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ADVANCE REPORT

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EFFECT OF MACH AND REYNOLDS NUMBERS ON THE POWER-OFF MAXIMUM LIFT COEFFICIENT OBTAINABLE ON A P-39N-1 AIRPLANE AS DETERMINED IN FLIGHT

By James M. Nissen and Burnett L. Gadeberg

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Moffett Field, California

Washington
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ADVANCE [redacted] REPORT

EFFECT OF MACH AND REYNOLDS NUMBERS ON THE POWER-OFF MAXIMUM LIFT COEFFICIENT OBTAINABLE ON A P-39N-1 AIRPLANE AS DETERMINED IN FLIGHT

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SUMMARY

Flight tests were conducted on a Bell P-39N-1 airplane to determine the variation with Mach and Reynolds numbers of the maximum lift coefficient obtainable. The tests were made with the propeller in the constant-speed high-pitch setting and with the power off in order to minimize the effect of the power and the propeller. The investigation covered a Mach-number range of 0.146 to 0.655 and a Reynolds-number range of 4,900,000 to 17,100,000.

The results of the investigation indicated that the maximum lift coefficient obtainable decreased markedly with and increase in Mach number for a given Reynolds number and exhibited a tendency to increase with an increase in Reynolds number for a given Mach number up to a Mach number of 0.55.

INTRODUCTION

The maneuverability, turning performance, and the maximum maneuvering loads of a modern fighter-type airplane are closely associated with the maximum lift coefficients obtainable in flight; and since the maximum lift coefficient obtainable is a function of Mach and Reynolds numbers, the suitability of fighter airplanes at various altitudes and airspeeds may be seen to be partially determined by this function.

The data presented herein were obtained incidental to other research being conducted on the airplane, and it is

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intended to supplement them with more complete data for both the power-on and power-off conditions in the near future.

Although some data have been obtained in wind tunnels on the effect of Mach and Reynolds numbers on the maximum lift coefficient, comparatively little data on this subject have been obtained in flight. It is believed, therefore, that this information also may prove of some value in correlating wind-tunnel and flight data.

DESCRIPTION OF THE AIRPLANE

The Bell P-39N-1 airplane is a single-place, single-engine, low-wing, cantilever monoplane with retractable landing gear and partial-span split flaps. Figure 1 is a three-view drawing of the airplane and figure 2 shows the airplane as instrumented for flight tests. The general specifications of the airplane are as follows:

Airplane	
Name and type	Bell P-39N-1-BE Army Air Forces Number 42-18476
Engine	
Name and type	Allison, liquid-cooled V-12, in-line, V-1710-85
Propeller	Aeroproducts, constant-speed
Diameter	11 ft 7 in.
Number of blades	3
Hub drawing number	A-632S-C1
Blade drawing number	A-20-156-17
Weight	
Empty	5761 lb
Gross (normal)	7663 lb
Weight, as flown (at take-off)	7990 lb

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Wing loading (normal)	35.9 lb per sq ft
Power loading (normal)	7.66 lb per bhp
Center of gravity (as flown)	0.255 M.A.C.
Wing	
Area	213.22 sq ft
Span	34 ft
Root chord (22 in. from center line)	98.6 in.
Tip chord (204 in. from center line)	50 in.
Taper ratio	1.97
Incidence	2°
Dihedral	4°
Sweepback (leading edge)	4°35'
Aspect ratio	5.42
M.A.C.	
Length	80.64 in.
Location	Leading edge of M.A.C. is 5.41 in. aft and 6.62 in. above leading edge of root chord
Airfoil	
Root	NACA 0015
Tip	NACA 23009
Thickness ratio (measured)	
Station (from fuselage center line)	
22	0.1480
46	0.1448

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70	0.1400
94	0.1338
118	0.1261
142	0.1180
166	0.1091
190	0.1000

Horizontal tail surfaces

Area	40.99 sq ft
Span	156 in.
Maximum chord	52.25 in.
Distance from normal center of gravity to one-third maximum chord	14.95 ft

Elevator (modified Handley-Page type)

Area (including balance)	16.89 sq ft

Stabilizer

Area	24.1 sq ft
Incidence	2°15'

Vertical tail surfaces

Fin

Area	7.94 sq ft
Setting	1° to left

Rudder (modified Handley-Page type)

Area (including balance)	11.07 sq ft

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INSTRUMENT INSTALLATION

Standard NACA flight instruments were used to obtain photographic records from which airspeed, altitude, and the normal acceleration factor were determined. The airspeed and altitude recorders were separately connected to an airspeed head consisting of a total-head and two static tubes. The airspeed head, which was vaned and pivoted about two axes so as to aline itself with the relative wind, was located on a boom extending 65 percent of the local chord ahead of the leading edge of the wing at the right outboard gun location. The airspeed and altitude recorders were mounted at the base of the boom permitting the use of short and about equal lengths of connecting tubing, thereby minimizing the effects of lag in the airspeed and altitude recordings. The total-head and static lines of the airspeed recorder were so balanced that air flow caused by a change of density during flight would cause no error in the recorded airspeed.

The calibration of the airspeed recorder was determined by recording the pressure in the static side of the pitot-static head, as measured by the recording altimeter, while flying past a reference point the pressure altitude of which was known at the time the tests were made. The calibration was then determined by assuming that the total error in airspeed was that due to the error in the measurement of static pressure. This error manifested itself by the variation of the recorded altitude as the airplane was flown at the constant-pressure attitude at different airspeeds. Although the calibration was made at low altitudes, it was also used to determine airspeeds at high altitudes by assuming that the Mach number effects on the pitot-static head itself, and on the blocking effect of the wing and supporting tube, were negligible and that the calibration was primarily a function of lift coefficient. The calibration curve arrived at by the above procedure is shown in figure 3. It was realized that a change in the spanwise distribution of lift due to Mach number might affect the position error at a given over-all lift coefficient; but, from the small variation of this error with lift coefficient at the higher values of the lift coefficient (fig. 3), it was inferred that any discrepancy arising from this effect would be small. Using the standard formula for calibrating airspeed indicators, the correct indicated airspeed is defined by

$$V_i = 1703 \left[\left(\frac{H - p}{29.92} + 1 \right)^{0.286} - 1 \right]^{\frac{1}{2}}$$

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where

V_i indicated airspeed, miles per hour

H free-stream total pressure, inches of mercury

p free-stream static pressure, inches of mercury

TESTS, RESULTS, AND DISCUSSION

The data were obtained by making turns and pull-ups to the stall at 5,000, 10,000, 15,000, 20,000, and 25,000 feet pressure altitude. Turns to the stall were made up to an indicated airspeed of 225 miles per hour by the pilot's meter, and pull-ups were made above 250 miles per hour. In all cases the stall was approached as slowly as possible in order to minimize any effect of a rapid change in the angle of attack on the maximum lift. The stall was characterized by buffeting and pitching down of the airplane at the higher Mach numbers and by buffeting and rolling to the right at the lower Mach numbers. From this it is inferred that in the latter case tip stall was occurring while in the former case, root stall; and a consequent change in downwash at the tail which induced the nose-down tendency was present. This point is emphasized since the change with Mach number of the characteristic which limited the maximum lift obtainable and the unavoidable interrelation with controllability presumably affects importantly the trend of the maximum lift coefficient with Mach number as determined from the flight tests. If the data herein are used for any comparisons with wind-tunnel results, this fact should be borne in mind. The term "maximum lift coefficient," or "maximum normal acceleration factor," as used hereinafter in this report shall be understood to mean the maximum practical value as limited by both the effect of compressibility upon the wing itself and by the resultant effect upon the controllability of the airplane. The tests were conducted with the power off and with the propeller in the constant-speed, high-pitch position in an attempt to minimize, as far as possible, the effect of the power and the propeller.

Although the weight of the airplane as flown for the tests was 7990 pounds at take-off, allowance was made for the gasoline consumed until the time of the tests. The Mach number was determined from simultaneous readings of corrected indicated airspeed and corrected pressure altitude

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and the Reynolds number was determined from these same values of airspeed and altitude and from a knowledge of the free-air temperature. The values of free-air temperature were obtained from the average of radiosonde observations taken during the morning and evening of the dates of the flight tests. These temperatures are shown in figure 4.

In calculating the maximum lift coefficient from the flight data, the lift was assumed equal to WA_z. It is estimated that the error involved in this assumption is only of the order of 1 percent.

Thus

$$C_{L_{max}} = \frac{WA_z}{qS}$$

where

$C_{L_{max}}$ lift coefficient at the stall

W weight of the airplane, pounds

A_z acceleration factor at the stall measured perpendicular to the thrust line of the airplane (positive in steady normal flight). The acceleration factor is the recorded normal acceleration in feet per second per second divided by g.

S wing area, square feet

q dynamic pressure at the stall, pounds per square foot

$$= \frac{1}{2}\rho V^2 = \frac{H - p}{\left(1 + \frac{M^2}{4} + \frac{M^4}{40} + \ldots\right)}$$

V true airspeed, feet per second

ρ air density, slugs per cubic foot

M Mach number, V/a

a speed of sound, feet per second

g acceleration of gravity, 32.2 feet per second per second

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Curves of the normal acceleration factor obtained at the stall, plotted as a function of indicated airspeed for 5,000, 10,000, 15,000, 20,000, and 25,000 feet pressure altitude, are shown in figure 5. In this figure the accelerations have been corrected to an airplane weight of 7500 pounds, which represents a wing loading of 35.2 pounds per square foot. The indicated stalling speed at a normal acceleration factor of one was the same at all altitudes tested. During the stalls at $A_z = 1$ the Mach and Reynolds numbers varied from 0.146 and 5,600,000 at 5,000 feet altitude to 0.220 and 4,700,000 at 25,000 feet altitude.

Values of the normal acceleration factor obtained at the stall are shown in figure 6 plotted as a function of altitude for constant values of indicated airspeed. It is apparent that the maximum acceleration factor obtainable for a constant indicated airspeed decreases only slightly with an increase of altitude for low indicated airspeeds, but decreases by progressively greater amounts with increasing altitude at higher indicated airspeeds.

Curves, derived from figure 5, of the maximum lift coefficient plotted as a function of Mach number for each of the five altitudes are shown in figure 7. It will be noted that there is some difference between the variations of maximum lift coefficient with Mach number at different altitudes. This is attributed to the decrease of Reynolds number with altitude.

Through the use of figure 8, which shows the variations of Mach and Reynolds numbers at various altitudes at the time the test flights were made, curves of the variation of the maximum lift coefficient with Mach number for various Reynolds numbers were determined as shown in figure 9. The maximum lift coefficient decreased markedly with Mach number for all Reynolds numbers tested.

Figure 8 was also used to determine the variation of the maximum lift coefficient with Reynolds number for various Mach numbers as shown in figure 10. It is apparent that even at relatively high Mach numbers the maximum lift coefficient is influenced by changes in Reynolds number. Thus, if the maximum lift coefficient obtained at one altitude at a given Mach number is to be applied to compute performance at other altitudes, the effect of Reynolds number as well as that of Mach number should be considered.

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CONCLUSIONS

On the basis of the tests conducted, the following conclusions may be drawn for the P-39N-1 airplane:

1. The maximum practical lift coefficient, or normal acceleration factor, obtainable in flight was limited not only by the effects of compressibility upon the wing itself, but also by the related effect upon the controllability of the airplane.

2. Above a Mach number of 0.2 the maximum lift coefficient obtainable decreased with an increase in Mach number for a given Reynolds number at an average rate, $\Delta C_L/\Delta M$, of approximately -1.5.

3. Above a Mach number of 0.2 the maximum lift coefficient obtainable exhibited a tendency to increase with an increasing Reynolds number for a given Mach number up to a Mach number of 0.55.

4. The increase in the maximum obtainable normal acceleration factor with indicated airspeed is less for flight at high altitudes than for flight at low altitudes.

5. For a given indicated airspeed the maximum lift coefficient obtainable decreases with an increase in altitude due to an increase in Mach number and a decrease in Reynolds number.

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National Advisory Committee for Aeronautics,
Moffett Field, Calif.

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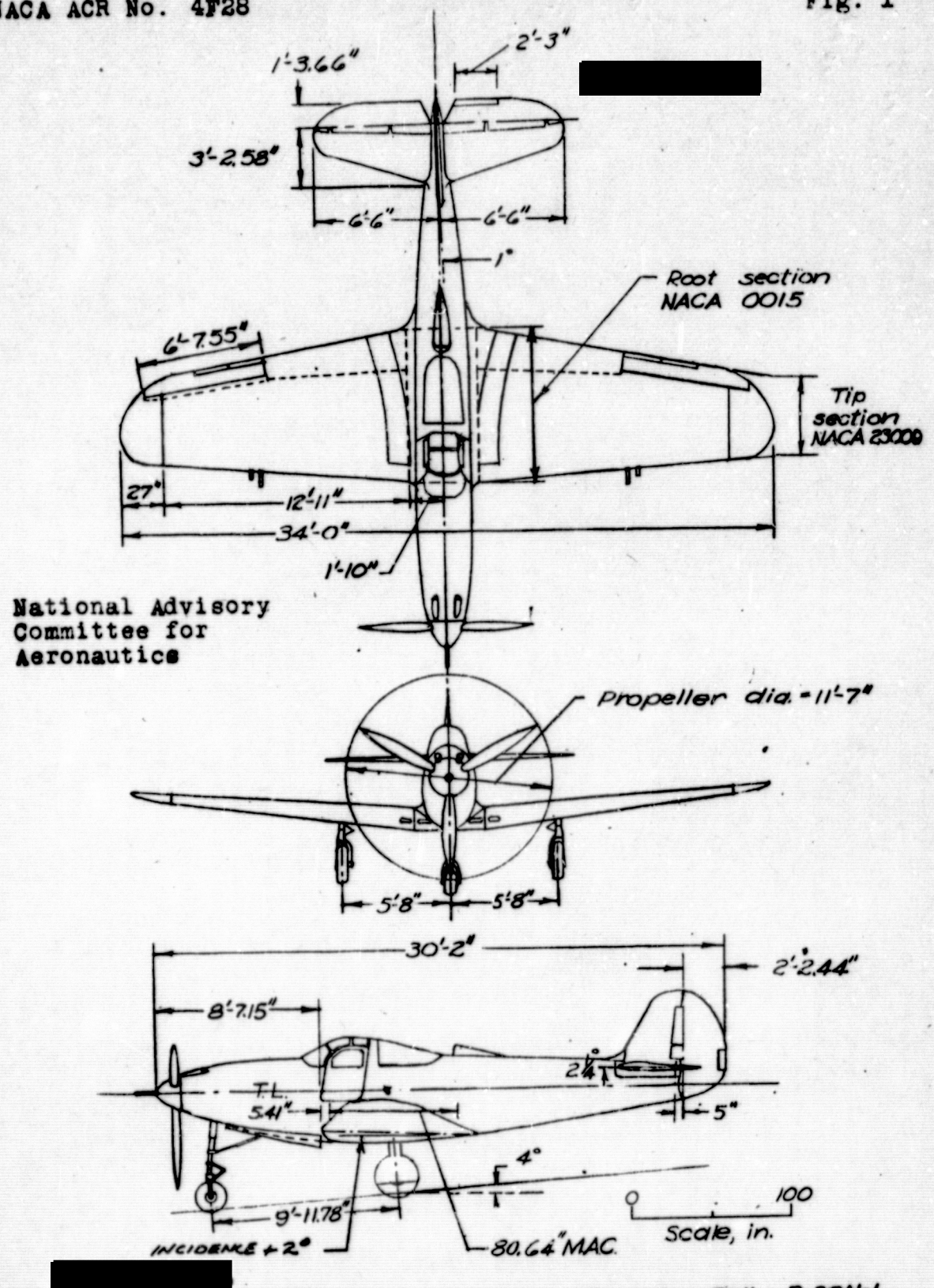


Figure 1.- Three - view drawing of the Bell P-39N-1 airplane.

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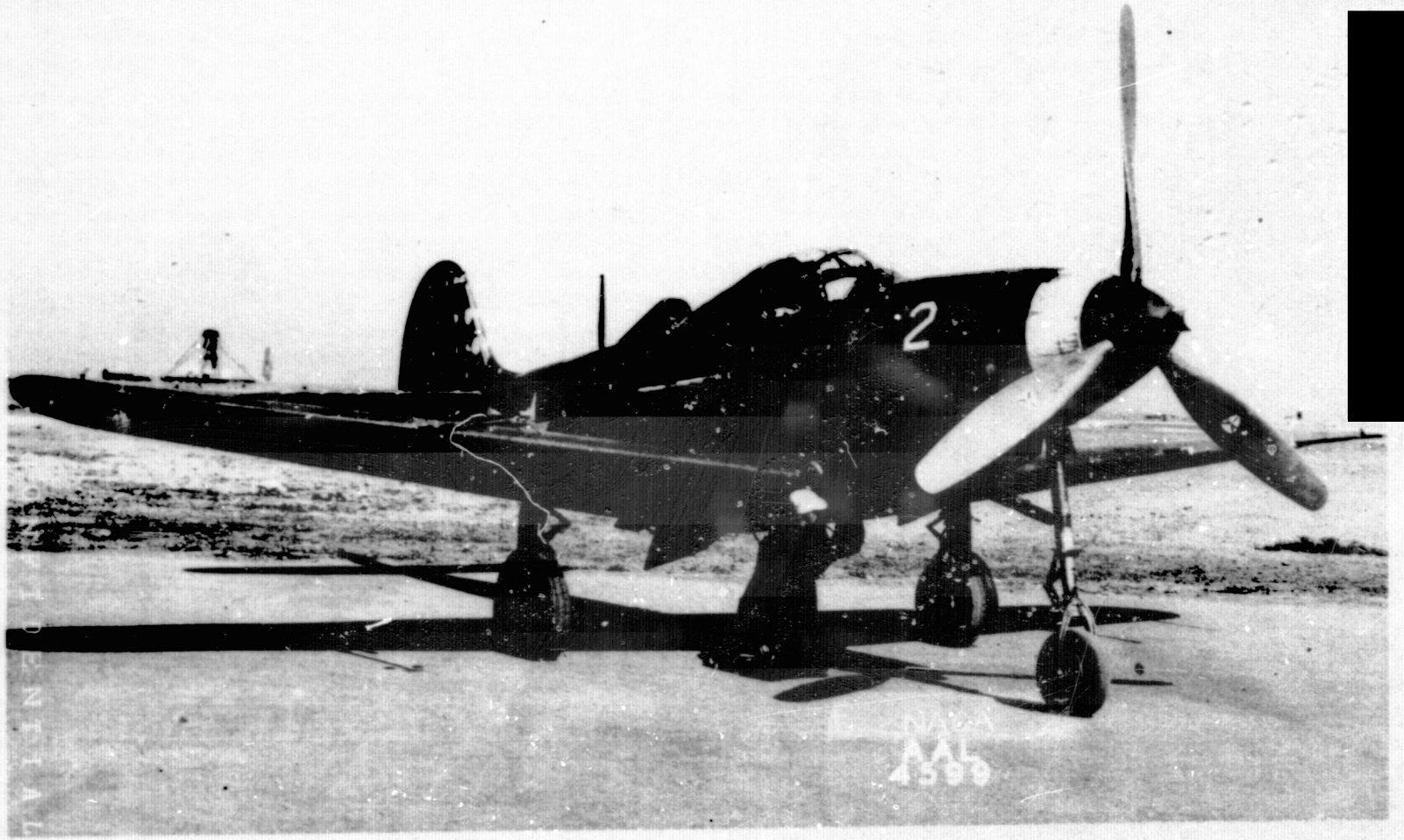


Figure 2.- The P-39N-1 airplane as instrumented for test flights.

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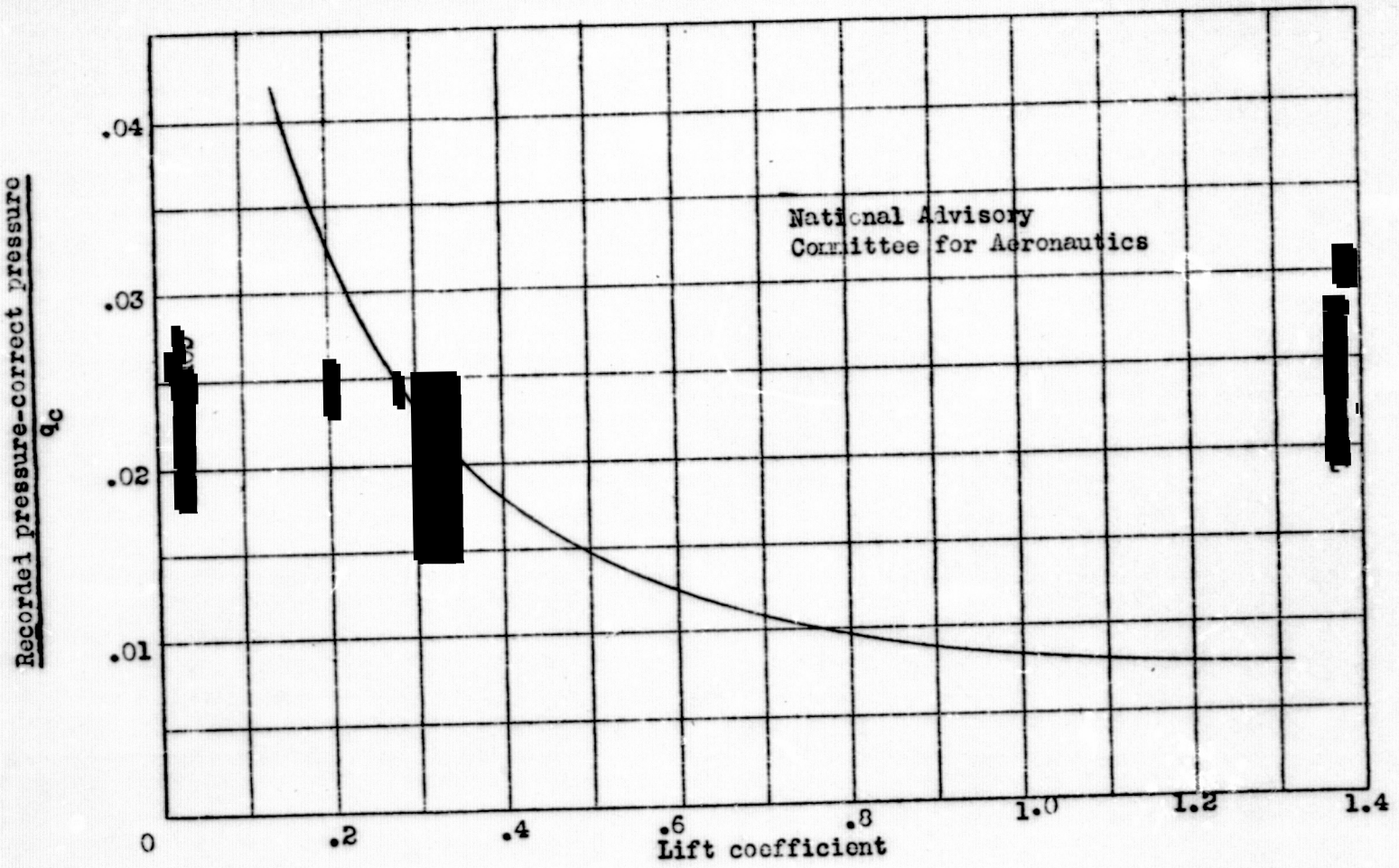


Figure 3.- Error in the recorded static pressure as a function of the lift coefficient. P-39N-1 airplane.

Fig. 4

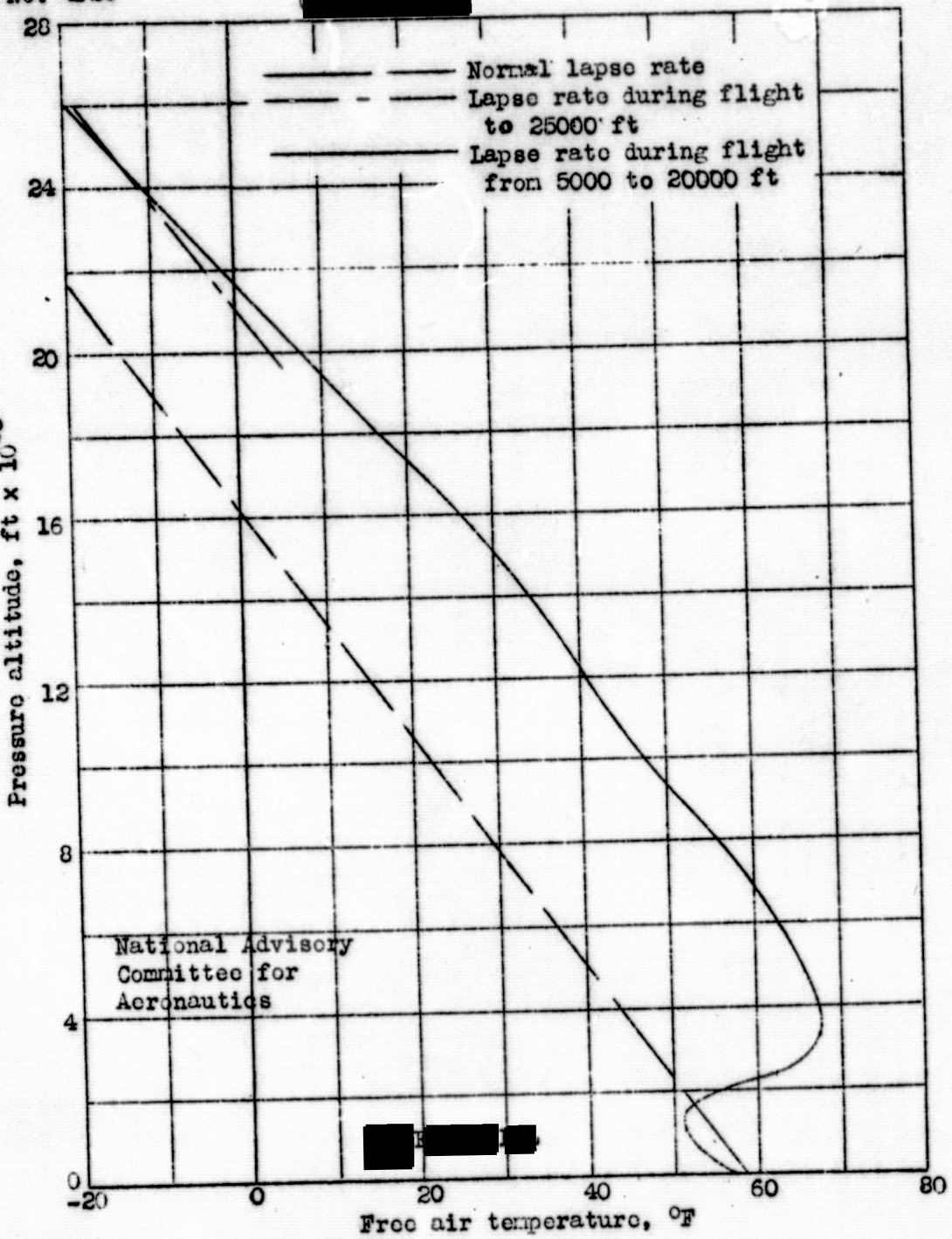


Figure 4.- Atmospheric lapse rates existing during flights from 5,000 to 25,000 feet altitude, P-39N-1 airplane.

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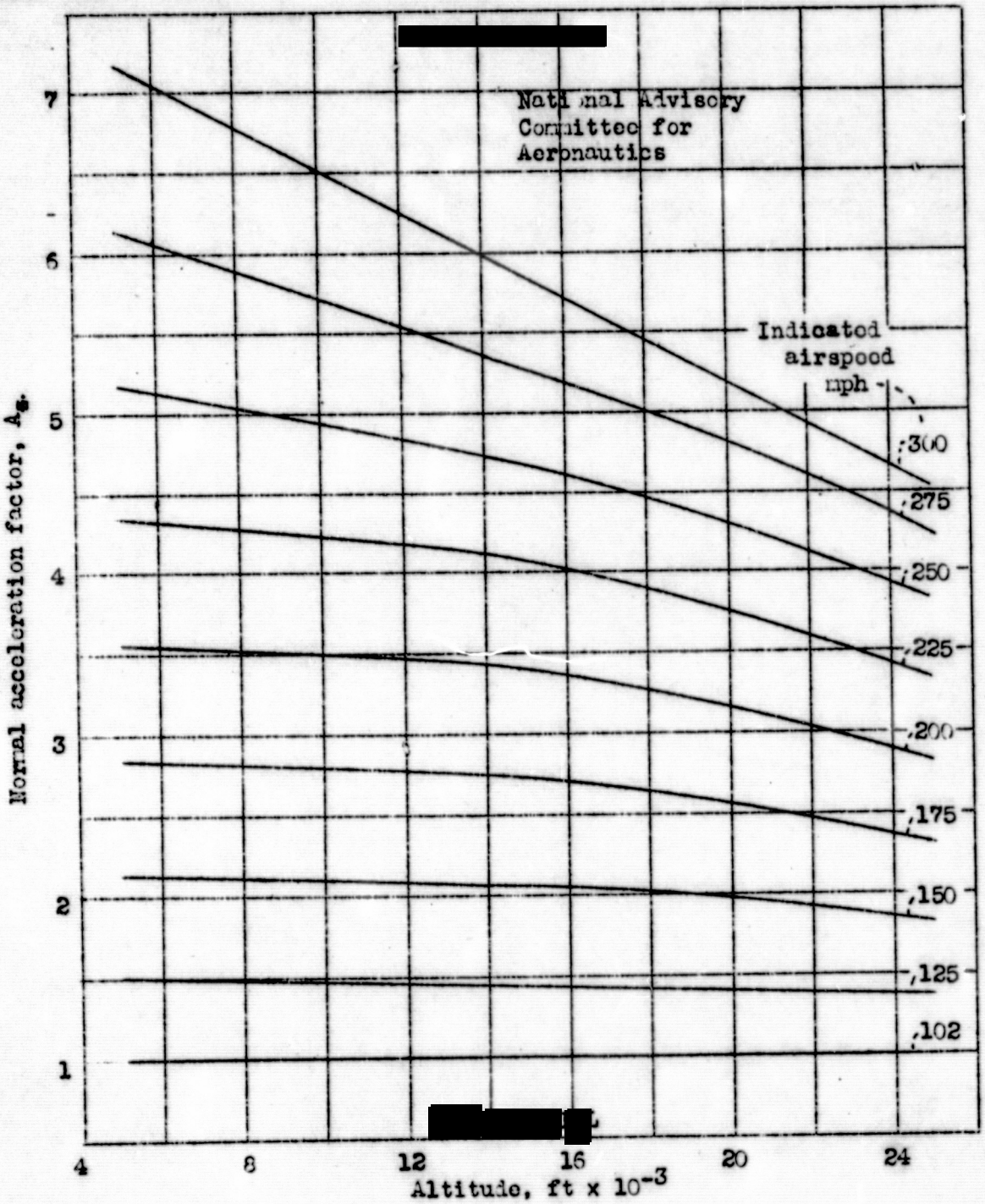


Figure 6.- Maximum normal acceleration factor obtainable at stall plotted as a function of altitude for flight at various airspeeds, power off, P-39N-1 airplane.

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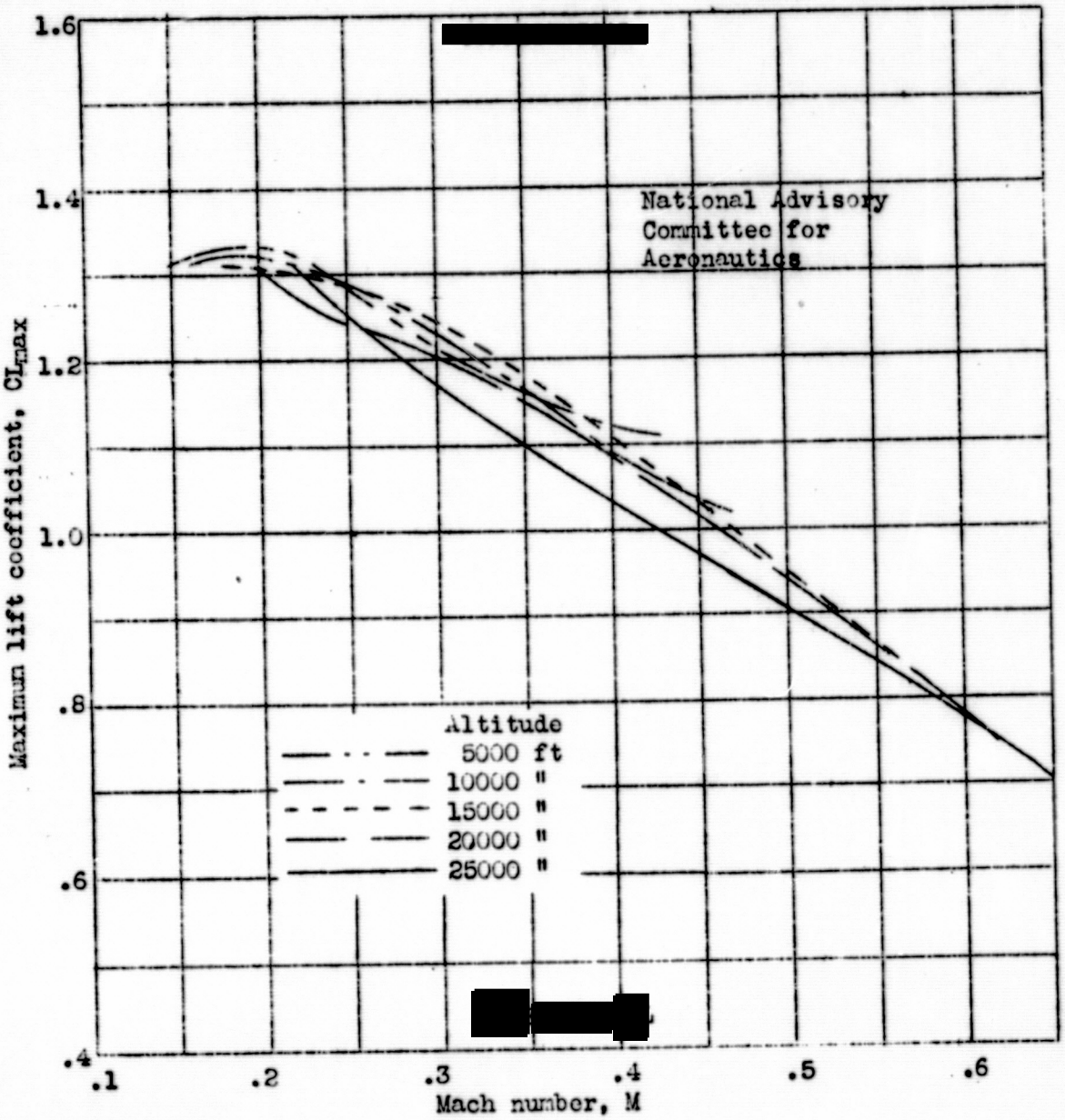


Figure 7.- Variation of maximum lift coefficient with Mach number at various altitudes, power off, P-39N-1 airplane.

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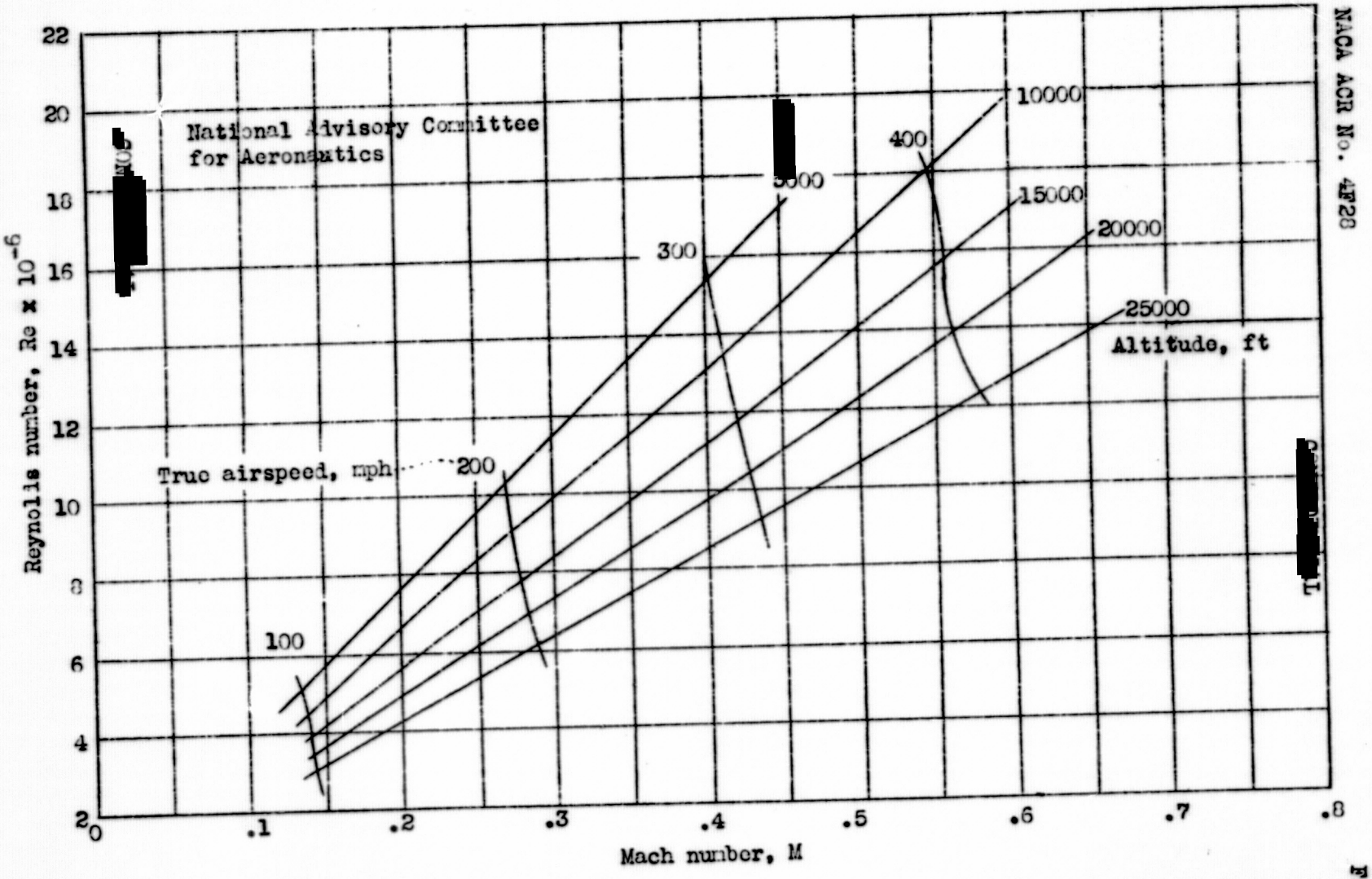


Figure 8.- Variation of Reynolds number with Mach number during flights from 5,000 to 25,000 feet altitude, P-39N-1 airplane.

Fig. 8

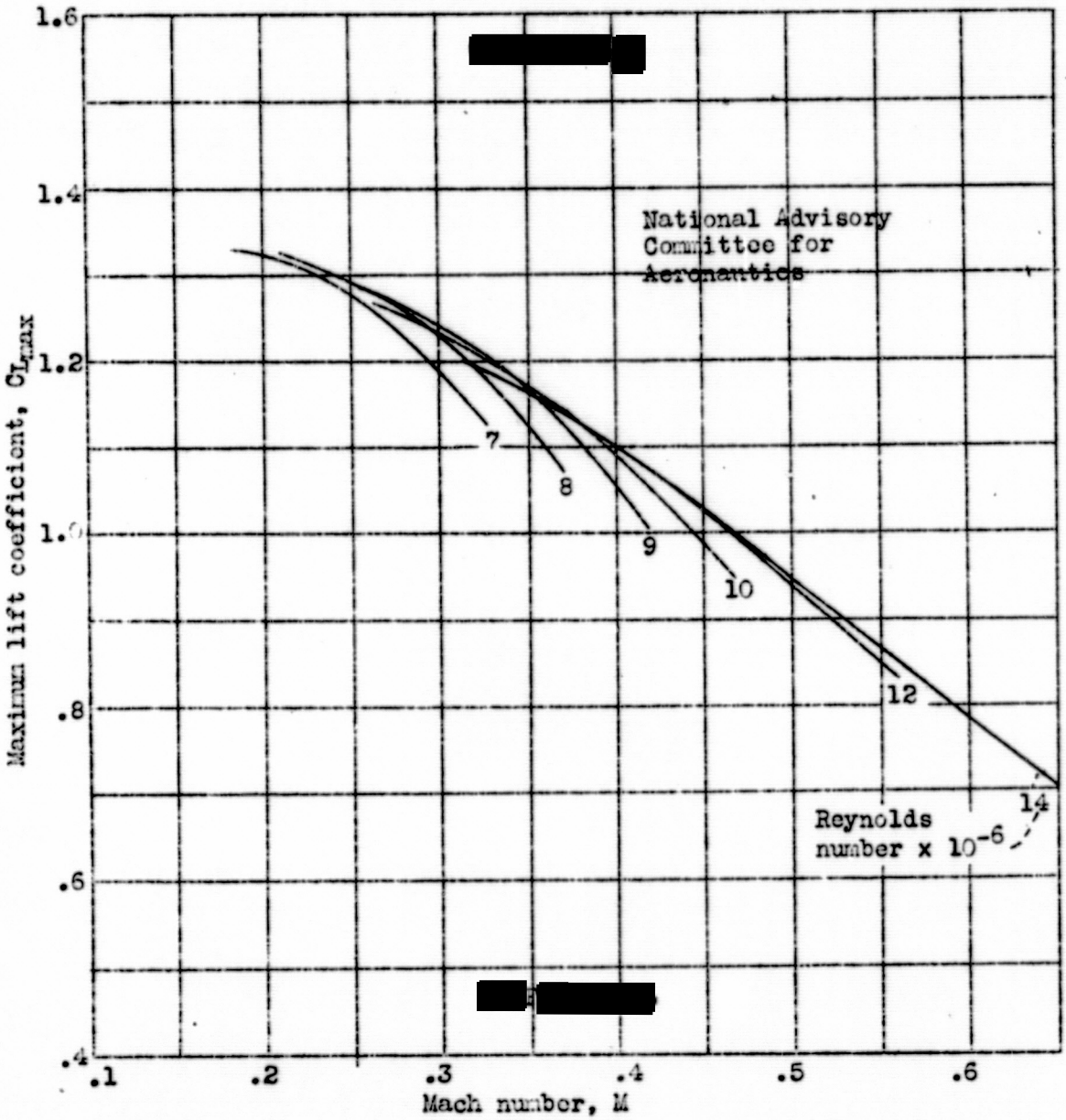


Figure 9.- Variation of maximum lift coefficient with Mach number for constant values of Reynolds number, power off, P-39N-1 airplane.

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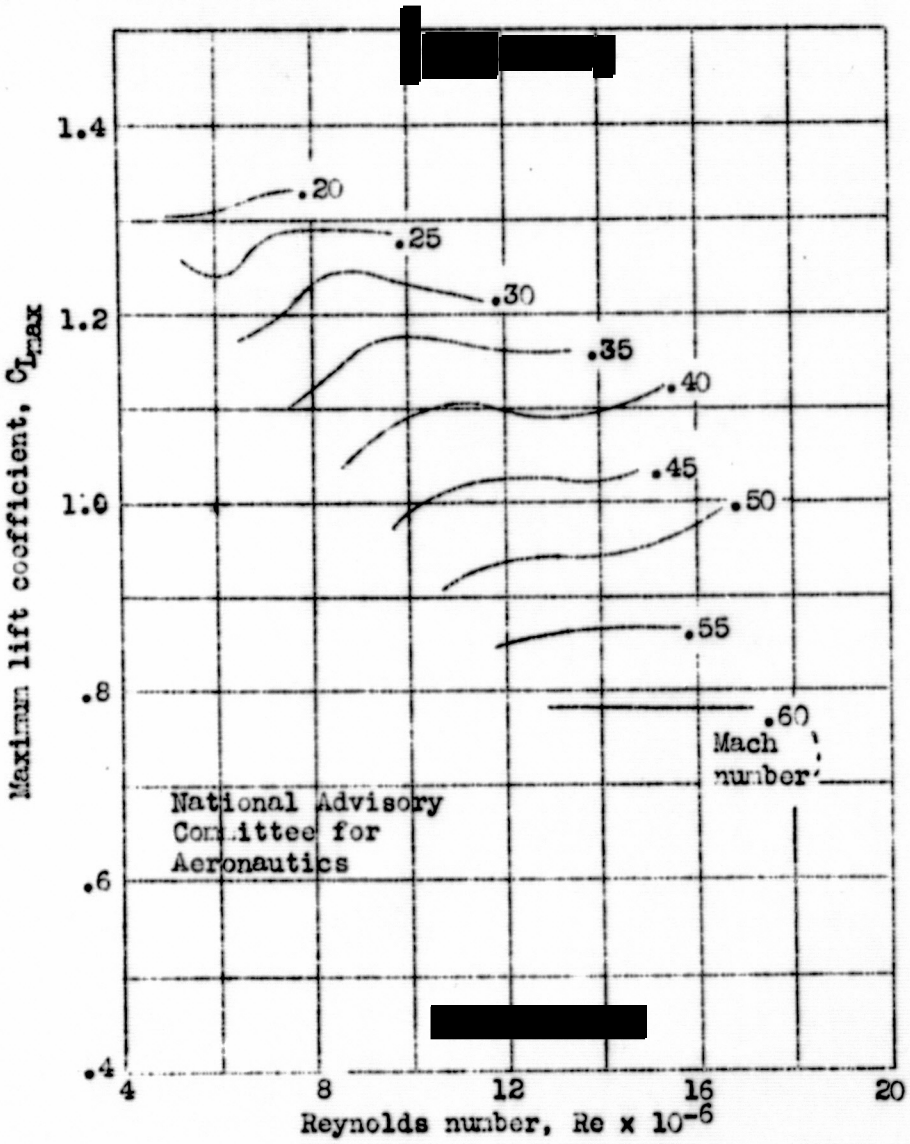


Figure 10.- Variation of maximum lift coefficient with Reynolds number for constant values of Mach number, power off, P-39N-1 airplane.

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