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EFFECTS OF COMPRESSIBILITY ON MAXIMUM LIFT COEFFICIENTS

FOR SIX PROPELLER AIRFOILS

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# NACA

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## ADVANCE CONFIDENTIAL REPORT

## EFFECTS OF COMPRESSIBILITY ON MAXIMUM LIFT COEFFICIENTS

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### SUMMARY

An extension of previously reported data on the variation of maximum lift coefficient with Mach number, camber, and thickness ratio is presented. The data were obtained from pressure-distribution tests in the Langley 8-foot high-speed tunnel of six propeller airfoils of 1-foot chord.

It was found that the maximum lift coefficients of all the airfoils were markedly affected by compressibility at Mach numbers as low as 0.2. At Mach numbers above the order of 0.45, large increases in maximum lift coefficient occurred. The combination of a thickness ratio of 0.15 and a design lift coefficient of 0.7 was found to be critical, with adverse effects on maximum lift coefficient occurring over most of the speed range investigated.

### INTRODUCTION

It has been pointed out in reference 1 that the prediction of high-lift performance of airfoils at high speeds based on low-speed data can be seriously in error. The low critical speeds occurring at high lifts and the separation produced by severe pressure gradients over the airfoil affect the maximum lift coefficient at Mach numbers as low as 0.2. It was also indicated that large increases in the maximum lift coefficient are to be expected at values of Mach number above 0.5. The data presented herein include an extension to higher Mach numbers of the data for the three airfoils presented in reference 1 as well as data for three additional airfoils tested to establish more definitely the effects of camber and thickness ratio.

The effects of compressibility on maximum lift coefficient as presented in reference 1 and herein have particular application to propeller performance at take-off and some climb conditions. These applications combine high-lift loadings on the blades with high speeds over a considerable portion of the blades.

The tests were conducted in the Langley 8-foot high-speed tunnel on models of 1-foot chord to obtain full-scale propeller Reynolds numbers and reduced tunnel-wall effects. Measurements consisted, principally, of the pressure distribution at the center of the airfoil model. This method of measurement gives effectively two-dimensional results, which best illustrate the type of phenomenon that occurs.

## SYMBOLS

$V$        air-stream velocity, feet per second

$a$        speed of sound in air, feet per second

$M$        air-stream Mach number  $(V/a)$

$R$        Reynolds number

$l$        section lift, pounds per foot of span

$c$        model chord, feet

$\rho$        air-stream mass density, slugs per cubic foot

$q$        air-stream dynamic pressure, pounds per square foot  $\left(\frac{1}{2}\rho V^2\right)$

$c_l$        section lift coefficient  $(l/qc)$

$c_{l_{max}}$        maximum section lift coefficient

$t$        model maximum thickness, feet

$p$        local static pressure, pounds per square foot

$p_0$        air-stream static pressure, pounds per square foot

P        pressure coefficient $\left(\dfrac{p - p_0}{q}\right)$

## APPARATUS AND METHODS

The Langley 8-foot high-speed tunnel is a single-return, circular-section, closed-throat tunnel. The airspeed is continuously controllable from about 75 to 550 miles per hour. The turbulence of the air stream, as indicated by transition measurements on airfoils, is unusually low but somewhat higher than that of free air.

Six models having NACA 16-209, 16-509, 16-709, 16-215, 16-515, and 16-715 airfoil sections of 1-foot chord were investigated. Thirty pressure orifices distributed along the chord were located at essentially the same spanwise station at the center of the air stream. The chordwise orifice locations and airfoil shapes are shown in figure 1. The airfoil ordinates were calculated by the methods described in reference 2.

The model, when mounted in the tunnel, completely spanned the jet (fig. 2). Except for auxiliary streamline-wire bracing, required by structural considerations, the standard mounting and setup for the Langley 8-foot high-speed tunnel were employed. Tests at low and medium speeds with and without braces indicated that interference of the auxiliary supports on the flow at the measurement station was negligible.

The surface pressure orifices in the airfoil were connected to a multiple-tube manometer located outside the test section. The pressure tubing connecting the orifices had a small diameter and was located within the airfoil. The pressures at all orifices were recorded simultaneously by photographing the multiple-tube manometer.

The Mach number range extended from 0.12 to 0.68 and the Reynolds number range, from $0.87 \times 10^6$ to $3.75 \times 10^6$. The variation of Reynolds number with Mach number is shown in figure 3.

## RESULTS

The maximum section lift coefficient $c_{l_{max}}$ was
determined as the highest value of lift coefficient in
the positive range of angle of attack. The values of
normal-force coofficient were obtained from integration
of the chordwise pressure distribution. Analysis has
shown that, up to the value of maximum lift coefficient,
the normal-force coefficient and the lift coefficient
are essentially the same.

The angles of attack at which the maximum lift
coefficient occurred for the various airfolls are presented
in table I. The variation of maximum section lift coef-
ficient with Mach number is given in figure 4. The small
arrowheads on the curves between Mach numbers of 0.40
and 0.50 indicate the points beyond which the critical
speed has been exceeded. Figure 5 shows the pressure
distribution over the NACA 16-215 airfoil for Mach numbers
of 0.25 and 0.40 at an angle of attack of 13°, which is
the stall angle at $M = 0.25$. Pressure distributions
over the NACA 16-515 airfoil are presented in figure 6
for Mach numbers of 0.40, 0.55, and 0.60 at an angle of
attack of 11°. Pressure distributions over the
NACA 16-715 airfoil are given in figure 7 for Mach numbers
of 0.33, 0.60, and 0.67 at an angle of attack of 10°.
These angles of attack, 11° and 10°, are near the angle
of stall and represent the phenomena that occur in the
neighborhood of maximum lift. The lower-surface pressure
distributions are presented only for the lowest values of
Mach number in figures 6 and 7 because little change
takes place over the lower surface in the range of
Mach number shown.

These data, as presented, have not been corrected
for wind-tunnel-wall interference. An analysis, however,
has been made of these effects according to the methods
of reference 3 which gave the following maximum corrections:

| | Mach number | Lift | Moment coefficient | Drag coefficient | Angle of attack |
|---|---|---|---|---|---|
| Design lift range | Less than 1 percent | Less than -1 percent | -0.002 | Less than 0.0001 | Less than $\pm 0.01°$ |
| High lift range | 1 percent | -2 percent | -0.002 | 0.001 | $\pm 0.05°$ |

The method of correction used has only qualitative application at high values of lift coefficient and supercritical values of Mach number. The corrections obtained, however, give a good estimate of the order of the interference effects.


DISCUSSION


It has been shown in references 1 and 4 that the maximum lift coefficient may be adversely affected at a Mach number as low as 0.20. Reference 1 shows that, when Mach number and Reynolds number were changed simultaneously, the maximum lift coefficient increased until a Mach number of 0.20 was reached. Further increase in Mach number led to adverse compressibility effects and separation phenomena. This effect, as has been shown in reference 1, prevented further rise in $c_{l_{max}}$ in the case of a thin airfoil and caused a pronounced reduction in $c_{l_{max}}$ in the case of a relatively thick airfoil. It is further indicated in reference 1 that increases in Mach number above 0.50 will lead to marked increases in airfoil maximum lift coefficients.

The data presented herein, which include an extension of the ranges of camber, thickness, and Mach number of reference 1, substantiate and extend the conclusions of reference 1.

The angles of attack for maximum lift (table I) are higher for the thicker airfoils than for the thinner airfoils and, in general, increase with camber for the thicker airfoils throughout the range of Mach number. For a given camber and thickness, the angle for maximum lift tends to decrease with increasing Mach number.

The results presented in figure 4 show that the
maximum section lift coefficient is a function of
Mach number, thickness ratio, and camber. The variation
of the maximum lift coefficient with Mach number for a
given camber and thickness generally shows an increase
to a Mach number of 0.25. Between Mach numbers of 0.25
and 0.50 virtually no further increase in the maximum
lift coefficient is found for the thinner airfoils and a
large decrease is found for the 15-percent-thick air-
foils. This general variation is not followed by the
NACA 16-209 airfoil. The nearly constant value of the
maximum lift coefficient up to a Mach number of 0.50 for
this airfoil is in accord with previous results, which
indicated that thin airfoils, because they have small
leading-edge radii, have effectively fixed degrees of
separation and separation points. Beyond a Mach number
of approximately 0.50 the maximum lift coefficient
increases sharply for all the airfoils and decreases
again only for the NACA 16-715 airfoil beyond a Mach
number of 0.60.

The general effect of increasing either thickness
or camber is to increase the value of the maximum lift
coefficient. The effect of increasing thickness on the
variation of $c_{l_{max}}$ with Mach number is to accentuate
the adverse effects in the region between Mach numbers
of 0.25 and 0.50.

The increase of maximum lift coefficient with Mach
number observed at low speeds up to a Mach number of 0.25
is similar to the variation with Reynolds number for
airfoils of medium thickness. Further increase in speed
above a Mach number of 0.25 leads to larger adverse
pressure gradients that induce greater thickening of the
boundary layer or separation, which reduce the circulation
around the airfoil. This effect is illustrated by the
pressure distribution over the NACA 16-215 airfoil
(fig. 5) at an angle of attack of 13°, which is the angle
of stall at a Mach number of 0.25 and is beyond the
angle of stall at a Mach number of 0.40. The pressures
near the trailing edge indicate the increase in separation
between Mach numbers of 0.25 and 0.40, and the pressures
over the rest of the airfoil show the loss of lift
accompanying these changes.

The large increases in maximum lift coefficient
beyond a Mach number of 0.50 (fig. 4) are due to the
rearward movement of the peak pressures over the airfoil
after the critical speed of the section has been reached
and strong compression shock has been established. (The
critical Mach number at maximum lift is 0.45 or below
for all these airfoils except the NACA 16-209 airfoil,
for which the critical Mach number is 0.50.) The effect
of these phenomena is illustrated in figures 6 and 7,
which show the increase in area under the pressure-
distribution curve that accompanies supercritical compressi-
bility effects as the Mach number is increased.

The failure of the NACA 16-715 airfoil to develop a
maximum lift coefficient much greater than that for the
NACA 16-515 airfoil in the Mach number range from
0.12 to 0.25 (fig. 4) is ascribed to a critical combina-
tion of camber and thickness which leads to extreme
adverse pressure-recovery gradients over the rear portion
of the airfoil. These adverse pressure gradients result
in separation at lower lift coefficients than might be
expected for an airfoil of this camber. The angles of
attack for maximum lift (table I) indicate that for an
increase of camber from 0.5 to 0.7 for airfoils having
a thickness ratio of 0.15, there is no increase in angle
of attack at Mach numbers less than 0.33. Between Mach
numbers of 0.33 and 0.48, the angle of maximum lift for
the NACA 16-715 airfoil is higher than that for the
NACA 16-515 airfoil and thus indicates an improvement of
the flow. This improvement is illustrated in figure 4
in the Mach number region of 0.35, where the increment
of maximum section lift coefficient between the NACA 16-515
and 16-715 airfoils is of the same order as that between
the NACA 16-509 and 16-709 airfoils. The critical nature
of the NACA 16-715 airfoils is further indicated by the
loss of maximum section lift coefficient that occurs
beyond a Mach number of 0.60. This loss is principally
a separation phenomenon and is illustrated by the pressure
distribution at a Mach number of 0.67 shown in figure 7.

The increase of maximum lift coefficient with camber
results from the fact that at an angle of attack of 0°
the airfoils having higher camber develop higher lift
coefficients, have essentially the same lift-curve slope,
and (as illustrated in table I) stall at about the same
angle of attack. The effect on airfoil characteristics
of increasing thickness is to increase the value of the

stall angle. A further effect of thickness is to change
the type of flow over the leading edge, as pointed out
in reference 5.

It was pointed out in reference 4 that some improve-
ment in propeller take-off thrust may be effected in
marginal designs through utilization of the increase in
maximum lift obtained from increased design camber.
Because it has been indicated that small changes in
camber have little effect on the critical speeds of these
sections when operating in the design range, small
increases in camber will not materially affect the high-
speed operation.

The large increases in maximum section lift coef-
ficient in the high Mach number range above 0.50 may be
utilized for increased take-off thrust because a large
portion of the propeller blade operates at relatively
high values of Mach number even at low forward speeds.
Eventual losses are to be expected, however, at Mach
numbers higher than the range of the present tests, as
indicated by the results for the NACA 16-715 airfoil.

These results for the NACA 16-715 airfoil are
significant when contrasted with the results for the
NACA 16-709 and 16-215 airfoils throughout the range of
Mach number tested. The results indicate that the
combination of a thickness ratio of 0.15 and a design
$c_l$ of 0.7 is unconservative, and special care must be
taken in the selection of sections in or beyond this
critical range of combinations of thickness ratio and
design lift coefficient.

The results of those tests are believed to be common,
at least qualitatively, to all airfoils in current use
because in the stall region all airfoils exhibit a
characteristic pressure peak near the leading edge and,
hence, have qualitatively the same type of flow pattern.

## CONCLUSIONS

The pressure-distribution measurements of six
NACA 16-series airfoils made to investigate the effects
of compressibility on the maximum section lift coef-
ficients indicated that:

1. The maximum lift coefficient of airfoils was affected by compressibility at Mach numbers as low as 0.20.

2. At high Mach numbers large increases in the maximum lift coefficient occurred.

3. Increase in camber resulted in increases in the maximum lift coefficient.

4. Increase in thickness caused an increase in the maximum lift coefficient and accentuated the compressibility effects.

5. The combination of a thickness ratio of 0.15 and a design lift coefficient of 0.7 resulted in an unconservative section with losses in maximum lift coefficient at high Mach numbers. Special care must be exercised in the selection of sections in this critical range of combinations of thickness ratio and design lift coefficient.

6. For propeller designs that are marginal in take-off, improvement in take-off thrust may be realized by increase in design camber.

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                Langley Field, Va.

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     pressibility on the Maximum Lift Coefficient.
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     pressibility.  NACA ACR No. L4E17, 1944.

TABLE I

ANGLES OF ATTACK FOR MAXIMUM LIFT COEFFICIENT

| NACA airfoil section / Mach number | Angle of attack (deg) | | | | | |
|---|---|---|---|---|---|---|
| | 16-209 | 16-509 | 16-709 | 16-215 | 16-515 | 16-715 |
| 0.12 | 10.0 | 10.0 | 11.0 | 12.0 | 15.0 | 15.0 |
| .20 | 10.0 | 9.5 | 10.5 | 12.5 | 15.5 | 15.0 |
| .25 | 9.0 | 8.5 | 10.0 | 13.0 | 15.0 | 15.0 |
| .33 | 10.0 | 9.0 | 9.5 | 12.0 | 14.0 | 15.0 |
| .40 | 10.0 | 8.5 | 9.0 | 11.0 | 13.0 | 14.0 |
| .48 | 9.0 | 9.0 | 9.0 | 11.0 | 12.0 | 14.0 |
| .53 | 9.0 | 8.0 | 9.0 | 10.5 | 12.0 | 11.0 |
| .60 | 8.0 | 9.0 | 9.0 | 9.5 | ------ | 11.0 |
| .64 | ------ | ------ | 8.0 | 8.0 | ------ | 10.0 |
| .68 | ------ | ------ | ------ | ------ | ------ | 8.0 |

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NACA 16-709

NACA 16-715
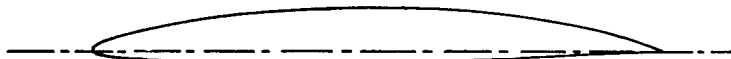
NACA 16-509
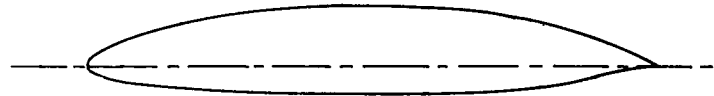
NACA 16-515

Orifices
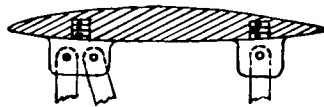
NACA 16-209

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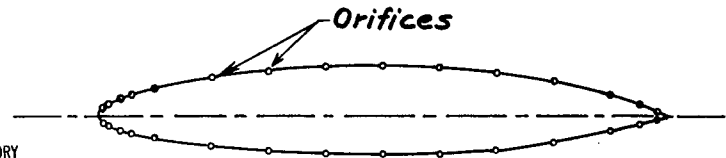
NACA 16-215

*Figure 1.- Airfoil profiles tested and distribution of pressure orifices.*

Fig. 1

Spanwise location
of orifices

Streamline-
wire bracing
⅝" diam.
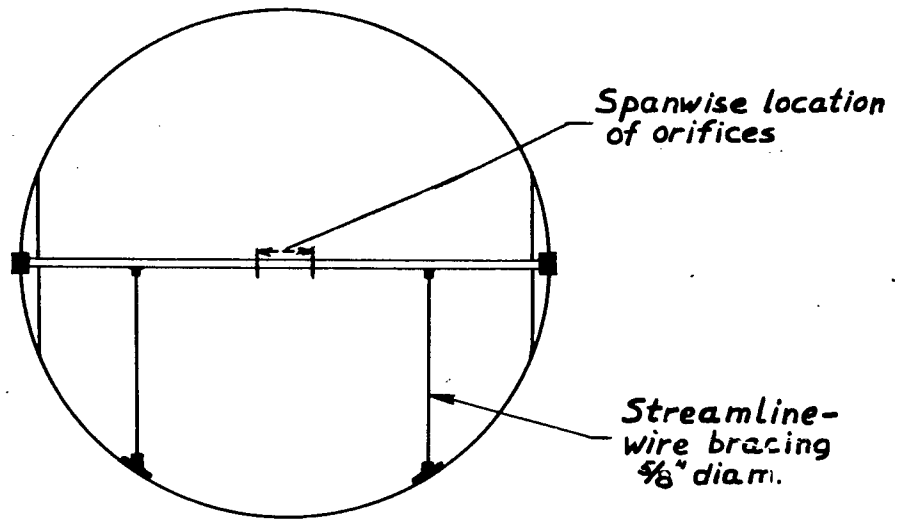
(a) Airfoil mounting in tunnel.

Streamline
wire
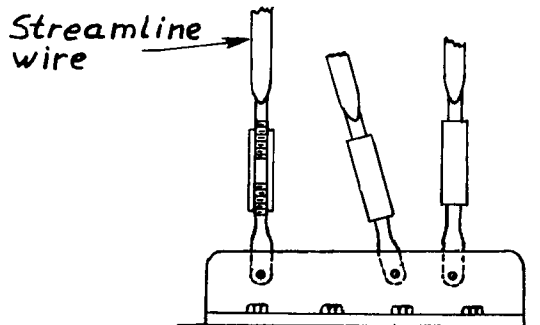
(b) Detail of bracing.

Figure 2.— Diagrammatic sketch of airfoil mounting
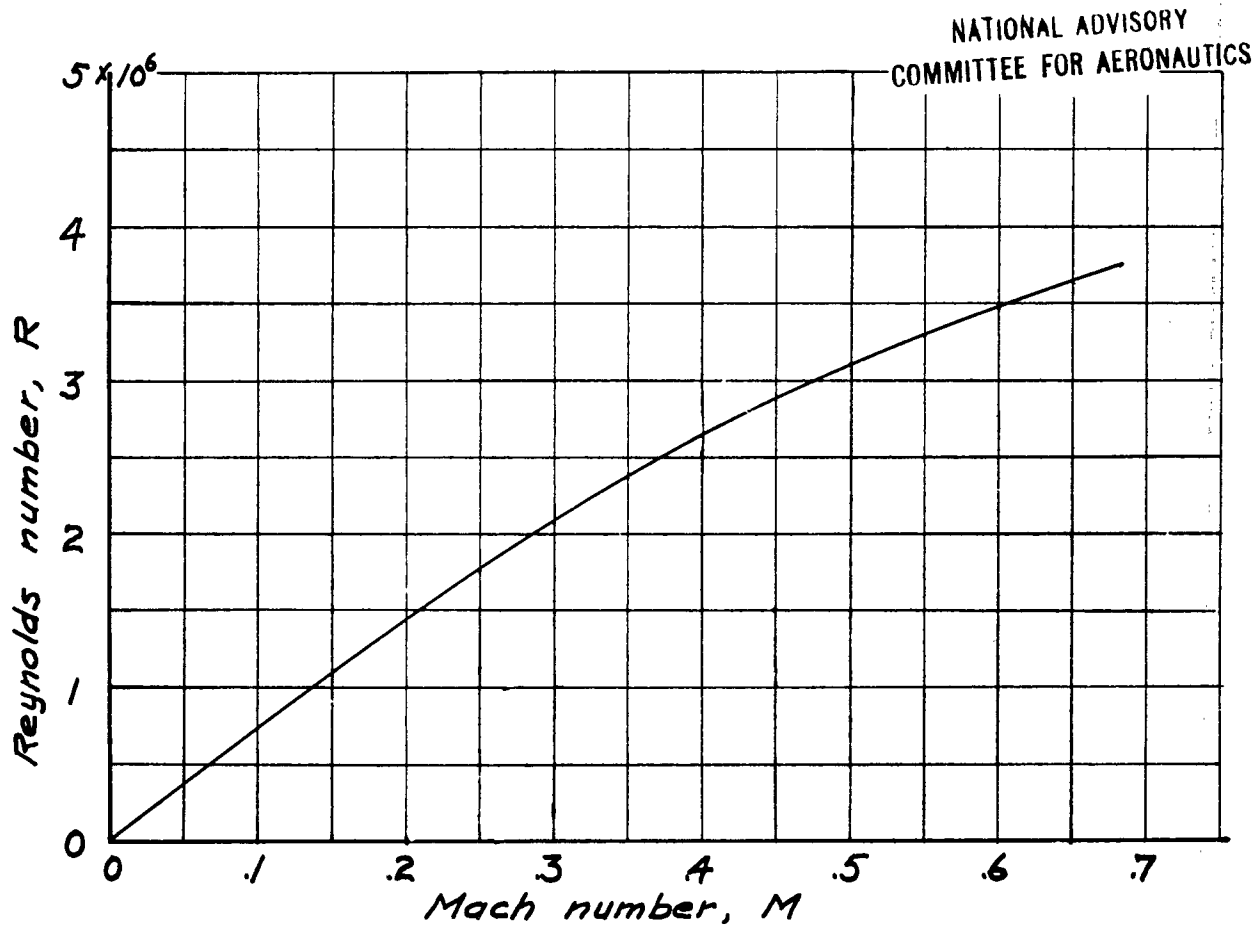and bracing in the Langley 8-foot high-speed tunnel.

Figure 3 .— Variation of Reynolds number (approx.) with
Mach number for 1-foot-chord airfoils in the Langley
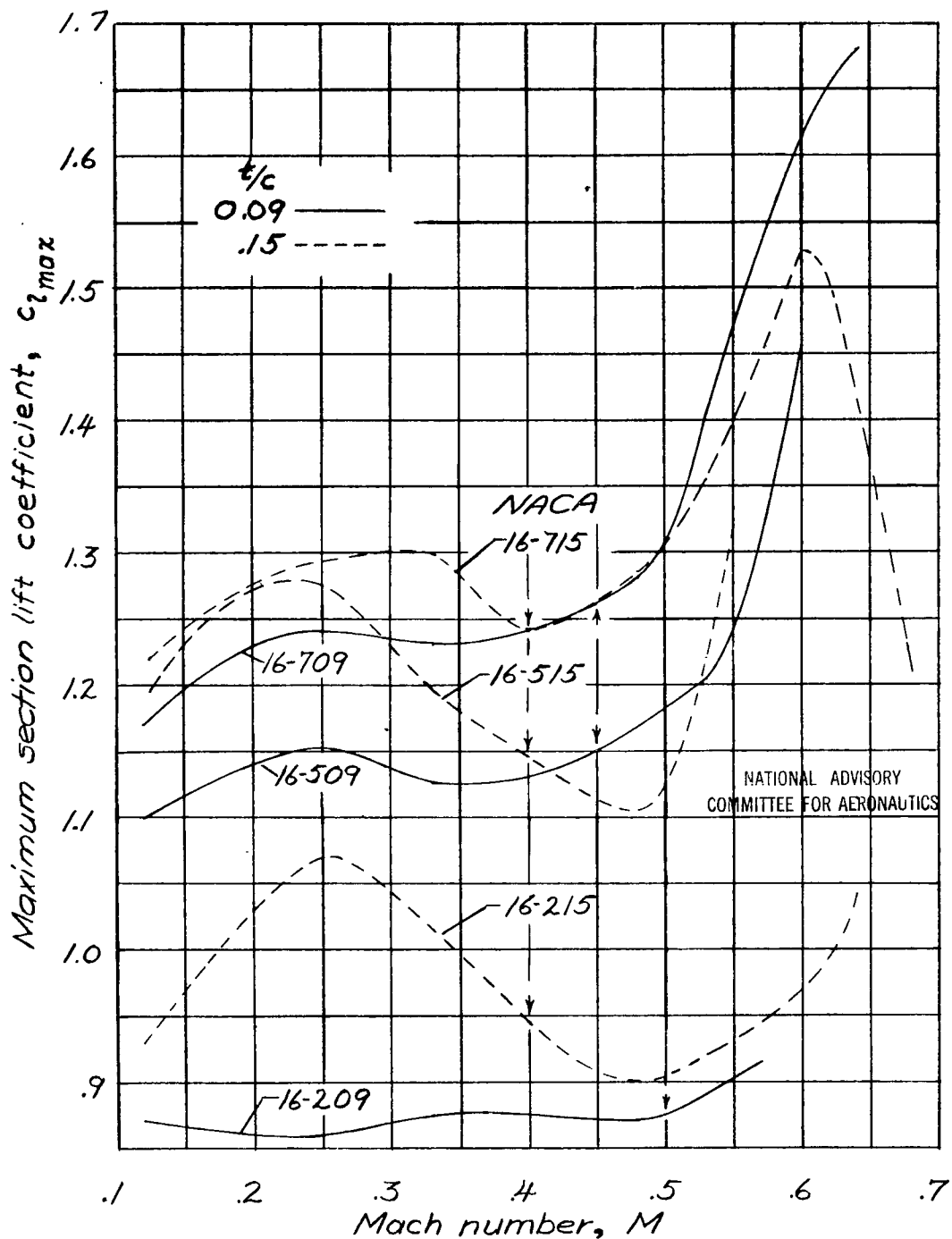8-foot high-speed tunnel.

Fig. 3

Figure 4.— Variation of maximum section lift coefficient with Mach number for six 1-foot-chord NACA 16-series airfoils.
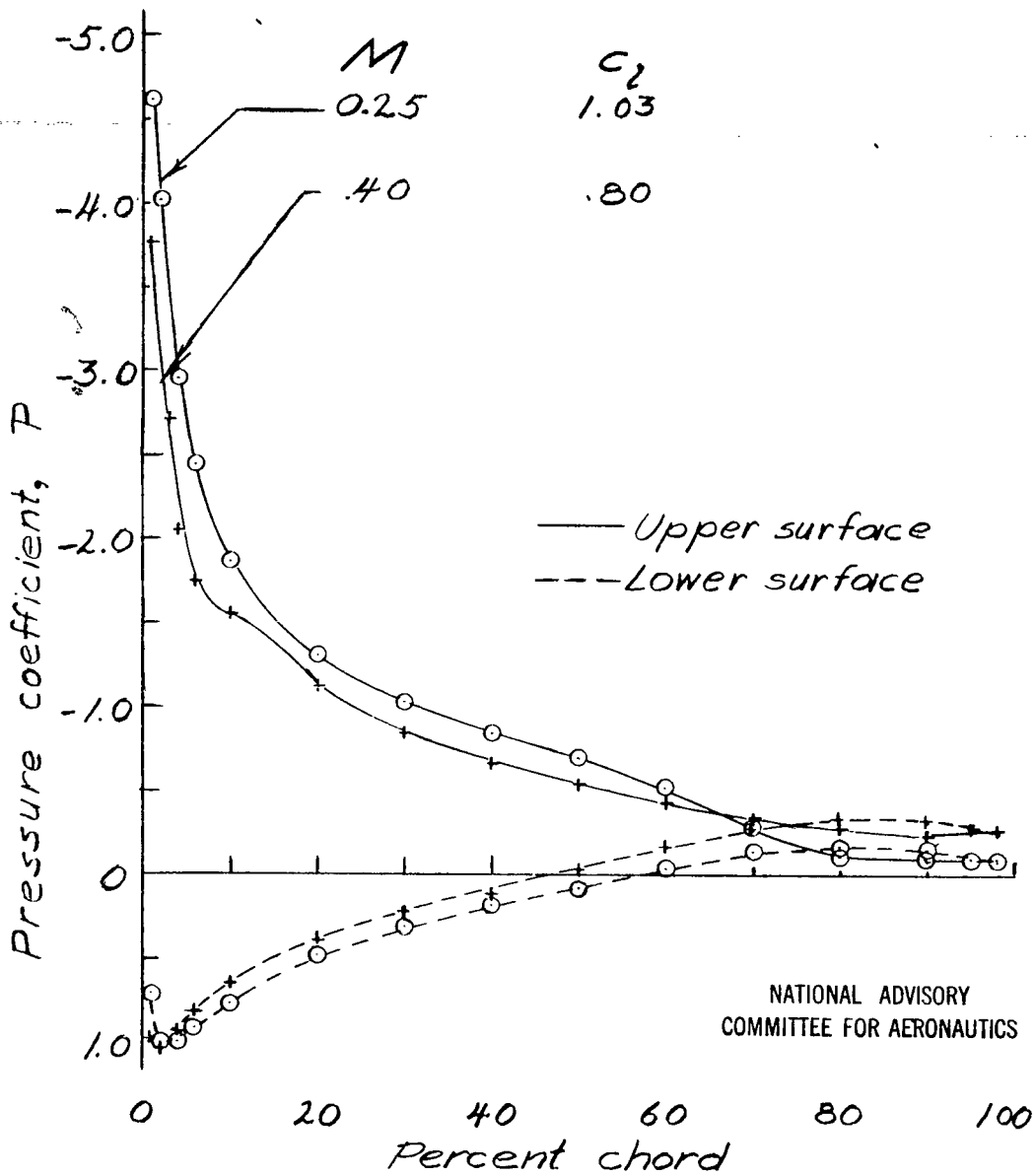
Figure 5.— Pressure distribution over the
NACA 16-215 airfoil at an angle of attack
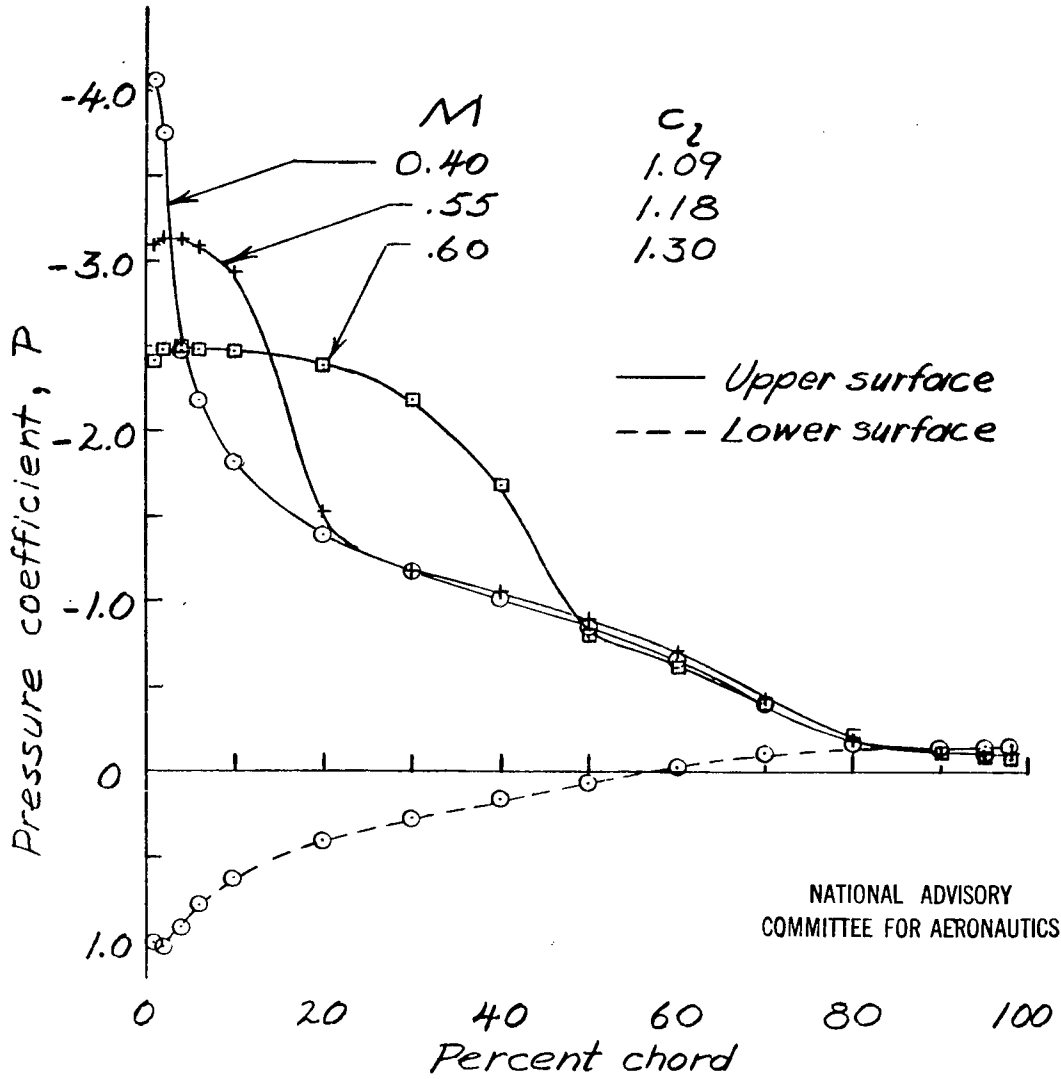of 13° at Mach numbers of 0.25 and 0.40.

Figure 6.- Pressure distribution over the NACA 16-515 airfoil at an angle of attack of 11° at Mach numbers of 0.40, 0.55, and 0.60.
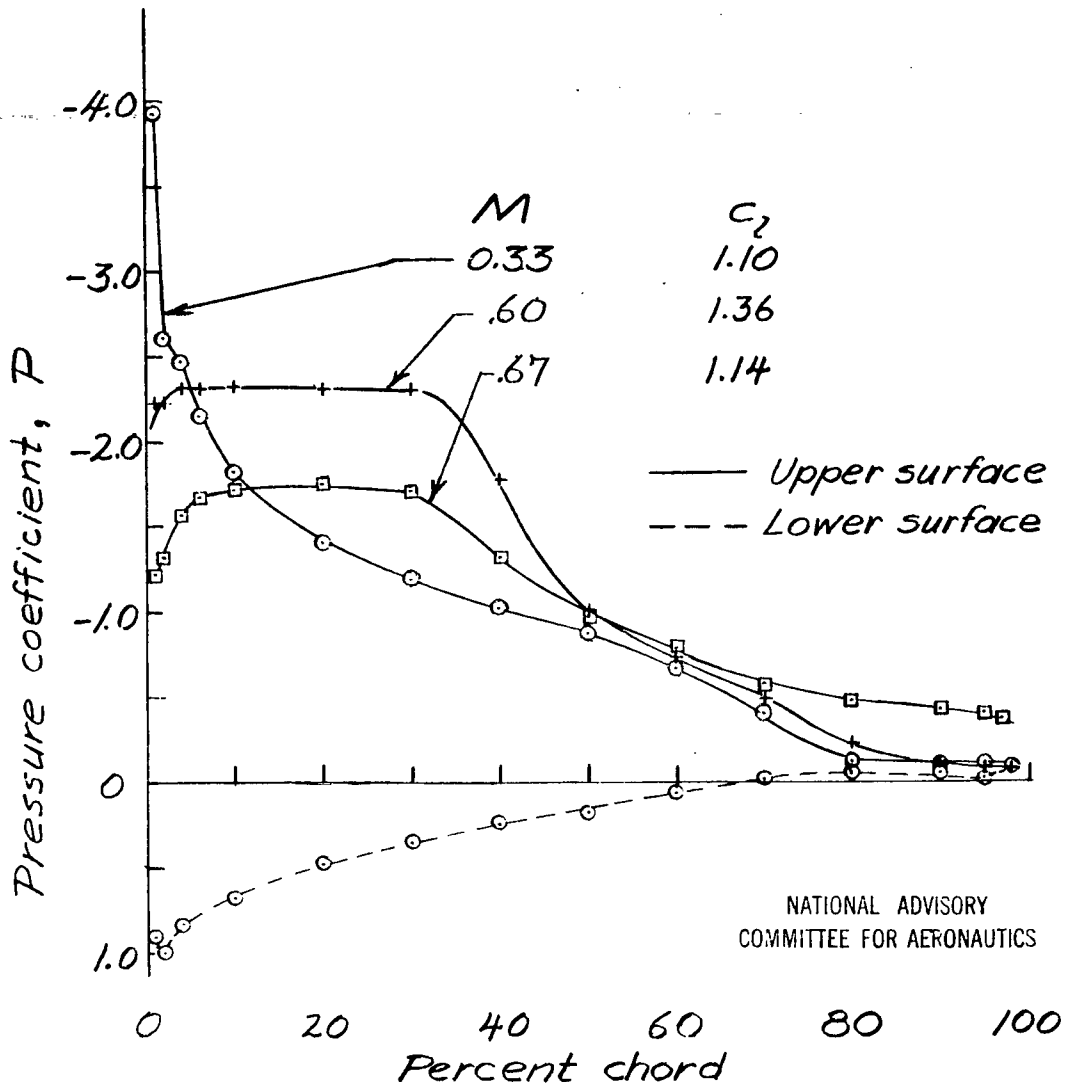
Figure 7.– Pressure distribution over the NACA 16-715 airfoil at an angle of attack of 10° at Mach numbers of 0.33, 0.60, and 0.67.