation was a fully articulated, 2-blade rotor with flapping hinge located on the center line of rotation and the drag hinge 12 inches outboard of the center line. The distance from the ground to the rotor hub center was 42 feet.

A sketch of the rotor blade with pertinent dimensions is shown in figure 1. The rotor blades used in the investigation were of wood core construction and incorporated a steel strap located near the leading edge to bring the center of gravity to the 24-percent chord.

The radius of the blade from the center line of rotation was 26.4 feet and the rotor solidity was 0.0260. The blade had a root chord of 20.64 inches and tapered linearly to 10.13 inches at the tip. The blade airfoil section at the root was an NACA 0017.63 tapering to an NACA 0014 at the 3/4 radius and to an NACA 0012.35 at the blade tip. The rotor blade had 7.5° of nonlinear washout (about 0.205° per foot of twist from the root to the 3/4 radius and 0.389° per foot of twist from there to the tip).

The rotor blades, as received in the factory production finish condition deviated from the true airfoil sections - the main deviation being approximately a 50-percent greater leading-edge radius. However, the airfoil was smooth and free from flat spots over most of the blade span.

At the conclusion of the performance tests with factory production finish blades, the outboard 4 feet of the blade spar was built up to an airfoil contour more comparable to that specified in the blade design. The leading-edge radius was reduced to within 3 percent of that for the true airfoil. A portion of the test was then repeated to check the extent this refinement affected the rotor performance.

Since helicopter rotor blades may develop leading-edge roughness during normal service because of abrasion or the accumulation of foreign particles, a portion of the test program was repeated with leading-edge roughness (No. 60 carborundum particles which are approximately 0.011 inch in diameter) applied to the rotor blades. Roughness was added to the forward 8 percent of the blade chord (upper and lower surfaces) over the entire blade span with the particles covering 5 to 10 percent of the area. This roughness represents more than that usually due to normal service and is used only to demonstrate the effects of an extreme condition.

Test Methods and Accuracy

The tests were run by setting a given rotor-blade collective pitch angle and varying the rotor speed through a range of tip Mach numbers until either the blade-limiting stress was obtained or the tower drive-shaft limiting torque value had been reached. The blade pitch settings were increased until either one of these limits had been reached or until a further increase in blade pitch did not increase rotor thrust. At each pitch setting, data were recorded both from visual dial readings and by an oscillograph. Quantities measured were rotor thrust, rotor torque, blade pitch angle, blade pitching moment, rotor shaft rotational speed, blade drag angle, and blade flapping angle. The range of test conditions was chosen to exceed the blade tip Mach numbers corresponding to the force break in order to establish the rate of increase of the compressibility losses with tip angle of attack and Mach number.

The test blades were designed to rotate in a direction opposite that which is normal for the helicopter tower drive. Inasmuch as the tower drive could not be reversed, the rotor was installed in an inverted position (positive rotor thrust was in a downward direction).

The estimated accuracies of the basic quantities measured during the tests are as follows: rotor thrust, ±20 pounds; rotor torque, ±50 pound-feet, rotor rotational speed, ±1 revolution per minute; and all angular measurements, $\pm 0.2^{\circ}$. The overall accuracy of the plotted results is believed to be within ±3 percent. For example, at a rotor-blade mean lift coefficient $\bar{c}_l$ of 0.4 $\left(C_T = 0.001735\right)$ and an $M_t = 0.70$, the accuracy of the plotted data based on repeatability was about 2 percent for the thrust value of 5,428 pounds, or torque value of 6,786 pound-feet, and 0.3 percent for value of rotational speed of 280 rpm.

METHOD OF ANALYSIS

In order to show the onset and rate of growth of the compressibility drag rise and drag increase due to blade stall, that part of the power affected by these losses had to be isolated. An analysis of the problem indicates that the profile-drag power and, hence, the profile-drag torque coefficient, would be chiefly affected.

A convenient reference for the rate of growth of profile torque losses is the ratio of the profile-drag torque coefficient deduced from the test results to that calculated by using conventional strip analysis and incompressible drag coefficients. The resulting ratios $\frac{\left(C_{Q,o}\right)_{measured}}{\left(C_{Q,o}\right)_{calculated}}$ are plotted as a function of the blade-tip angle of attack. Values of $C_{Q,o}$ were calculated using the conventional airfoil drag polar

$$c_{d,o} = 0.0087 - 0.0216\alpha_r + 0.400\alpha_r^2$$

and a lift-curve slope of 5.73 per radian. This drag polar has been used in several analyses (refs. 3 to 6), and experience has shown that its use provides a good approximation of the performance at low tip speed of well-built blades below maximum lift. The measured rotor profile-drag torque coefficients were determined by subtracting the calculated induced torque coefficient from the measured torque coefficient. The data were plotted as a function of tip angle of attack and blade-tip Mach number. The rotor blade tip angles of attack were determined by plotting the rotor thrust coefficient as a function of measured rotor-tip pitch angle and subtracting, at the same thrust coefficient, an induced angle at the blade tip φ_t obtained from a strip analysis.

Pitching-moment coefficients deduced from the force measured in the rotor-blade pitch-control linkage are presented in this paper. The coefficients thus included the contribution of both aerodynamic and mass forces. In this respect, it is to be noted that the presence or absence of abrupt changes in the pitching-moment coefficient is of more significance than the actual values.

RESULTS AND DISCUSSION

The basic hovering performance and rotor-blade pitching moments are presented first. From these basic curves, an analysis of the effects of compressibility and stall are presented. Data on the effect of tip speed on the average rotor-blade lift-curve slope are also presented.

Hovering Performance

Smooth blades.- The hovering performance of the smooth blades (factory production finish) is shown in figure 2 as the variation of thrust coefficient and torque coefficient for a blade-tip Mach number range from 0.33 to 0.88. An incompressible rotor performance curve was calculated by using the method discussed in the section "Method of Analysis" and is also plotted for comparison with the experimental data. The calculated curve was obtained by using the conventional strip-analysis procedure and a 3-percent tip loss factor (outer 3 percent of the blade produces no lift but has profile drag). In general, the experimental curves at low speed $\left(M_t = 0.33 \text{ and } 0.42\right)$ show good agreement with the calculated curve up to rotor thrust coefficients of about 0.0034 $\left(\overline{c}_l = 0.78\right)$. The maximum blade mean lift coefficient $\overline{c}_l$ was calculated using the maximum low-speed rotor thrust coefficient measured for the factory production finish blades having NACA 0012.4 airfoil tip sections. This value of $\overline{c}_l$ was 0.98 at a value of C_T of 0.00425, as compared to maximum blade mean lift coefficient of 1.07 for the blades having NACA 23015 airfoil tip sections (ref. 1), and maximum blade mean lift coefficient of 1.17 for blades having NACA 63_2-015 airfoil tip sections (ref. 2). The maximum value of $\overline{c}_l$ may be slightly higher than the value of 0.98 for the 0012 tip section blades since the slope of the performance curve is still slightly positive at the highest pitch angle tested.

Increasing tip speed caused the experimental rotor performance curve to break away from the calculated curve at progressively lower values of thrust coefficient. The point at which this occurred indicates the critical value at which compressibility and stall losses begin.

At $M_t = 0.70$ and $\overline{c}_l$ of 0.6 $\left(C_T = 0.0026\right)$, these blades have about a 10-percent increase in power due to compressibility losses as shown in figure 2(b), as compared to the 4.4 percent measured at the same operating conditions for blades having NACA 63_2-015 airfoil sections (ref. 2).

At the highest tip speeds tested $\left(M_t = 0.83 \text{ and } 0.88\right)$ the experimental rotor-performance curves show an increase in zero lift drag of about 35 percent and 80 percent, respectively.

The repeat tests to determine the effect of modifying the outer 4 feet of the rotor blade to closer airfoil contour tolerances (NACA 0012 at tip) showed no difference (within the test accuracy) from that obtained with the blades in the factory production finish condition. This does not appear unreasonable in view of the original smooth and fair surface condition of the rotor blades. An indication of the relative smoothness of a rotor-blade surface is given by the profile-drag coefficient at zero rotor thrust. The present factory production finish blade, of which the skin surface is referred to as "smooth," had a zero-thrust drag coefficient of 0.0080. With leading-edge roughness, the drag coefficient at zero thrust increased to about 0.0105. If the blade had had an aerodynamically smooth surface, the drag coefficient would be expected to be about 0.0065, as indicated by the two-dimensional data (ref. 7).

Leading-edge roughness.- A comparison of the rotor performance with smooth blades and with leading-edge roughness applied is shown in figures 3(a), (b), and (c) $\left(M_t = 0.47, 0.74, 0.83\right)$. The addition of leading-edge roughness increased the zero-thrust profile-torque coefficient by about 30 percent. At the highest thrust coefficients shown, the increase in profile-torque coefficient is about twice that shown at zero lift.

Rotor efficiency.- The efficiency of the rotor blade at various Mach numbers expressed in terms of figure of merit, is shown in figure 4. The maximum low tip speed figure of merit of the blades was about 70 to 75 percent. As the tip Mach number was increased, the figure of merit of the rotor decreased. For example, at a tip Mach number of 0.83 the maximum figure of merit dropped to about 50 percent.

Effect of tip Mach number on blade lift-curve slope.- A variation of the airfoil-section lift-curve slope with M_t has been indicated by two-dimensional data. The rotor-blade airfoil mean lift-curve slope can be deduced from figure 5, which is a plot of rotor thrust coefficient against blade pitch angle at the 3/4 radius for a tip Mach number range of 0.33 to 0.88. The lift-curve-slope values, so deduced for various tip Mach numbers, have been plotted in figure 6. At the low tip Mach numbers, the airfoil-section lift-curve slope which predicts the rotor thrust coefficient was about 5.73, which agrees with the value normally used for low-speed calculations. At a tip Mach number of 0.74 a value of 6.33 for the airfoil-section lift-curve slope was required and at the highest tip Mach number tested $\left(M_t = 0.88\right)$ the value for the lift-curve slope was reduced to about 5.28. These curves show a characteristic shape similar to that obtained from two-dimensional airfoil-section data.

Rotor-Blade Pitching Moments

Rotor-blade pitching-moment data are necessary to determine the rotor control forces and are important in blade vibration and stability analyses.

Smooth blades.- A comparison of the blade pitching-moment characteristics for representative tip Mach numbers of 0.33 to 0.83 for the rotor tested (smooth condition) as a function of rotor thrust coefficient is shown in figure 7(a). The pitching-moment data represent the measured rotor-blade moments about the blade pitch axis and include aerodynamic and blade mass forces.

Changes in blade pitching-moment coefficient as thrust coefficient is increased are probably largely due to the displacement of the blade center of pressure from the blade pitch axis. Since the position of the center of pressure is not precisely known and since the actual measured pitching moments are reasonably small (-110 to 20 pound-feet) no attempt was made to separate the mass moments from the aerodynamic moments. For the purpose of this study, the presence or absence of abrupt changes in pitching-moment slopes is considered more important than the actual values.

Throughout most of the operating range the pitching moments were negative (that is, nose down). At the lower tip Mach numbers some variation of blade pitching-moment coefficient with rotor thrust coefficient is noted; whereas, at the high tip Mach numbers, pitching moments remained fairly constant with thrust coefficient. One would expect, as the blade approached maximum thrust coefficient and a portion of the blade is stalled, a large negative pitching moment due to rearward shift of the center of pressure; however, evidently the test range did not extend far enough to obtain the large negative moments.

The point at which compressibility or stall begins to increase the rotor profile power is noted on the curves (figure 7(a)). It is significant that, although the tip of the blades is encountering compressibility losses, there was very little variation in the pitching moments as thrust values were increased well beyond the onset of compressibility losses.

Leading-edge roughness.- The effect of leading-edge roughness on the blade pitching moments is shown in figure 7(b) for tip Mach numbers of 0.47, 0.74, and 0.83. Leading-edge roughness slightly reduces the nose-down moments. This reduction indicates that the center of pressure is slightly forward as compared with the smooth condition.

Rotor Profile-Drag Torque

The principal effect of compressibility and stall on the rotor performance is an increase in the rotor profile-drag torque. The dividing line between stalling losses and compressibility losses is never clearly defined and these losses are usually combined. In general, however, stalling losses are predominant in the lower blade-tip Mach number range at high angles of attack and compressibility losses are predominant at the higher blade-tip Mach numbers.

Figure 8 presents the ratios of $\left(C_{Q,o}\right)_{measured}$ to $\left(C_{Q,o}\right)_{calculated}$ plotted as a function of calculated blade-tip angle of attack. At the lowest Mach number $\left(M_t = 0.33\right)$, the ratio of profile-drag torque coefficients remains near unity up to a calculated blade-tip angle of about 8.5°. Above a blade-tip angle of 8.5°, the ratios of profile-drag torque coefficients begin to increase indicating an increased drag due to flow separation on the airfoil. Curves for the high tip Mach numbers (0.83 and 0.88) show that the ratios of profile-drag torque coefficients have already diverged even at zero tip angle of attack.

The rate of growth of profile-drag torque is also shown in figure 8. For example, at $M_t = 0.79$ the initial rate of growth of the compressibility losses indicates about an 18-percent increase in profile-drag torque per degree of blade-tip angle increase. As the blade-tip angle of attack is increased further, the rate of growth becomes twice its initial rate.

At low tip Mach numbers the blade-tip angles of attack indicated for drag divergence seem low. This point will be discussed in a later section.

An alternate method of showing the rotor-blade drag rise, is shown in figure 9 as a plot of ratios of profile-drag torque coefficient against rotor-blade mean lift coefficient. It is seen that at low speed $\left(M_t = 0.33\right)$ the maximum mean lift coefficient before drag increase is about 0.87. This value of $\bar{c}_l$ is about 8 percent lower than that obtained with a rotor having an NACA 23015 airfoil (ref. 1) and about 9 percent lower than that obtained with a rotor having an NACA 63_2-015 airfoil (ref. 2).

Comparison With Two-Dimensional Drag-Divergence Data

Since the compressibility losses can be an appreciable part of the total power, as indicated by figure 2, it is important to be able to predict the tip Mach number and tip angle of attack at which these losses occur.

A comparison of the rotor drag-divergence values with those predicted from two-dimensional airfoil data is presented in figure 10. The measured rotor drag-divergence Mach numbers were obtained from figure 8 and were taken as the point at which the profile-torque-coefficient ratio at a given calculated blade-tip angle of attack departs from a value of unity. Two curves are presented to show the trend predicted from two-dimensional airfoil data. For the upper curve, the drag-divergence Mach number is conventionally defined as the point at which the slope of the drag coefficient plotted against Mach number (at a given angle of attack) attains a value of 0.1. A better reference, and one that is more consistent with the manner in which the rotor drag-divergence Mach numbers are defined, is afforded by the lower curve. Here, the drag-divergence Mach number for the two-dimensional airfoil is taken as the point at which the variation of drag coefficient with Mach number first begins to depart from its incompressible value. The test results show that the rotor was able to operate at blade-tip Mach numbers as much as 0.1 higher before the drag rise occurred than would be predicted from the airfoil data at low angles of attack. Results of other high tip speed tests (refs. 1 and 2) showed similar trends at the lower angles of attack.

At tip angles of attack above 6°, the rotor experimental tip Mach numbers for drag divergence are lower than those indicated by the two-dimensional data. For example, at $M_t = 0.33$ the data show the drag rise starting at a tip angle of 8.5° for this particular blade. This tip angle value is lower than would be expected since both two-dimensional test data and experience with rotor blades having airfoils with about the same leading-edge radius indicate that tip angles of 10° or 11° would be more representative. It therefore seems reasonable to suspect that this low angle for drag rise is associated with the geometric characteristics for this particular blade and may not be representative of the drag rise angle of an NACA 0012 airfoil.

CONCLUSIONS

The hovering performance and blade pitching-moment characteristics of a rotor having an NACA 0012 tip section, an NACA 0018 root section, and -7.5° of twist has been determined experimentally over a tip Mach number range from 0.33 to 0.88. As a result of this investigation the following conclusions are noted:

1. The primary effect of compressibility was a rapid increase in the profile-drag torque coefficient once the critical combination of tip angle of attack and tip speed was exceeded.

2. The value of airfoil-section lift-curve slope required for an analytical prediction of the rotor thrust varied from a value of 5.73 at a tip Mach number M_t of 0.33 to a value of 6.32 at $M_t = 0.74$. The variation in lift-curve slope with Mach number has a characteristic shape similar to that shown by two-dimensional airfoil-section data.

3. It is significant that there was very little variation in the pitching moments as the rotor thrust was increased well beyond the onset of compressibility losses.

4. At the calculated tip angles of attack from 0° to about 2.5°, the rotor-blade drag-divergence Mach number measured was about 0.01 to 0.02 higher than that indicated by the two-dimensional data where Mach number for drag divergence was defined as $\frac{\Delta c_d}{\Delta M_t} = 0.1$ (where c_d is the section drag coefficient) and about 0.1 higher where Mach number for drag divergence was defined as the start of drag rise.

5. The efficiency, expressed as figure of merit, of the rotor decreased in value as the tip Mach number was increased. For example, at $M_t = 0.33$ the maximum figure of merit was about 75 percent and was reduced to a value of about 50 percent at $M_t = 0.83$.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., June 12, 1957.

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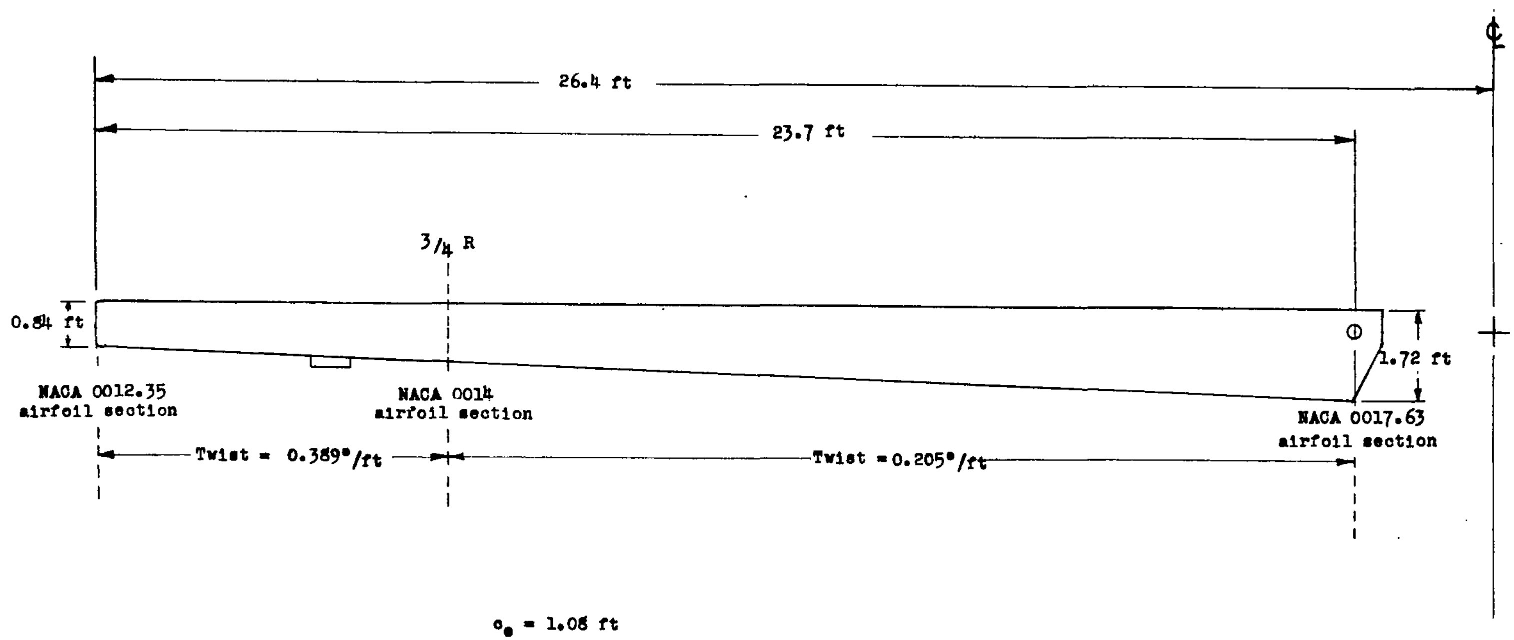
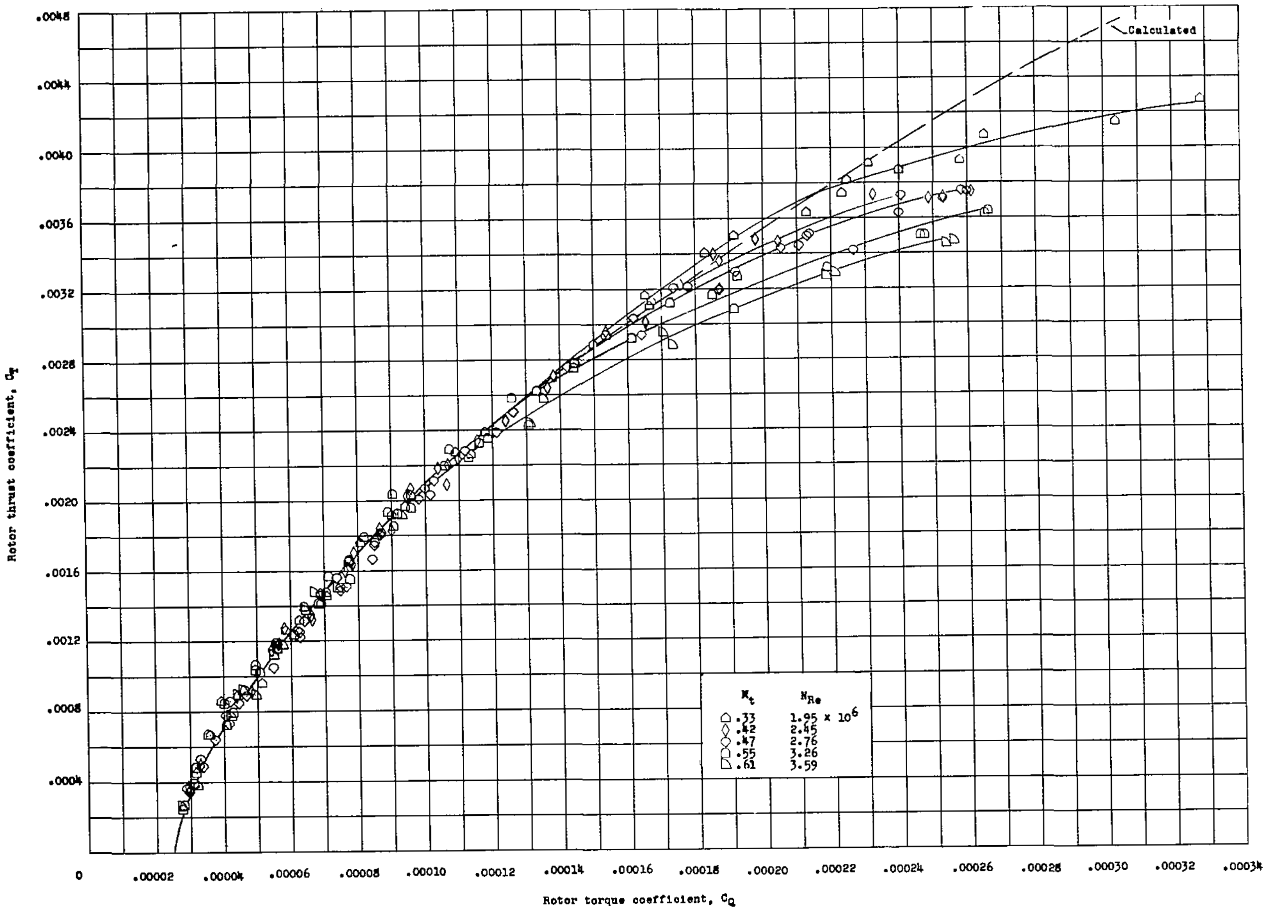
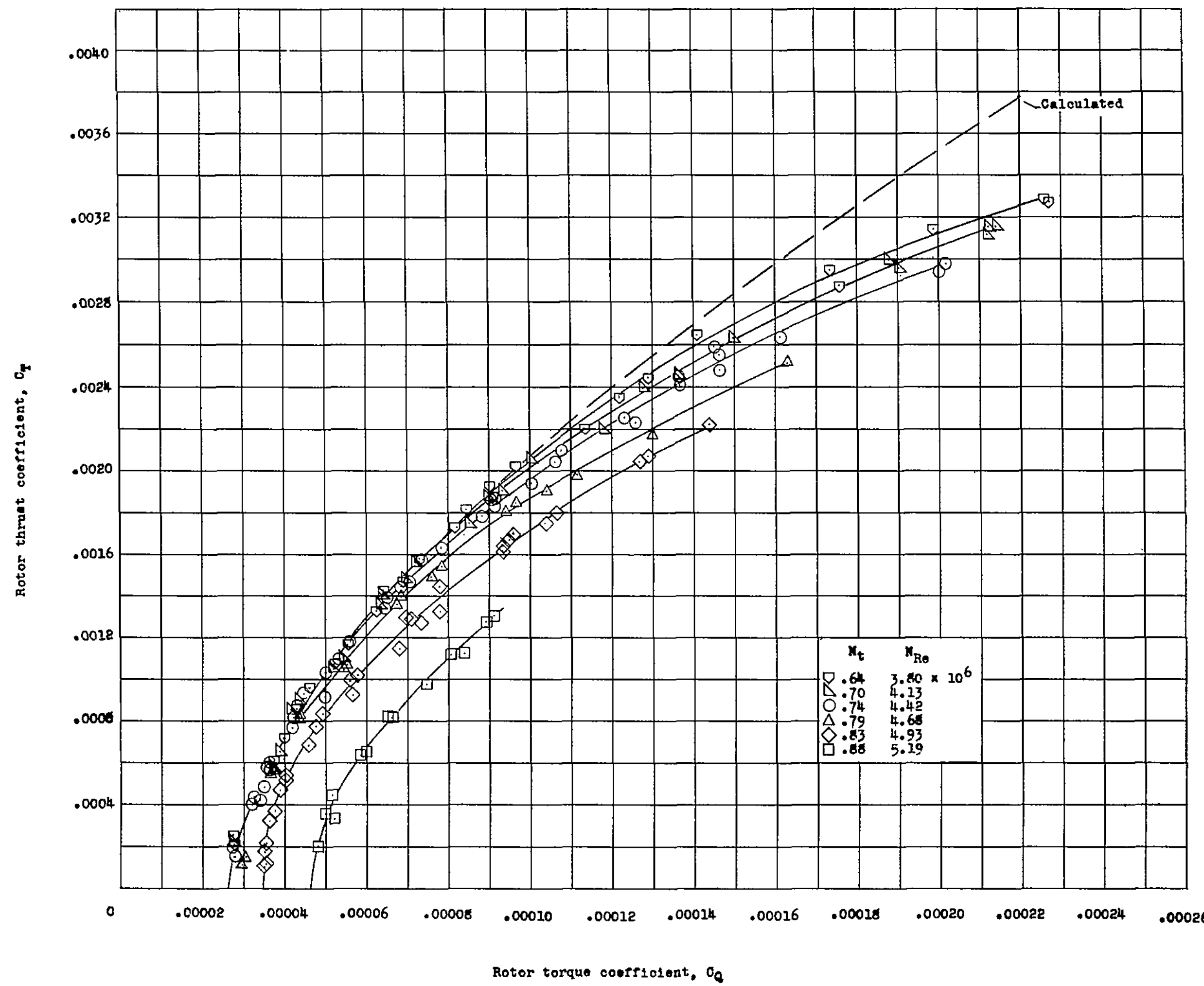


Figure 1.- Sketch of rotor blade having an NACA 0012 tip section and an NACA 0018 root section.



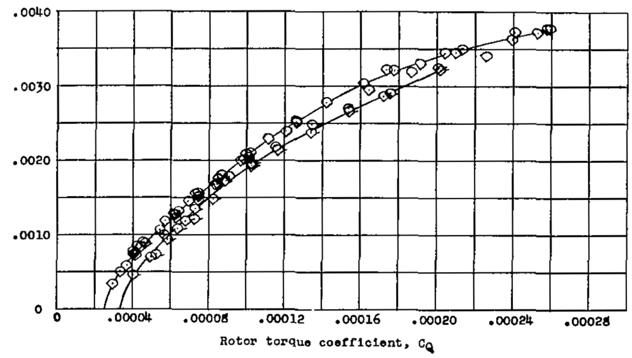
(a) $M_t = 0.33$ to 0.61.

Figure 2.- Hovering performance of rotor blades (factory production finish) having NACA 0012 airfoil tip sections. Calculated curve based on $c_{d,o} = 0.0087 - 0.0216\alpha_r + 0.400\alpha_r^2$ and $c_l = a\alpha_r$; $\sigma = 0.026$.

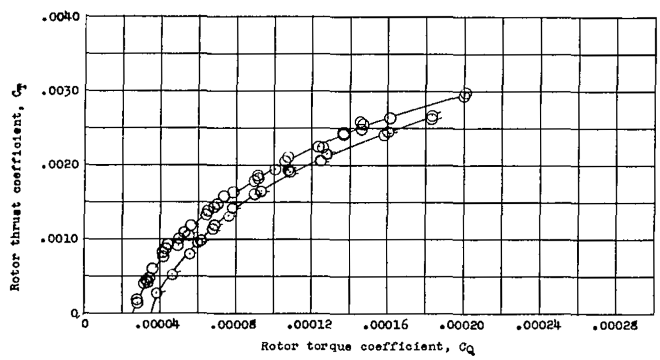


(b) $M_t = 0.64$ to 0.88.

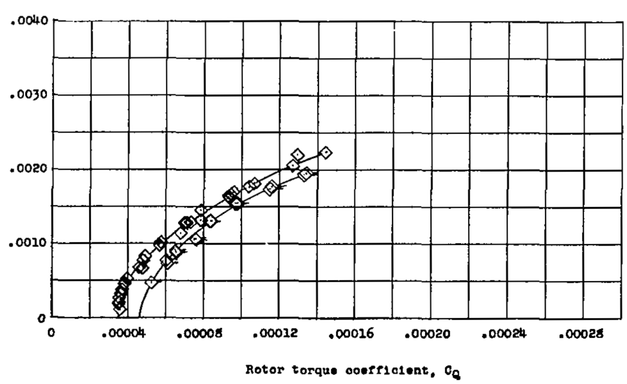
Figure 2.- Concluded.



(a) $M_t = 0.47$.



(b) $M_t = 0.74$.



(c) $M_t = 0.83$.

Figure 3.- Effect of blade leading-edge roughness on rotor hovering performance of factory production finish blade. Flagged symbols denote leading-edge roughness.

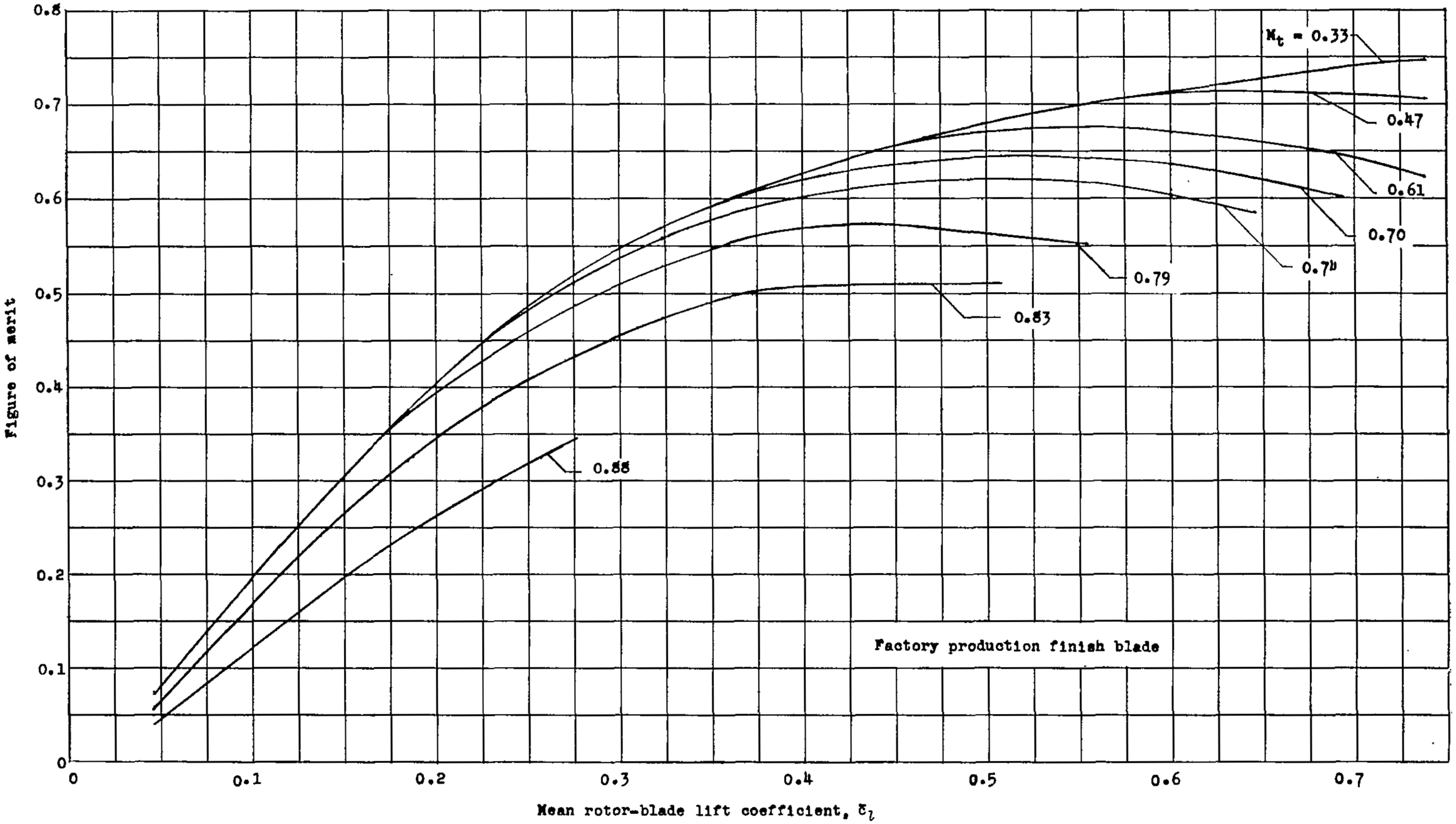


Figure 4.- Effect of tip Mach number on rotor figure of merit.

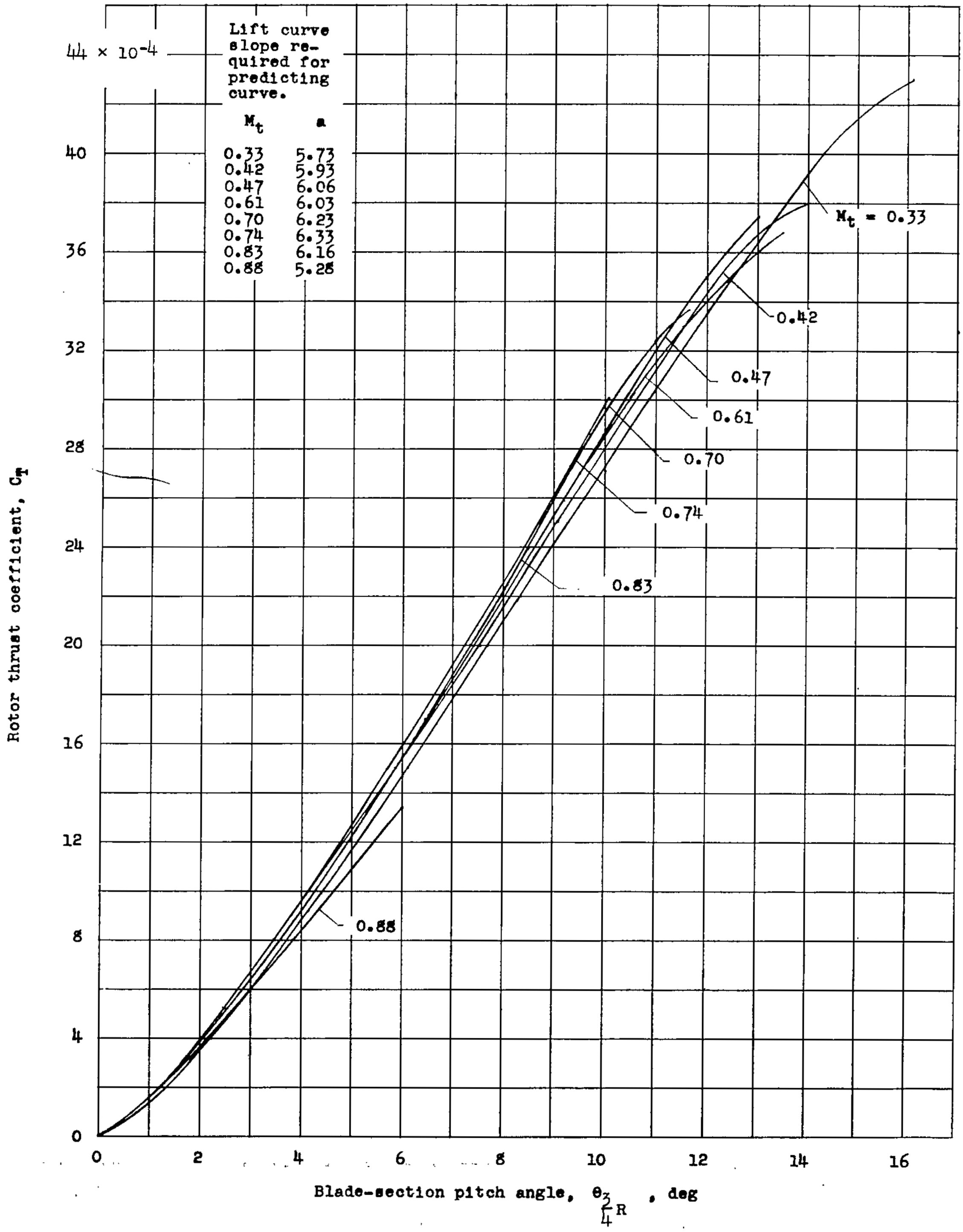


Figure 5.- Effect of tip Mach number on blade lift-curve slope.

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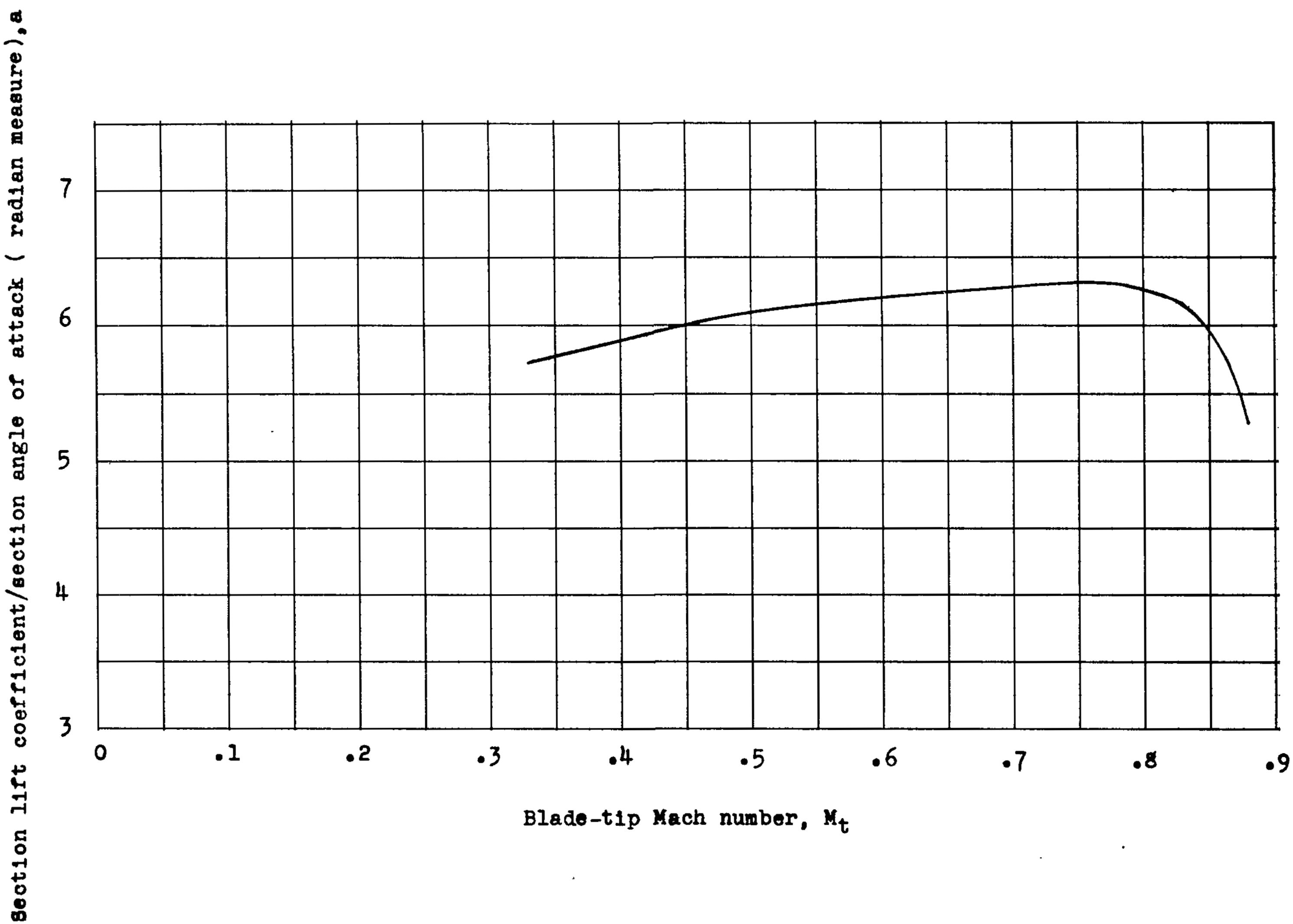
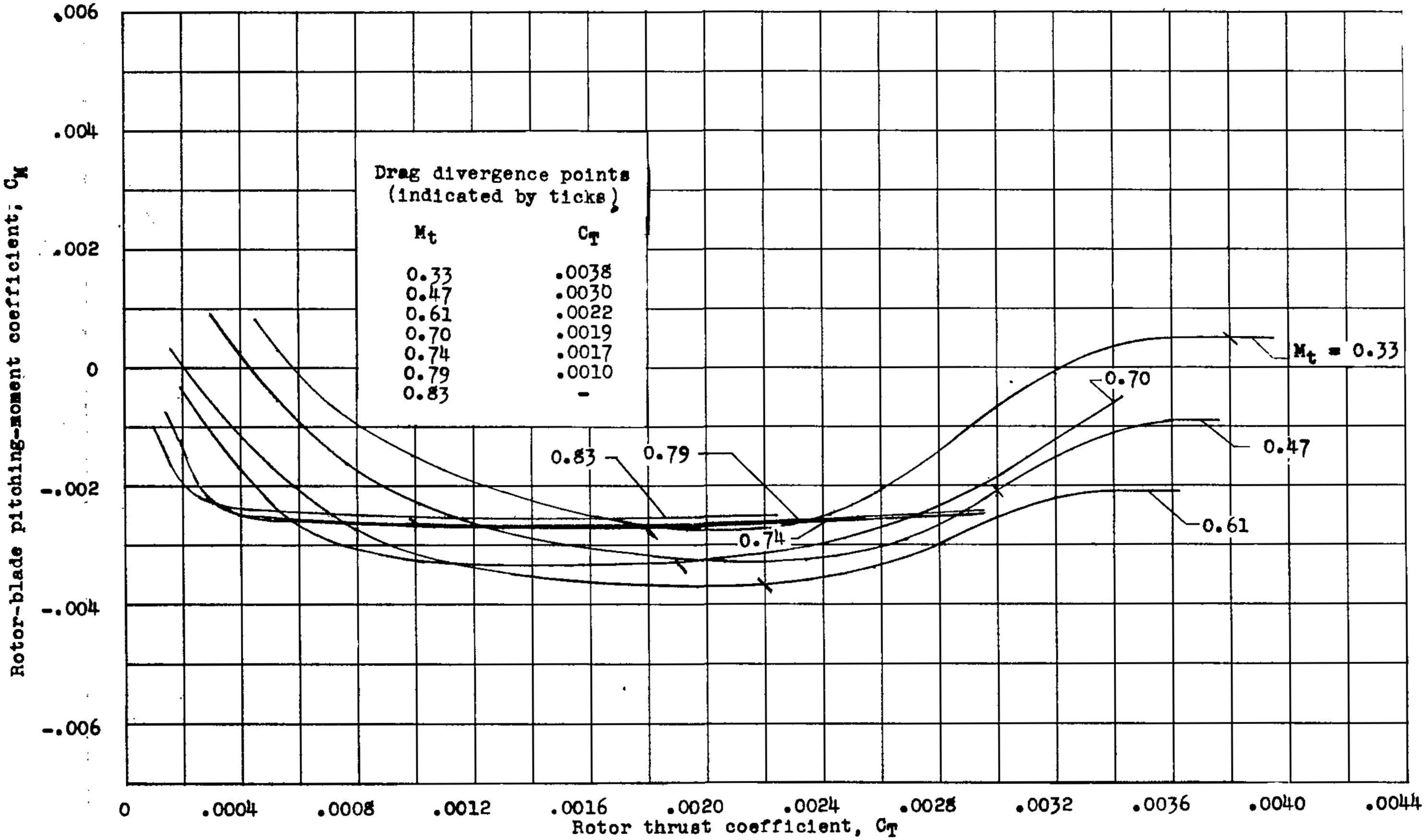
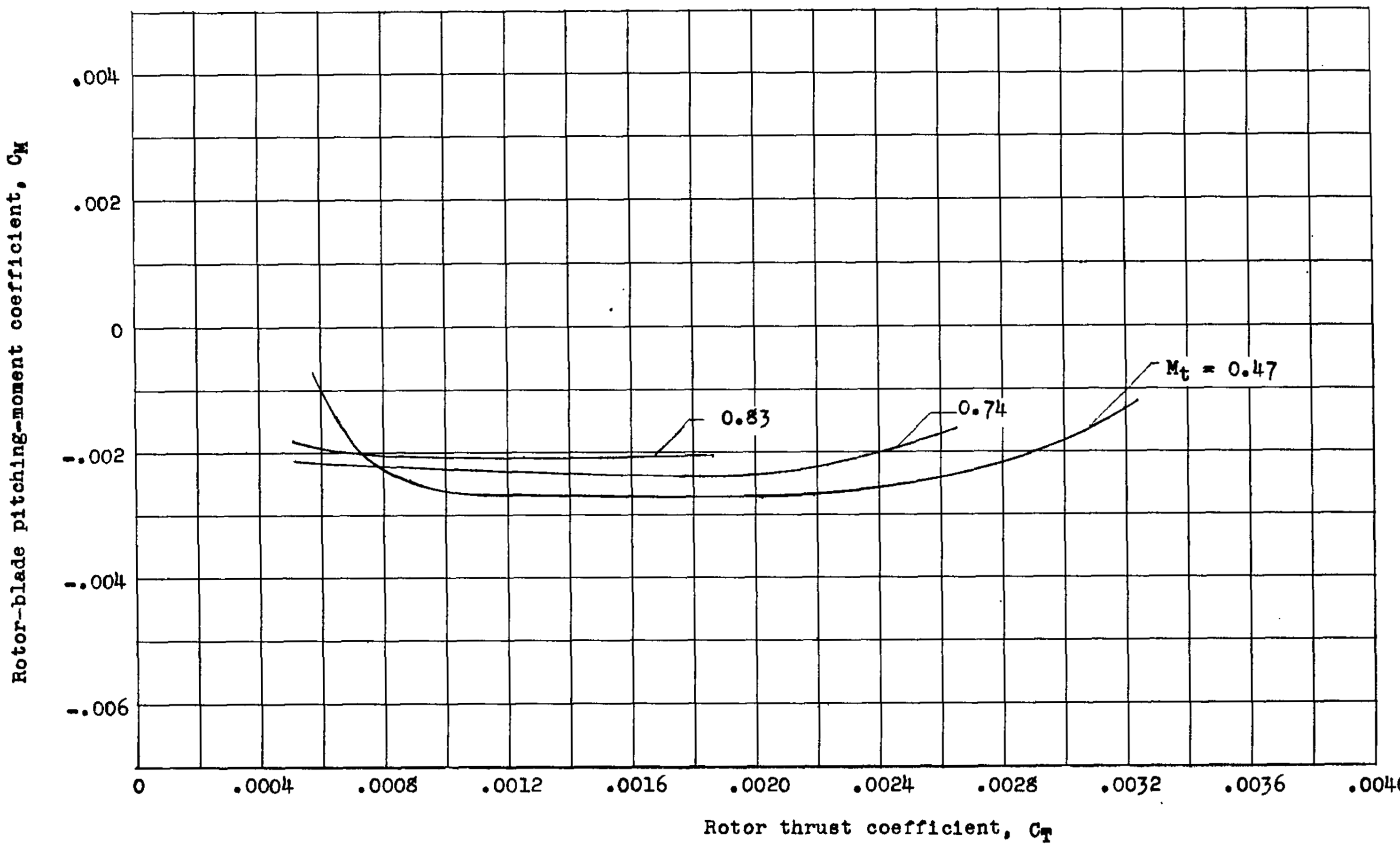


Figure 6.- Lift-curve slope required for predicting rotor thrust at various Mach numbers.



(a) Smooth.

Figure 7.- Effect of tip Mach number on the pitching moments of rotor blades (factory production finish) having NACA 0012 tip airfoil sections.



(b) Leading-edge roughness.

Figure 7.- Concluded.

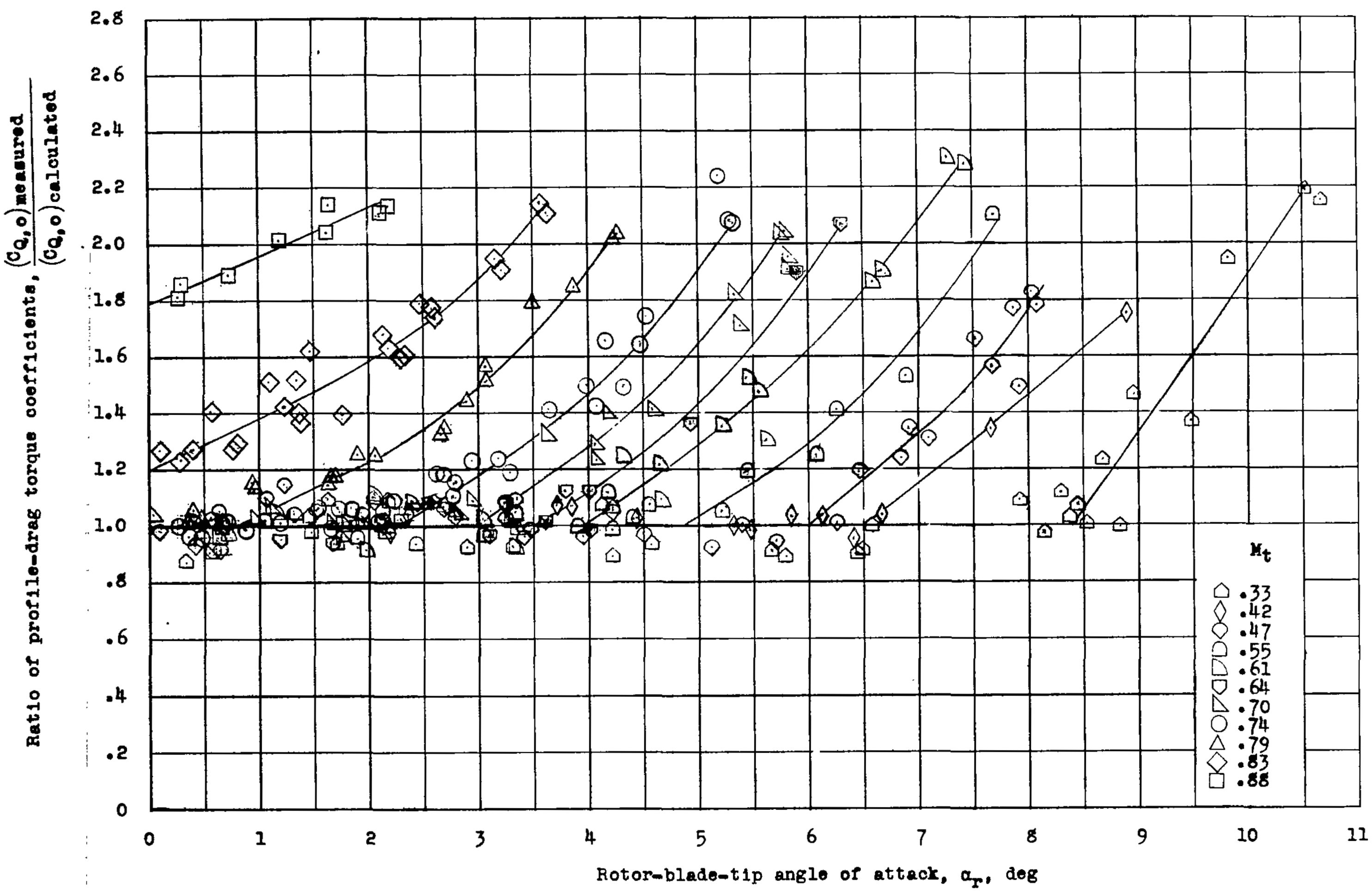


Figure 8.- Effect of tip angle of attack and Mach number on profile-drag torque of factory production finish rotor blade.

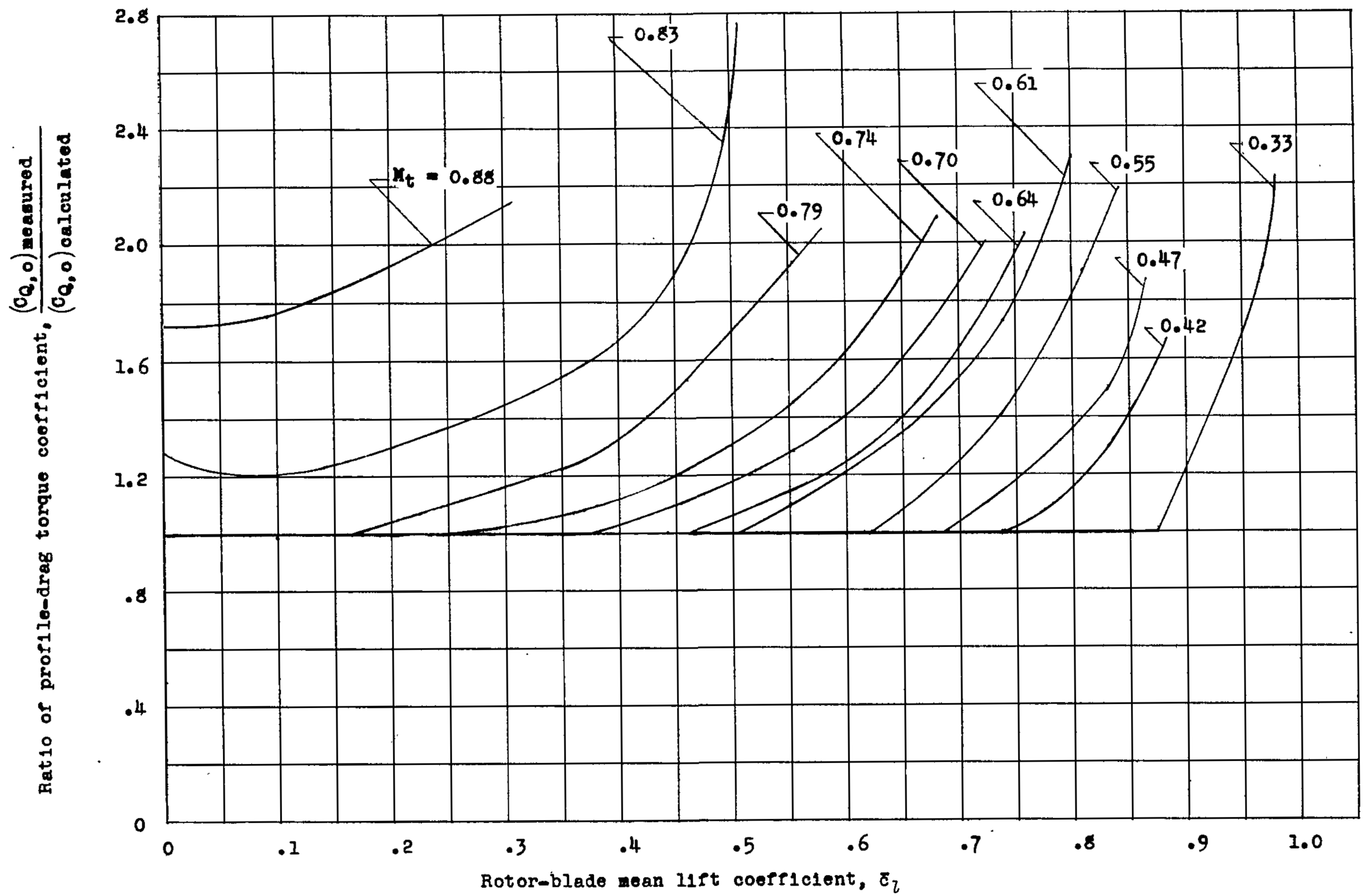


Figure 9.- Effect of rotor-blade mean lift coefficient and Mach number on profile-drag torque of factory production finish rotor blade.

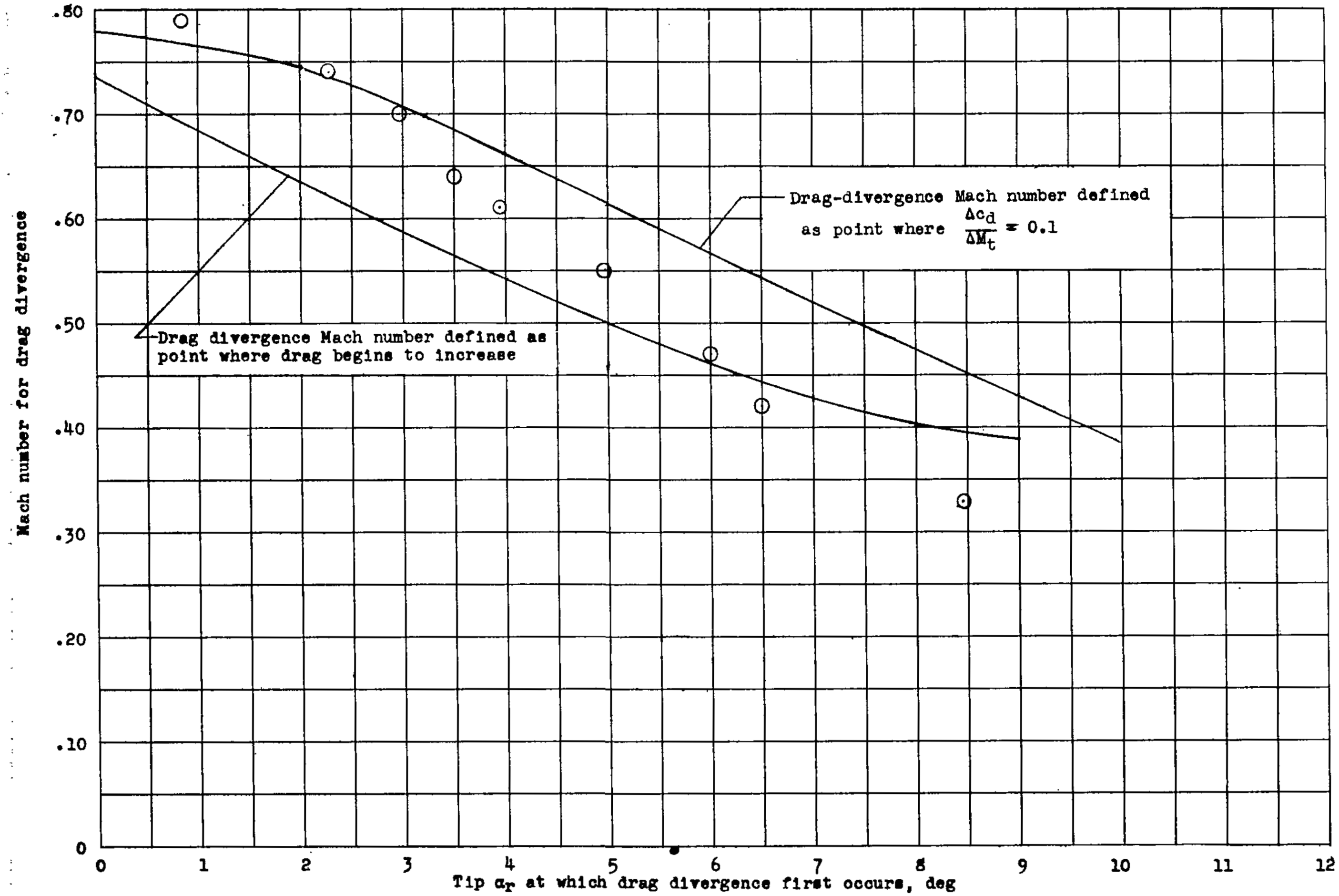


Figure 10.- Comparison of two-dimensional airfoil drag-divergence data with the rotor experimental data. Curves from unpublished two-dimensional data for NACA 0012 airfoil; symbols represent M_t and α_r at which experimental data from the factory production rotor blade separate from the curve calculated by using the airfoil drag polar $c_{d,o} = 0.0087 - 0.0216\alpha_r + 0.400\alpha_r^2$.