Two-Dimensional Aerodynamic Characteristics of the OLS/TAAT Airfoil

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SYMBOLS

\(C_p\) static-pressure coefficient, \(C_p = \frac{p - p_\infty}{(1/2)\rho V_\infty^2}\)

\(c\) airfoil chord, in.

\(c_d\) section profile-drag coefficient, \(c_d = \sum_{\text{wake}} c'_d \frac{\Delta h}{c}\)

\(c'_d\) point-drag coefficient,

\[
c'_d = 2 \left( \frac{P}{P_\infty} \right)^{s/7} \left[ \frac{(p/p)^{2/7} - 1}{(p_{L}/p_\infty)^{2/7} - 1} \right]^{\frac{1}{17}} \left\{ \left( \frac{P_{L}}{P_{L,\infty}} \right)^{1/7} - \left[ \frac{(p_{L}/p_\infty)^{2/7} - 1}{(p_{L,\infty}/p_\infty)^{2/7} - 1} \right]^{1/2} \right\}
\]

\(c_l\) section lift coefficient, \(c_l = c_n \cos(\alpha) - (c_d - c_n \sin(\alpha))\tan(\alpha)\)

\(c_m\) section pitching-moment coefficient about quarter-chord

\(c_n\) section normal-force coefficient, \(c_n = \sum P_r (\Delta x/c) + \sum P_r (\Delta x/c)_{\text{u.s.}}\)

\(h\) height of wake-survey probe tubes from given reference plane, in.

\(M\) Mach number

\(P\) pressure, psi

\(p\) static pressure, psi

\(Q\) dynamic pressure, psi

\(R_n\) Reynolds number based on airfoil chord and free-stream conditions

\(t\) airfoil thickness, in.

\(v\) velocity, ft/sec

\(x\) airfoil abscissa, in.

\(z\) airfoil ordinate, in.

\(\alpha_c\) angle of attack corrected for lift-interference effects, \(\deg\)

\(\rho\) density, slugs/ft\(^3\)

Subscripts:

\(t\) total

\(\infty\) free stream

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TWO-DIMENSIONAL AERODYNAMIC CHARACTERISTICS OF THE OLS/TAAT AIRFOIL

Michael E. Watts, Jeffrey L. Cross, and Kevin W. Noonan*

Ames Research Center

SUMMARY

Two flight tests have been conducted that obtained extensive pressure data on a modified AH-1G rotor system. These two tests, the Operational Loads Survey (OLS) and the Tip Aerodynamics and Acoustics Test (TAAT) used the same rotor set. In the analysis of these data bases, accurate two-dimensional airfoil data is invaluable, for not only does it allow comparison studies between two- and three-dimensional flow, but it also provides accurate tables of the airfoil characteristics for use in comprehensive rotorcraft analysis codes. To provide this two-dimensional data base, a model of the OLS/TAAT airfoil was tested over a Reynolds number range from $3 \times 10^6$ to $7 \times 10^6$ and between Mach numbers of 0.34 to 0.88 in the NASA Langley Research Center's 6- by 28-Inch Transonic Tunnel. The two-dimensional airfoil data is presented as chordwise pressure coefficient plots, as well as lift, drag, and pitching-moment coefficient plots and tables.

INTRODUCTION

Two flight tests have been conducted that obtained extensive pressure data on a modified AH-1G rotor system. These two tests, the Operational Loads Survey (OLS) and the Tip Aerodynamics and Acoustics Test (TAAT) used the same rotor set. In the analysis of these data bases, accurate two-dimensional airfoil data is invaluable, for not only does it allow comparison studies between two- and three-dimensional flow, but it also provides accurate tables of the airfoil characteristics for use in comprehensive rotorcraft analysis codes. To provide this two-dimensional data base, a model of the OLS/TAAT airfoil was tested over a Reynolds number range from $3 \times 10^6$ to $7 \times 10^6$ and between Mach numbers of 0.34 to 0.88 in the NASA Langley Research Center's 6- by 28-Inch Transonic Tunnel. The two-dimensional airfoil data is presented as chordwise pressure coefficient plots, as well as lift, drag, and pitching-moment coefficient plots and tables.

TEST FACILITY

This test was conducted at the Langley Research Center's 6- by 28-Inch Transonic Tunnel. The pertinent tunnel specifications, obtained from references 1 and 2, are presented in table 1, and drawings of the tunnel are presented in figures 1 and 2. The wind tunnel has solid sidewalls with a slotted floor and ceiling. The slots consist of four longitudinal half slots along the junction of the tunnel's floor, ceiling, and sidewalls. The slots provide the tunnel with a 5% openness ratio with the flow that passes through the slots reentering the main flow aft of the model. This bypass process is controlled by a set of flaps located 12.75 in. downstream from the test section center. The rectangular test section extends from 38 in. upstream of the center to 44.25 in. downstream. The sidewalls of the test section house 9.5-in. diam turntables that support the model. The tunnel gets its supply of dry, compressed air from the 26-in. Transonic Blowdown Tunnel situated below the 6- by 28-Inch Tunnel.

The tunnel has four primary control systems that regulate Mach number, stagnation pressure, model angle of attack, and flow control. The tunnel Mach number is controlled by two sliding choker doors located downstream of the test section. The stagnation pressure is controlled with a valve control system that allows for constant Reynolds number testing. The turntables are controlled, as are the primary controls, by a closed-loop servovalve system. The flap system, mentioned earlier, is manually operated using an electrical drive system.

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TEST SET-UP AND INSTRUMENTATION

The wind tunnel instrumentation system has the capability of measuring a total of 64 channels of pressure measurements. These pressure measurements are broken into four primary groupings of measurements. The first set consists of model static pressure orifices and are generally aligned in a chordwise array so as to measure the static pressure distribution characteristics of that airfoil. The model static pressure measurements are used in the calculations of the section lift and pitching-moment coefficients. The second instrumentation set is a traversing survey probe which measures the pressure in the flow aft of the model. The airfoil drag measurements are obtained with the use of the pressures measured by this traversing rake (fig. 3). The rake can be traversed from 10 in. above to 10 in. below tunnel centerline. The computation of drag requires the use of static pressure measured downstream of the model by the third set of instrumentation. A fourth set of instrumentation is used to measure free-stream total and static pressures.

The model tested was a symmetrical airfoil known as the OLS/TAAT airfoil, which is a slightly modified Bell 540 airfoil. The 540 was the standard airfoil used on the rotor of the AH-1 series of helicopters. The OLS/TAAT airfoil was obtained by adding a 0.1-in.-thick glove to the basic 540, and extending the trailing edge to obtain a blade thickness ratio of 9.71%. The full-scale airfoil coordinates are presented in appendix A. The 6-in. chord model was constructed out of stainless steel, and contained a pressure tap at the leading edge with 22 pressure tap locations on each of the top and bottom surface for a total of 45 ports. The exact chordwise pressure tap locations are presented in table II. The model was constructed to contour tolerances of \( \pm 5.0 \times 10^{-4} \) in.

PRESENTATION OF RESULTS

Table III presents the test conditions that were obtained during this investigation arranged by Mach and corrected angle of attack. To produce the airfoil sectional normal force and pitching moment coefficients, the pressure coefficient data have been integrated about the airfoil, using a spline integration routine. The airfoil sectional drag coefficients have been calculated from the rake array data using the standard 6- by 28-Inch Tunnel data processing computational procedure. The lift coefficient has been obtained from the drag and normal force coefficients. The nondimensional aerodynamic coefficient results are presented in the figures called out in table IV and the tabular results are presented in table V. The tabular results are grouped by Mach number and angle of attack. Through each angle of attack sweep the Mach number and Reynolds number shifted slightly; however, the values given in these figures are the nominal values during each sweep.

TWO DIMENSIONALITY OF FLOW

The results of a previous investigation of rotorcraft airfoils in the Langley Research Center’s 6- by 28-Inch Transonic Tunnel (ref. 3) have shown that the indicated maximum normal-force coefficient is reduced by tunnel-wall boundary-layer influences. This reduction is characteristic of two-dimensional wind tunnels without proper sidewall boundary-layer control and occurs because the tunnel-wall boundary layer is thicker than that of the airfoil; therefore, initial separation begins at the tunnel wall.

Although it is not possible to determine precisely the affected Mach number range or the loss in maximum normal-force coefficient of the airfoils reported herein, a comparison of the NACA 0012 data measured in this facility with 0.05 open slots with unpublished data from two other facilities has been useful in indicating the magnitude of these losses. The maximum normal-force coefficients measured in the Langley Research Center’s Low-Turbulence Pressure Tunnel and the United Technologies Research Center 8-Foot Subsonic/Transonic Wind Tunnel at similar Reynolds numbers and at a Mach number of 0.36 are higher than that from the Langley Research Center’s 6- by 28-Inch Transonic Tunnel by about 0.15. The difference between the data from the Langley Research Center’s 6- by 28-Inch Transonic Tunnel and the United Technologies Research Center data decreased to 0.05 at a Mach number of about 0.53. Incremental values for other airfoils may vary slightly because of specific configuration influences.

An investigation conducted in the Office National d’Études et de Recherches Aérospatiales (ONERA) R1 Ch wind tunnel (ref. 4) has shown that the tunnel sidewall boundary layer can affect the normal-force coefficients at all angles of attack (that is, with either attached or separated boundary layers). In this investigation, the sidewall boundary layer thickness was varied by applying sidewall suction upstream of the model while the Mach number and Reynolds number were held constant. Generally, an increase in sidewall boundary layer thickness resulted in a decrease in the normal-force coefficient at a given angle of attack; the trend reversed at Mach numbers greater than 0.85 with a supercritical airfoil.

CORRECTION FOR LIFT INTERFERENCE

The corrections for lift interference, which have been applied to the angles of attack, were obtained from
The basic equations for the correction are

\[ \alpha_c = \alpha + \Delta \alpha \]

where

\[ \Delta \alpha = -c_n \frac{c}{8} \left( \frac{c}{36.195} \right) \left( \frac{1}{k+1} \right) \left( \frac{180}{\pi} \right) \]

\[ k = \frac{a}{h} K \]

where \( a \) is the slot spacing and \( h \) is the semihighet of the tunnel. The slotted-wall boundary condition coefficient \( k \) for the present tunnel configuration is 0.4211K. A value of 3.5 was selected for the slotted-wall performance coefficient \( K \), based on the data and discussion presented in reference 6. These substitutions yield a correction given by the equation

\[ \Delta \alpha = -c_n c(0.0800) \]

where \( c \) is in centimeters, \( \alpha \) is in degrees, and the constant of 0.08 is in degrees per centimeter.

**CONCLUSION**

An investigation was conducted in the Langley Research Center's 6- by 28-Inch Transonic Wind Tunnel to determine the two-dimensional aerodynamic characteristics of the OLS/TAAT airfoil. The tests were conducted at Reynolds numbers of \( 3.0 \times 10^6 \) to \( 7.0 \times 10^6 \), typical of full scale \( R_n \) at a Mach number range from 0.34 to 0.88. The results of this investigation are presented as both chordwise-pressure coefficient plots, and integrated lift, drag and pitching-moment coefficient plots.
APPENDIX A

Appendix A contains the upper surface ordinates of the OLS/TAAT full scale airfoil in inches. The $x$ dimension is distance along the chord mean line with 0.0 being the leading edge and $z$ is the distance from the mean line. The airfoil is symmetrical, therefore the lower surface ordinates are the negative of the $z$ value.

**TABLE A1. OLS/TAAT FULL SCALE AIRFOIL COORDINATES**

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REFERENCES


3. Noonan, Kevin W.; and Bingham, Gene J.: Two-Dimensional Aerodynamic Characteristics of Several Rotorcraft Airfoils at Mach Numbers From 0.35 to 0.90. NASA TM X-73990, 1977.


TABLE I.– LANGLEY 6- BY 28-INCH TRANSONIC WIND TUNNEL CHARACTERISTICS

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<tr>
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<tr>
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<td>-0.5507</td>
<td>-0.5477</td>
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<tr>
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<td>-1.65</td>
<td>-0.3637</td>
<td>-0.3626</td>
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<tr>
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<td>1.64</td>
<td>0.3856</td>
<td>0.3841</td>
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<tr>
<td>0.873</td>
<td>2.54</td>
<td>0.5222</td>
<td>0.5196</td>
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Figure 2 – Cross section drawing of side view of Langley Research Center's 6- by 28-Inch Transonic Tunnel (in.)
Figure 3.— Details of multitube probe used on traversing survey mechanism (in.).
Figure 4.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.34$; (a) $\alpha_c = -3.51^\circ$, (b) $\alpha_c = -1.78^\circ$. 

\[
\begin{align*}
\text{RUN 2} & \quad \text{POINT 15} & \quad \text{MACH} = 0.338 & \quad R_n = 4.70 \times 10^6 \\
& & \quad c_f = -0.4594 & \quad c_d = 0.0077 \\
& & \quad c_m = 0.0034
\end{align*}
\]

○ UPPER SURFACE
○ LOWER SURFACE

\[
\begin{align*}
\text{RUN 2} & \quad \text{POINT 14} & \quad \text{MACH} = 0.339 & \quad R_n = 4.63 \times 10^6 \\
& & \quad c_f = -0.2261 & \quad c_d = 0.0077 \\
& & \quad c_m = 0.0033
\end{align*}
\]

○ UPPER SURFACE
○ LOWER SURFACE
RUN 1
POINT 2
MACH = 0.346
$R_n = 4.63 \times 10^6$
$c_l = 0.0001$
$c_d = 0.0072$
$c_m = 0.0037$

RUN 19
POINT 135
MACH = 0.332
$R_n = 4.68 \times 10^6$
$c_l = -0.0013$
$c_d = 0.0067$
$c_m = 0.0030$

Figure 4.—Continued; (c) $\alpha_c = 0.02^\circ$, (d) $\alpha_c = 0.02^\circ$. 
Figure 4.—Continued; (e) $\alpha_c = 1.87^\circ$, (f) $\alpha_c = 1.87^\circ$. 

RUN 19  
POINT 136  
MACH = 0.345  
$c_l = 0.2397$  
$c_d = 0.0074$  
$c_m = 0.0039$  
$R_n = 4.81 \times 10^6$  

RUN 1  
POINT 3  
MACH = 0.336  
$c_l = 0.2405$  
$c_d = 0.0073$  
$c_m = 0.0037$  
$R_n = 4.57 \times 10^6$
RUN 19  POINT 137  MACH = 0.343  \( R_n = 4.71 \times 10^6 \)
\( c_l = 0.4674 \)  \( c_d = 0.0080 \)  \( c_m = 0.0039 \)

RUN 19  POINT 138  MACH = 0.345  \( R_n = 4.76 \times 10^6 \)
\( c_l = 0.6962 \)  \( c_d = 0.0089 \)  \( c_m = 0.0052 \)

Figure 4.: Continued; (g) \( \alpha_c = 3.61^\circ \), (h) \( \alpha_c = 5.36^\circ \).
Figure 4.— Continued; (i) $\alpha_C = 7.13^\circ$, (j) $\alpha_C = 7.16^\circ$. 

RUN 1
POINT 6
MACH = 0.339
$C_f = 0.9172$
$C_d = 0.0085$
$C_m = 0.0061$

RUN 19
POINT 139
MACH = 0.339
$C_f = 0.9264$
$C_d = 0.0107$
$C_m = 0.0066$
Figure 4.— Continued; (k) $\alpha_c = 8.06^\circ$, (l) $\alpha_c = 8.95^\circ$. 
Figure 4.— Continued; (m) $\alpha_c = 9.92^\circ$, (n) $\alpha_c = 10.90^\circ$. 
RUN 1  
POINT 11  
MACH = 0.340  
$c_l = 1.1594$  
$c_d = 0.1006$  
$c_m = -0.0691$  
$R_n = 4.75 \times 10^6$  

Figure 4.— Concluded; (o) $\alpha_c = 11.97^\circ$.  

- UPPER SURFACE  
- LOWER SURFACE
Figure 5.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.39$; (a) $\alpha_c = -3.51^\circ$, (b) $\alpha_c = -1.68^\circ$. 
Figure 5.—Continued; (c) $\alpha_c = -0.05^\circ$, (d) $\alpha_c = 0.00^\circ$. 

RUN 4  POINT 28  MACH = 0.390  $R_n = 5.19 \times 10^6$
$c_l = -0.0041$  $c_d = 0.0068$  $c_m = 0.0026$

RUN 3  POINT 18  MACH = 0.394  $R_n = 5.24 \times 10^6$
$c_l = 0.0049$  $c_d = 0.0070$  $c_m = 0.0026$
RUN 3
POINT 19
MACH = 0.389
$R_n = 5.31 \times 10^6$
$c_l = 0.2212$
$c_d = 0.0066$
$c_m = 0.0029$

RUN 3
POINT 20
MACH = 0.391
$R_n = 5.39 \times 10^6$
$c_l = 0.4490$
$c_d = 0.0080$
$c_m = 0.0036$

Figure 5.— Continued; (e) $\alpha_c = 1.69^\circ$, (f) $\alpha_c = 3.43^\circ$. 
Figure 5.— Continued; (g) $\alpha_c = 5.24^\circ$, (h) $\alpha_c = 7.05^\circ$. 
Figure 5.— Continued; (i) $\alpha_c = 7.94^\circ$, (j) $\alpha_c = 8.84^\circ$. 
RUN 3  POINT 25  MACH = 0.391  R_n = 5.73 \times 10^6
\begin{align*}
c_l &= 1.0865 \\
c_d &= 0.0454 \\
c_m &= -0.0034
\end{align*}

Figure 5.— Concluded; (k) \( \alpha_c = 9.99^\circ \).
Figure 6.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.44$; (a) $\alpha_c = -3.43^\circ$, (b) $\alpha_c = -1.70^\circ$. 
Figure 6.— Continued; (c) $\alpha_c = 0.00^\circ$, (d) $\alpha_c = 1.70^\circ$. 
RUN 5  POINT 34  MACH = 0.437  $R_n = 5.85 \times 10^6$
$c_l = 0.4625$ $c_d = 0.0073$ $c_m = 0.0034$

- UPPER SURFACE
- LOWER SURFACE

RUN 5  POINT 35  MACH = 0.438  $R_n = 5.94 \times 10^6$
$c_l = 0.6939$ $c_d = 0.0074$ $c_m = 0.0061$

- UPPER SURFACE
- LOWER SURFACE

Figure 6.—Continued; (e) $\alpha_c = 3.53^\circ$, (f) $\alpha_c = 5.31^\circ$. 
Figure 6.— Continued; (g) $\alpha_c = 7.19^\circ$, (h) $\alpha_c = 8.07^\circ$.  

RUN 5  
POINT 36  
MACH = 0.440  
$R_n = 5.98 \times 10^6$  
$c_l = 0.9252$  
c_d = 0.0120  
c_m = 0.0113

RUN 5  
POINT 38  
MACH = 0.440  
$R_n = 6.10 \times 10^6$  
$c_l = 0.9788$  
c_d = 0.0254  
c_m = 0.0123
RUN 5
POINT 39  MACH = 0.435
$c_l = 1.0072$  $c_d = 0.0373$
$R_n = 6.12 \times 10^6$  $c_m = 0.0028$

\[\text{UPPER SURFACE} \quad \text{LOWER SURFACE}\]

Figure 6.- Continued; (i) $\alpha_c = 8.94^\circ$, (j) $\alpha_c = 9.95^\circ$. 
RUN 5
POINT 41
$c_f = 1.0343$
$c_d = 0.1066$
$c_m = -0.0423$

$MACH = 0.430$
$R_n = 6.18 \times 10^6$

Figure 6.— Concluded; (k) $\alpha_c = 11.01^\circ$. 
Figure 7.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.49$; (a) $\alpha_c = -3.52^\circ$, (b) $\alpha_c = -3.50^\circ$. 
Figure 7.—Continued; (c) $\alpha_c = -1.78^\circ$, (d) $\alpha_c = -1.71^\circ$. 
Figure 7. Continued; (e) $\alpha_c = 0.01^\circ$, (f) $\alpha_c = 0.00^\circ$. 
Figure 7.— Continued; (g) $\alpha_c = 1.81^\circ$, (h) $\alpha_c = 3.42^\circ$. 
Figure 7.—Continued; (i) $\alpha_c = 5.19^\circ$, (j) $\alpha_c = 7.05^\circ$. 
RUN 7  POINT 54  MACH = 0.488  \( R_n = 6.84 \times 10^6 \)
\( c_l = 0.8960 \)  \( c_d = 0.0155 \)  \( c_m = 0.0142 \)

RUN 22  POINT 155  MACH = 0.491  \( R_n = 6.80 \times 10^6 \)
\( c_l = 0.9461 \)  \( c_d = 0.0442 \)  \( c_m = 0.0059 \)

Figure 7.— Continued; (k) \( \alpha_c = 7.13^\circ \), (l) \( \alpha_c = 8.00^\circ \).
Figure 7.— Continued; (m) $\alpha_c = 8.06^\circ$, (n) $\alpha_c = 8.95^\circ$. 
RUN 7  
POINT 56  
MACH = 0.487  
$c_f = 0.9562$  
$c_d = 0.0751$  
$c_m = -0.0111$  

RUN 22  
POINT 157  
MACH = 0.488  
$c_f = 0.9741$  
$c_d = 0.0994$  
$c_m = -0.0316$  

Figure 7.— Concluded; (o) $\alpha_c = 9.08^\circ$, (p) $\alpha_c = 10.00^\circ$.  

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Figure 8.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.54$; (a) $\alpha_c = -3.50^\circ$, (b) $\alpha_c = -1.80^\circ$. 
Figure 8.— Continued; (c) $\alpha_c = -0.03^\circ$, (d) $\alpha_c = -0.03^\circ$. 
Figure 8.—Continued; (e) $\alpha_c = 1.78^\circ$, (f) $\alpha_c = 3.61^\circ$. 
Figure 8.— Continued; (g) $\alpha_c = 5.34^\circ$, (h) $\alpha_c = 7.17^\circ$. 

RUN 8 POINT 65
MACH = 0.535
$c_l = 0.7269$
$c_d = 0.0111$
$c_m = 0.0107$
$R_n = 7.15 \times 10^6$

RUN 8 POINT 66
MACH = 0.533
$c_l = 0.8917$
$c_d = 0.0374$
$c_m = 0.0105$
$R_n = 7.16 \times 10^6$
RUN 8  POINT 67  MACH = 0.534  \( R_n = 7.25 \times 10^6 \)

\( c_l = 0.9252 \)  \( c_d = 0.0620 \)  \( c_m = -0.0096 \)

\( R_n = 7.25 \times 10^6 \)

Figure 8.— Concluded; (i) \( \alpha_c = 8.17° \).
Figure 9.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.59$; (a) $\alpha_c = -3.45^\circ$, (b) $\alpha_c = -1.81^\circ$. 
Figure 9.—Continued; (c) $\alpha_c = -0.07^\circ$, (d) $\alpha_c = -0.01^\circ$. 
RUN 9 POINT 74 MACH = 0.589  
$c_l = 0.2615$  
$c_d = 0.0065$  
$c_m = 0.0031$  
$R_n = 7.55 \times 10^6$

RUN 9 POINT 76 MACH = 0.583  
$c_l = 0.5158$  
$c_d = 0.0075$  
$c_m = 0.0087$  
$R_n = 7.66 \times 10^6$

Figure 9.—Continued; (e) $\alpha_c = 1.77^\circ$, (f) $\alpha_c = 3.45^\circ$. 

(Figure showing data on upper and lower surfaces with chord, $x/c$ axis)
Figure 9.— Concluded; (g) $\alpha_c = 5.28^\circ$, (h) $\alpha_c = 7.17^\circ$. 
Figure 10.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.64$; (a) $\alpha_c = -3.52^\circ$, (b) $\alpha_c = -1.66^\circ$
RUN 10  POINT 80  MACH = 0.639  \( R_n = 7.83 \times 10^6 \)
\( c_l = 0.0069 \)
\( c_d = 0.0065 \)
\( c_m = 0.0014 \)

RUN 10  POINT 83  MACH = 0.641  \( R_n = 8.00 \times 10^6 \)
\( c_l = 0.0103 \)
\( c_d = 0.0067 \)
\( c_m = 0.0012 \)

Figure 10.— Continued; (c) \( \alpha_c = -0.03^\circ \), (d) \( \alpha_c = 0.02^\circ \).
Figure 10.—Continued; (e) $\alpha_c = 1.77^\circ$, (f) $\alpha_c = 3.51^\circ$. 
Figure 10.— Concluded; (g) $\alpha_c = 5.27^\circ$, (h) $\alpha_c = 7.24^\circ$. 
Figure 11.-- Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.69$; (a) $\alpha_c = -3.44^\circ$, (b) $\alpha_c = -1.71^\circ$. 
RUN 12  POINT 91  MACH = 0.683  \( R_n = 8.08 \times 10^6 \)
\( c_l = 0.0072 \)  \( c_d = 0.0063 \)  \( c_m = 0.0015 \)

\( \text{\textbullet{} UPPER SURFACE} \)
\( \text{\textcircle{} LOWER SURFACE} \)

Figure 11.— Continued; (c) \( \alpha_c = -0.01^\circ \), (d) \( \alpha_c = 0.06^\circ \).
Figure 11.—Continued; (e) $\alpha_c = 1.78^\circ$, (f) $\alpha_c = 3.49^\circ$. 
Figure 11.— Concluded; (g) $\alpha_c = 5.29^\circ$. 

RUN 12

POINT 97

MACH = 0.686

$R_n = 8.74 \times 10^6$

$c_l = 0.7299$  

$c_d = 0.0567$

$c_m = -0.0119$
Figure 12.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.74$; (a) $\alpha_c = -3.42^\circ$, (b) $\alpha_c = -1.72^\circ$. 
Figure 12.—Continued; (c) $\alpha_c = -0.02^\circ$, (d) $\alpha_c = 0.00^\circ$. 

RUN 13  POINT 100  MACH = 0.731  $R_n = 8.61 \times 10^6$

$c_l = 0.0107$  $c_d = 0.0066$  $c_m = 0.0013$

RUN 13  POINT 103  MACH = 0.736  $R_n = 8.79 \times 10^6$

$c_l = 0.0005$  $c_d = 0.0065$  $c_m = 0.0000$
Figure 12.— Continued; (e) $\alpha_c = 1.71^\circ$, (f) $\alpha_c = 3.45^\circ$. 
RUN 13  POINT 106  MACH = 0.736  $R_n = 9.17 \times 10^6$
\(c_l = 0.6662\)  \(c_d = 0.0590\)  \(c_m = -0.0319\)

- UPPER SURFACE
- LOWER SURFACE

Figure 12.— Concluded; (g) \(\alpha_c = 4.34^\circ\).
Figure 13.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.79$; (a) $\alpha_c = -3.41^\circ$, (b) $\alpha_c = -1.64^\circ$. 
Figure 13.—Continued; (c) $\alpha_c = -0.07^\circ$, (d) $\alpha_c = 1.66^\circ$. 
Figure 13.— Concluded; (e) $\alpha_c = 2.54^\circ$, (f) $\alpha_c = 3.39^\circ$. 
Figure 14.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.84$; (a) $\alpha_c = -3.29^\circ$, (b) $\alpha_c = -1.73^\circ$. 
Figure 14.—Continued; (c) $\alpha_c = -0.05^\circ$, (d) $\alpha_c = 1.64^\circ$. 
Figure 14—Concluded; (e) $\alpha_c = 2.47^\circ$, (f) $\alpha_c = 3.44^\circ$. 
Figure 15.— Chordwise pressure distribution of the OLS/TAAT airfoil, $M = 0.88$; (a) $\alpha_c = -3.49^\circ$, (b) $\alpha_c = -1.65^\circ$. 

RUN 18  POINT 131  MACH = 0.876  $R_n = 9.65 \times 10^6$

$c_f = -0.5477$  $c_d = 0.0649$  $c_m = 0.0950$

RUN 18  POINT 130  MACH = 0.881  $R_n = 9.56 \times 10^6$

$c_f = -0.3626$  $c_d = 0.0446$  $c_m = 0.0923$
RUN 18  POINT 132  MACH = 0.887  \( R_n = 9.93 \times 10^6 \)
\( c_l = 0.3841 \)  \( c_d = 0.0605 \)  \( c_m = -0.0966 \)

\[ \begin{array}{c}
\text{NEGATIVE } C_p \\
0 & 0.5 & 1.0 & 1.5 & 2.0 & 2.5 & 3.0 \\
\end{array} \]

\[ \begin{array}{c}
\text{UPPER SURFACE} \\
\text{LOWER SURFACE} \\
\end{array} \]

RUN 18  POINT 133  MACH = 0.873  \( R_n = 10.10 \times 10^6 \)
\( c_l = 0.5196 \)  \( c_d = 0.0701 \)  \( c_m = -0.1035 \)

\[ \begin{array}{c}
\text{NEGATIVE } C_p \\
0 & 0.5 & 1.0 & 1.5 & 2.0 & 2.5 & 3.0 \\
\end{array} \]

\[ \begin{array}{c}
\text{UPPER SURFACE} \\
\text{LOWER SURFACE} \\
\end{array} \]

Figure 15.— Concluded; (c) \( \alpha_c = 1.64^\circ \), (d) \( \alpha_c = 2.54^\circ \).
Figure 16.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.34$, $R_n = 4.7 \times 10^6$. 
Figure 17.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.39$, $R_n = 5.4 \times 10^6$. 
Figure 18.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.44$, $R_n = 6.0 \times 10^6$. 

MACH = 0.44 \hspace{1em} R_n = 6.0 \times 10^6
Figure 19.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.49$, $R_n = 6.6 \times 10^6$. 
Figure 20.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.54$, $R_n = 7.0 \times 10^6$. 

\begin{align*}
MACH &= 0.54 \quad R_n = 7.0 \times 10^6 \\
\text{c}_l &= 0.54 \\
\text{c}_d &= 0.15 \\
\text{c}_{m\%} &= 0.05
\end{align*}
Figure 21.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.59, R_{n} = 7.6 \times 10^{6}$. 

$MACH = 0.59 \quad R_{n} = 7.6 \times 10^{6}$
Figure 22.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.64$, $R_n = 8.1 \times 10^6$. 
Figure 23.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.69, R_n = 8.4 \times 10^6$. 
Figure 24.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.74$, $R_n = 8.8 \times 10^6$. 
Figure 25. – Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.78$, $R_n = 8.9 \times 10^6$. 
Figure 26.— Aerodynamic characteristics of the OLS/TAAT airfoil, $M = 0.83, R_n = 9.3 \times 10^6$. 

$MACH = 0.83 \quad R_n = 9.3 \times 10^6$
Figure 27. – Aerodynamic characteristics of the OLS/TAAT airfoil, \( M = 0.88, R_n = 9.8 \times 10^6 \).