FLOW CHARACTERISTICS ABOUT A TRAILING EDGE

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ABSTRACT

The mandate of this thesis was to determine the mean and turbulent flow quantities of an attached, fully-developed boundary layer of an aft-loaded airfoil. The steady, two dimensional flow characteristics of the near and far wake were also studied. This was done to gain an understanding of the viscous effects near the trailing edge of the airfoil in the hope of providing accurate data for airfoil performance models.

Tests were conducted at a Reynolds number of 1.7 million based on chord and a Mach number of 0.14. The airfoil model on loan from Boeing of Canada's deHavilland Division was restricted to zero incidence and the boundary layer was artificially tripped at 0.075c. Artificial tripping was employed to provide a well defined transition location over that of free transition for numerical analysis.

The hot-wire anemometer measurements taken in the boundary layer and wake near the trailing edge reveal that a complex interaction takes place just aft of the airfoil where the upper and lower surface flows combine. The boundary layer thickness was found to increase quickly over the last few percent chord with a corresponding increase in turbulence intensity. The extreme asymmetry in the near wake profile was seen to transform into an essentially symmetric profile by approximately X/C= 1.075. Parameters, such as, skin friction coefficients, displacement and momentum thicknesses were determined
from the x-wire profiles. Numerical predictions by the computer code AADP9 of these parameters, together with the coefficients of lift, drag, and quarter chord moment compared favorably with the experimental values.
CONTENTS

Acknowledgements (ii)
Abstract (iii)
Contents (v)
List of Figures (vii)
List of Tables (x)
List of Symbols (xi)

1.0 INTRODUCTION 1

2.0 EXPERIMENTAL EQUIPMENT 6
2.1 Wind Tunnel Facility 6
2.2 Airfoil Model 7
2.3 Surface Pressure and Drag Measurements 9
2.4 Hot Wire Boundary Layer and Wake Measurements 10

3.0 EXPERIMENTAL PROCEDURE 11
3.1 Pressure Measurements 11
3.2 Hot Wire Measurements 12
3.3 Hot Wire Anemometer Theory 14

4.0 THEORY AND ANALYSIS 21
4.1 Surface Pressure Integration 21
4.2 Wind Tunnel Boundary Corrections 22
4.3 Drag Calculation 24
4.4 Boundary Layer Integral Parameter Calculation 25
4.5 Estimation of Skin Friction 25
4.6 Cebeci/Smith Eddy-Viscosity Model 26

5.0 DISCUSSION OF RESULTS 31
5.1 Surface Pressure Profile 31
5.2 Lift, Moment and Drag 32
5.3 Wake Profiles 33
5.4 Boundary Layer Profiles 36
5.5 Comparison to Eddy-Viscosity Model 40
5.6 Comparison to AADP9 Computer Code 42
5.7 Estimation of Uncertainties 44
<table>
<thead>
<tr>
<th>Section</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>6.0 CONCLUSIONS AND RECOMMENDATIONS</td>
<td>48</td>
</tr>
<tr>
<td>7.0 LIST OF WORKS CONSULTED</td>
<td>51</td>
</tr>
<tr>
<td>TABLES</td>
<td>54</td>
</tr>
<tr>
<td>FIGURES</td>
<td>56</td>
</tr>
<tr>
<td>Figure</td>
<td>Description</td>
</tr>
<tr>
<td>--------</td>
<td>-------------</td>
</tr>
<tr>
<td>1</td>
<td>Wind Tunnel Configuration</td>
</tr>
<tr>
<td>2</td>
<td>Honey-comb and Screen Geometry</td>
</tr>
<tr>
<td>3</td>
<td>Modified de Havilland Airfoil</td>
</tr>
<tr>
<td>4</td>
<td>Pressure Data Acquisition System Block Diagram</td>
</tr>
<tr>
<td>5</td>
<td>General Traverse Geometry for Data Collection</td>
</tr>
<tr>
<td>6</td>
<td>Hot Wire Data Acquisition System Block Diagram</td>
</tr>
<tr>
<td>7</td>
<td>Typical Calibration for the A/D Converter and Pressure Transducer</td>
</tr>
<tr>
<td>8</td>
<td>Typical Hot Wire Calibration</td>
</tr>
<tr>
<td>9</td>
<td>Geometry for X-Wire Theory</td>
</tr>
<tr>
<td>10</td>
<td>Effect of Wind Tunnel Boundary Corrections on the Experimental Surface Pressure Profile</td>
</tr>
<tr>
<td>11</td>
<td>Mean Streamwise Velocity Profiles in the Near Wake</td>
</tr>
<tr>
<td>12</td>
<td>Mean Streamwise Velocity Profiles in the Wake</td>
</tr>
<tr>
<td>13</td>
<td>Mean Streamwise Velocity Profiles in the Wake</td>
</tr>
<tr>
<td>14</td>
<td>Development of Minimum Wake Velocity</td>
</tr>
<tr>
<td>15</td>
<td>Mean Transverse Velocity Profile in the Near Wake</td>
</tr>
<tr>
<td>16</td>
<td>Mean Transverse Velocity Profile in the Wake</td>
</tr>
<tr>
<td>17</td>
<td>Streamwise Turbulence Intensity Profiles in the Near Wake</td>
</tr>
<tr>
<td>18</td>
<td>Streamwise Turbulence Intensity Profiles in the Wake</td>
</tr>
<tr>
<td>19</td>
<td>Streamwise Turbulence Intensity Profiles in the Wake</td>
</tr>
<tr>
<td>20</td>
<td>Transverse Turbulence Intensity Profiles in the Near Wake</td>
</tr>
<tr>
<td>21</td>
<td>Transverse Turbulence Intensity Profiles in the Wake</td>
</tr>
</tbody>
</table>
Figure 22 Reynolds' Shear Stress Profiles in the Near Wake
Figure 23 Reynolds' Shear Stress Profiles in the Wake
Figure 24 Boundary Layer Streamwise Velocity Profiles
Figure 25 Boundary Layer Streamwise Velocity Profiles
Figure 26 Boundary Layer Streamwise Velocity Profiles
Figure 27 Boundary Layer Transverse Velocity Profiles
Figure 28 Boundary Layer Transverse Velocity Profiles
Figure 29 Boundary Layer Streamwise Turbulence Intensity Profiles
Figure 30 Boundary Layer Streamwise Turbulence Intensity Profiles
Figure 31 Boundary Layer Streamwise Turbulence Intensity Profiles
Figure 32 Boundary Layer Transverse Turbulence Intensity Profiles
Figure 33 Boundary Layer Transverse Turbulence Intensity Profiles
Figure 34 Boundary Layer Transverse Turbulence Intensity Profiles
Figure 35 Reynolds' Shear Stress Profile in the Boundary Layer
Figure 36 Reynolds' Shear Stress Profile in the Boundary Layer
Figure 37 Reynolds' Shear Stress Profile in the Boundary Layer
Figure 38 Measured Displacement and Momentum Thickness
Figure 39 Boundary Layer Profiles in Wall Coordinates
Figure 40 Comparison of Experimental Reynolds' Shear Stress with Eddy-Viscosity Model
Figure 41 Comparison of Experimental Surface Pressure Distribution at $\alpha = 0.002^\circ$ with AADP9
Figure 42  Comparison of Experimental Displacement and Momentum Thicknesses with AADP9

Figure 43  Comparison of Experimental Skin Friction Coefficient on the Upper Surface with AADP9
LIST OF TABLES

Table 1  Effect of Wind Tunnel Wall Corrections
Table 2  Effect of Location on Evaluation of Squire-Young Drag
Table 3  Boundary Layer Parameters
Table 4  Comparison of Experimental $C_L$, $C_D$, $C_{Mc/4}$ to AADP9 Predicted Coefficients
LIST OF SYMBOLS

A constant in King's law.
B constant in King's law.
C_N chordwise force coefficient.
C_D drag coefficient.
C_Dp pressure drag coefficient.
C_f skin friction coefficient.
C_L lift coefficient.
C_Mc/4 pitching moment coefficient about quarter chord.
C_N normal force coefficient.
c chord.
g stagnation pressure.
H shape factor, $\delta^*/\theta$
h wind tunnel height.
M Mach number.
n exponent in King's law.
p static pressure.
q dynamic pressure.
Re Reynolds number.
R Eff mean effective resultant velocity.
r' eff fluctuation of the effective resultant velocity.
U mean streamwise velocity.
U_\infty free stream velocity.
U_ref wind tunnel reference velocity.
\( U_e \) edge velocity.
\( u^+ \) velocity in wall coordinates, \( U/u_* \).
\( u' \) streamwise root mean squared velocity \( \sqrt{u'^2}/U_\infty \).
\( u_* \) friction velocity.
\( u'^2 \) Reynolds normal stress.
\(-u'v'\) Reynolds shear stress.
\( V \) mean transverse velocity or voltage.
\( v'^2 \) Reynolds normal stress.
\( v' \) transverse root mean squared velocity \( \sqrt{v'^2}/U_\infty \).
\( W \) mean spanwise velocity.
\( w' \) spanwise, root mean squared velocity \( \sqrt{w'^2}/U_\infty \).
\( y \) transverse coordinate.
\( y^+ \) transverse distance in wall coordinates, \( u_*/y/v \).
\( z \) spanwise coordinate.
\( \Lambda \) shape factor for boundary corrections.
\( \alpha \) angle of incidence.
\( \delta \) boundary layer thickness.
\( \delta^* \) displacement thickness.
\( e \) incremental velocity increase from blockage.
\( \theta \) momentum thickness.
\( \nu \) kinematic viscosity.
\( \rho \) density.
\( \sigma \) \( (\pi^2/48) (c/h)^2 \).
\( \tau_w \) wall shear stress.
SUBSCRIPTS

\( \infty \) free stream or reference conditions.

1 conditions at the trailing edge or in the wake.

2 conditions at a wake measurement position.

c corrected.

e edge.

l lower half.

sb solid blockage.

u uncorrected wind tunnel measurements or upper half.

wb wake blockage.

(xiii)
1.0 INTRODUCTION

The purpose of this investigation was to determine the mean and turbulent flow quantities of an attached, fully developed boundary layer of an aft-loaded subsonic airfoil. The steady two dimensional flow characteristics of the near and far wake were also studied. This was done in order to gain an understanding of the viscous effects near the trailing edge of the airfoil in the hope of providing accurate data for airfoil performance models.

The experiment was performed at the University of Toronto Institute for Aerospace Studies, (UTIAS), anechoic wind tunnel facility. Tests were conducted at a Mach number of 0.14 and a Reynolds number of 1.7 million based on chord and restricted to zero incidence. An airfoil on loan from Boeing of Canada's deHavilland division was modified to provide a sharp trailing edge with the boundary layer artificially tripped at 0.075\(c\). Artificial tripping of the boundary layer was employed to provide a well defined transition location for numerical analysis over that of free transition.

Surface pressure profiles were taken and integrated to give lift and pitching moment. The drag was measured from wake traverses. Cross-wire and single-wire anemometer measurements of the mean flow speed and turbulence intensity were taken for a lift coefficient of
The skin friction, displacement and momentum thicknesses were ascertained from the velocity profiles. The data collected was compared to the turbulent boundary layer predictions of a UTIAS developed airfoil design/analysis code utilizing the finite difference method of Keller/Cebeci and the momentum integral method of Green. Further, the measured Reynolds' shear stress profiles were compared to that predicted by the Cebeci/Smith eddy-viscosity model for turbulent boundary layers.

Investigation into the flow characteristics near the trailing edge flow of airfoils has motivated a number of researchers to attempt to quantify the nature of turbulence and its effect on the mean flow. An accurate method for the prediction of transition and subsequent flow development would be advantageous for aircraft design companies. For example, future airfoil designs could incorporate very low drag coupled with high lift characteristics without the need for extensive wind tunnel tests. To attain this goal various numerical methods have been devised to simulate the viscous-inviscid interaction, flow curvature, and pressure gradients at the trailing edge. Independent of the theory used the need for reliable experimental data is of utmost importance. To this end, a number of studies have been performed. The areas of study have been quite broad including both low/high Reynolds number and Mach number effects for symmetric and non-symmetric airfoils. Typically, mean flow measurements, surface pressure distributions, and the static pressure data across the trailing edge are presented, in combination with the
lift, drag, and moment calculations. More recent studies include
numerical predictions of the displacement and momentum thicknesses,
mean flow and turbulent quantities.

Due to the scope of this investigation a detailed study of some
early works by researchers like BREBNER et al. [1], HURLEY et al. [2],
SPAID et al. [3], COOK [4], and VISWANATH et al. [5] will not be
presented. Some of the more recent works include the following. Hah
and Lakshminarayana [6] reported on their measurements of the mean
and turbulence quantities for a NACA 0012 airfoil. For angles of
incidence of 3°, 6°, and 9°, an x-wire anemometer was placed in the
near and far wake. The test was conducted for a low Mach number at a
Reynolds number of 0.3 million based on chord. Boundary layer
measurements were not reported but the data showed that the wake
became more asymmetric and development of the mean and turbulence
profiles slowed as the angle of attack increased. The wake profiles
were seen to become symmetric by approximately one chord downstream.
At higher angles of incidence streamline curvature was evident even
though the model was symmetric. Hah and Lakshminarayana compared
their results to numerical predictions and included the effects of
flow curvature. The turbulence models used generally underestimated
the mean and turbulent flow properties.

YU's [7] report presents the measured mean and turbulent
quantities in the boundary layer and the near wake of a NACA 631-012
airfoil. The symmetric airfoil was placed in a low Mach number 1.25
million Reynolds number flow based on chord. The airfoil was restricted to an angle of attack of zero degrees. Using an x-wire anemometer YU investigated the boundary layer and wake characteristics for both free and fixed transition. The untripped flow exhibited similar characteristics to that of flat plate flow with zero pressure gradient. Inducing transition resulted in an increase in the boundary layer thickness and elimination of the spanwise variations in the flow.

NAKAYAMA'S [8,9] experiments reveal the mean flow, turbulence quantities, pressure fields for what NAKAYAMA refers to as a conventional airfoil (model A) and a supercritical airfoil (model B). Model A was tested at zero angle of attack, whereas, model B was tested at four and twelve degrees. The trailing edge flow characteristics were measured for a low speed (35 m/s) flow of Reynolds number of 1.2 million based on chord. The purpose of the 12° incidence was to induce flow separation in a small region near the trailing edge of model B. The upper surface separation flow characteristics were determined by using a laser doppler velocimeter. In most other cases, the flow field was measured with x-wire anemometers. At zero angle of attack, NAKAYAMA's model A boundary layer profiles exhibited development similar to that of a flat plate boundary layer as did YU's results. Flow curvature and the pressure gradients near the trailing edge were found to be minimal due to the fact that model A is only ten percent thick and nearly symmetric. Although the included angles at the trailing edge of the two models
were essentially equal, model B's lower surface near the trailing edge was very curved. This led to prominent streamline curvature, normal pressure gradients, and intense mixing of the upper and lower surface boundary layers near the trailing edge. Further, the strong adverse and favorable pressure gradients on the upper and lower surfaces, respectively, promoted very asymmetric conditions in the near wake region. The upper surface boundary layer increased in thickness while the lower surface boundary layer was found to decrease in thickness over the last few percent of the airfoil. This thesis is investigating attached boundary layer flows, thus, NAKAYAMA's separated flow case will not be reviewed.

As previously stated, researchers require experimental data for a number of flow conditions and models for comparison to various numerical theories. The researchers mentioned have all contributed to one degree or another to the understanding of trailing edge flows. Understanding of attached and separated flows would allow for improved numerical methods capable of accurately predicting the mean and turbulent flow data.
2.0 EXPERIMENTAL EQUIPMENT AND PROCEDURE

2.1 WIND TUNNEL FACILITY

This experiment was performed at the University of Toronto's Institute for Aerospace Studies (UTIAS) anechoic wind tunnel facility. The continuous flow, open-circuit, closed-wall wind tunnel is powered by a 150 H.P. motor with variably angled blades. The inlet is placed inside a 150 foot diameter geodesic dome which provides a quiescent source of input air that is essentially not effected by atmospheric winds. The dimensions of the tunnel are shown in Figure 1. An area reduction of 8:1 results in a working cross-sectional area of 24"x54" with a top speed of approximately 185 ft/sec. The free stream turbulence level in the empty tunnel is controlled through honeycomb and a series of screens placed at the inlet of the tunnel. The dimensions of which are detailed in Figure 2. The free stream r.m.s. turbulence level (u'/U∞) was measured with a hot wire anemometer and found to be 0.19% over 2 Hz to 20 kHz, however, a large portion of this is electrical noise. The use of a bandpass filter from 10 Hz to 20 kHz resulted in a r.m.s. level of less than 0.1%. The mean flow velocity in the tunnel throughout the empty test section was found to vary by generally less than 0.5%.
2.2 AIRFOIL MODEL

The airfoil model for this experiment was loaned to UTIAS by the deHavilland Division of Boeing Canada. The model consists of a hard foam interior molded about a central steel spar and covered by a smooth, scratch resistant, epoxy resin. As a result of the desire to compare measurements to the predicted quantities by various numerical techniques; it was expedient to alter the original airfoil. The rather blunt trailing edge was converted to a sharp trailing edge by the addition of a small extension with a convex upper surface of constant curvature (\( \kappa = 0.5c \)) and straight lower surface. This configuration became apparent after analysis of the various parameters by a computer code developed at UTIAS in order to maintain the airfoil's high aft-loading without causing boundary layer separation at the trailing edge at typical angles of incidence. The modification resulted in the chord increasing from 18.375" to 19.518" with an expected decrease in thickness from 18\% to 17\%. The model cross-section is shown in Figure 3.

The model was instrumented with 64 pressure taps of 1/32" O.D. stainless steel tubing mounted flush with the surface to determine the surface pressure distribution and the two dimensionality of the flow. However, the last few percent of the trailing edge were not instrumented with any pressure taps. Further, the boundary layer was tripped on both the upper and lower surfaces by 0.004" thick by 0.026" wide tape placed across the span of the airfoil at 0.075c.
This was done in order to enhance boundary layer growth, as well as, precipitate and provide a well defined transition location for the numerical analysis over that of free transition. Placement near the leading-edge extends its range of continued effectiveness at other angles of incidence over that of a further aft location. The airfoil angle of incidence was set by means of a scale providing an accuracy of ±0.02 degrees. The scale zero was determined using a 5" circular cylinder model mounted in place of the airfoil. The cylinder developed no mean lift so that corrections due to the wind tunnel walls were not required. Once the scale was set the airfoil's geometric angles of incidence could be measured and the corrected angles could subsequently be determined as described in Section 4.2. The cylinder had pressure taps mounted symmetrically about a reference line. By rotating the mounting until the difference between these taps was zero, the reference line was aligned with the flow direction. Since both the airfoil and cylinder had accurately placed alignment pins, the common mounting maintained a precise alignment of the airfoil upon removal of the cylinder.
2.3 SURFACE AND WAKE PRESSURE MEASUREMENTS

Pressure data was collected using the computer controlled acquisition system in Figure 4. Through the series of pressure taps connected to the 48 port model J9 Scanivalve rotary pressure scanning device the surface profile was determined. The Scanivalve was controlled through an interface and could be positioned as desired. The Endevco ±2 psi piezoresistive pressure transducer was mated to an Endevco model 4423 signal conditioner and connected to the Spectral Dynamics SD375 Digital Signal Analyzer.

The SD375 is a 12 bit fast fourier transform (FFT) machine capable of frequency and power spectral analysis, as well as, two channel digital sampling. For the purpose of this experiment the SD375 was used as an analog to digital board for the LSI-23 processor with a data transfer rate of 19200 baud through its RS232C port. The KOM MicroPac I computer controlled the Velmex model 8300 traverse controller. The controller would accept positioning data for the bi-axial traverse with a resolution of 0.001" and immediately alert the computer of the completion of any requested movement. The cross-stream axis was configured with a symmetric sting that passed through the tunnel wall into the flow stream (Figure 5). The sting accepted a number of transducers on its common mounting block such as a pitot-static probe as required for all wake drag measurements.
2.4 HOT WIRE BOUNDARY LAYER AND WAKE MEASUREMENTS

Both boundary layer and wake measurements were taken with single
and x-wire hot wire constant temperature anemometry probes. The
basic instrumentation used is shown Figure 6. The elements of the
DISA single wire system included the 55P15 0.5 μm diameter tungsten
miniature wire probe, 55H22 probe support, 55M01 main unit with the
55M10 standard bridge anemometry system and the 55D10 linearizer.
The x-wire system utilized dual channels of the same DISA equipment
except that the sensor and holder were replaced by the 55P63 and
55H25 respectively. All wires were calibrated with the mean flow
normal to the wire axis and the law of cosines was assumed. The
Spectral Dynamics SD375 Digital Signal Analyzer was controlled by the
LSI-23 processor to transfer data at 19200 baud. For any location
the positioning data was sent to the bi-directional Velmex traverse
followed by the command to capture and release the electrical signal
from the DISA 55P63 x-wires by the SD375. While on-line, the LSI
processor stored the raw data (typically 10-12 samples at any one
location), calculated the ensemble average and standard deviation and
then updated the terminal display with the transducer output and
position. Offline, the data was corrected for offsets and converted
from voltage to velocity units.
3.0 EXPERIMENTAL PROCEDURE

For any run during this experiment steps were taken to minimize the effect of temperature changes on the reliability of the results. All electronic devices including the hot wire systems, computer, and calibration equipment were thoroughly warmed up. At the start of each session the tunnel was allowed to run for ten minutes prior to the collection of data. This was done to evacuate and replace the quiescent air in the geodesic dome at the inlet to the wind tunnel, thus, effectively eliminating any temperature gradients in the flow. As the tests progressed flow temperatures were recorded at regular intervals.

3.1 PRESSURE MEASUREMENTS

The following procedure was used when collecting pressure data. The analog to digital (A/D) board of the SD375 was calibrated against a D.C. voltage supply using a Keithley DVM model 172A possessing a 1 mV resolution. The A/D converter board was found to be linear as indicated in Figure 7. The output from the pressure transducer was subsequently calibrated against a known pressure measured with a Betz manometer. A typical calibration may also be found in Figure 7. The transducer offset was set to zero prior to the start of any experiment and immediately rechecked for drift after the completion of the test. Very low frequency surging of the tunnel of less than
0.5% of the dynamic pressure resulted in the need for multiple pressure samples. In the case of the surface pressure measurements, software would be started to execute the sampling of the 48 pressure ports ten times. The raw data, as well as the ensemble average were then recorded. The drag of the airfoil was determined from the measured total and static pressures across the wake. The pitot/static tube was aligned on the span center line of the model and connected to the Scanivalve. A third port on the Scanivalve was used to correct for transducer drift.

3.2 HOT-WIRE MEASUREMENTS

The procedure for the hot wires was similar to that of the pressure system but with a slight modification. The output from hot-wire system used was nonlinear so that calibration was carried out in two parts. The hot-wire circuit was tuned so as to eliminate any electronic instability which could cause system oscillations leading to erroneous data. The nonlinear output voltage was calibrated over a range of flow speeds produced with a centrifugal fan calibrator. The calibrator consisted of a 209XL Canada Fan whose discharge was passed through honeycomb, screens, settling chamber and, finally, nozzle with a 4:1 contraction. The total pressure near the previously calibrated nozzle exit was measured with a Betz manometer and the corresponding hot wire voltage recorded. Once the nonlinear characteristics of the probe were known (see Figure 8) the signal was
then passed through a linearizer. With the probe mounted on the sting and the tunnel operating the probe would be aligned with the trailing-edge. The probe would then be placed at the starting point of the current traverse and collection of data begun. The alignment was performed with the tunnel operating since the sting was found to move downstream from its original position by >0.010" due to the looseness in the traverse guideways. However, there seemed to be no measurable problem with backlash. For the boundary layer traverses the starting point was obtained by moving the probe towards the surface of the airfoil until contact was made. Initially, the probe was set in the resistance measurement mode. Upon contact the resistance of the probe would drop to zero as the aluminum trailing edge shorted the hot wire tines. Immediately, the probe would be backed from the surface and the traverse started once the hot wire circuit, now turned on, had reached thermal equilibrium. Boundary layer traverses were taken normal to the surface as approximated by a stepping motion while the wake traverses were taken normal to the chord line. A right-hand coordinate system was used with the x-axis along the chord line originating at the leading edge of the airfoil and increasing downstream.
3.3 HOT WIRE ANEMOMETER THEORY

Hot-wire anemometry is based on the principle of measuring the amount of heat loss from an electrically heated wire caused by the flow of the fluid surrounding the probe. The Constant Temperature Anemometer (CTA) employs a wheatstone bridge circuit of which one arm is the sensor. The circuit's servo amplifier maintains the probe at a constant temperature for any particular operating condition. Assuming a constant flow temperature and fluid type any change in velocity will be seen as a change in current (voltage). As previously mentioned, the hot wires were calibrated against a jet of known speed. The non-linear hot wire voltage was fit then by the method least squares to King's Law,

\[ V^2 = A U^n + B \]  

(3-1)

in order to evaluate the exponent n for linearization. The linearization produced calibration curves of the form, \( V = A'U + B' \), where the constants \( A' \) and \( B' \) were determined by a least squares fit. It was believed that a more accurate representation of the calibration curve could be accomplished by employing a fourth order least square fit polynomial. The off-line theory utilized to calculate the quantities \( U, V, u', v', u'v' \) was as follows:
As shown in Figure 9, the velocity vectors of the general 3-D flow consist of a mean or time averaged value (denoted by the overhead bar) and a fluctuating component (indicated by the prime notation). Mathematically, the two components are defined as:

\[
\bar{U} = \frac{1}{T} \int_{0}^{T} U(t) \, dt \\
\bar{u}' = \frac{1}{T} \int_{0}^{T} u'(t) \, dt
\]  

(3-2) \hspace{1cm} (3-3)

Similar equations may be written for \( V \) and \( v' \) or \( W \) and \( w' \). Consider the inclined probe geometry shown in Figure 9 contained in the x-y plane experiencing a general 3-D flow.

\[
R_{Nxy} = (\bar{U} + u') \cos \alpha + (\bar{V} + v') \sin \alpha \\
R_{Txy} = (\bar{U} + u') \sin \alpha - (\bar{V} + v') \cos \alpha \\
R_{Nzz} = (\bar{W} + w')
\]  

(3-4) \hspace{1cm} (3-5) \hspace{1cm} (3-6)

Where:

- \( R_{Nxy} \) = velocity component normal to the hot-wire axis in the x-y plane
- \( R_{Txy} \) = velocity component tangential to the hot-wire axis in the x-y plane
- \( R_{Nzz} \) = velocity component normal to the hot-wire axis in the x-z plane
- \( \alpha \) = angle between x axis and line normal to the wire axis.

According to JORGENSEN [32] the directional sensitivity of a hot wire is representable by (in my notation)
\[ R_{\text{effective}}^2 = (R_{Nxy})^2 + k^2(R_{Ty})^2 + h^2(R_{Nzx})^2 \]  

(3-7)

where  
\( k = \text{tangential cooling coefficient} \)
\( h = \text{binormal cooling coefficient} \)

However, others such as CHAMPAGNE [33], have discovered that as the length diameter of a wire probe increases 'k' and 'h' approach zero and one respectively. Typically, 'k' varies from zero to 0.20 while 'h' covers the range 0.8 to 1.0. For the 55P63 x-wire probe used in this experiment 'h' was found to be about 0.20 while 'k' was taken to be 1.0. Therefore, the second term of equation (3-7) was considered small with respect to the first term due to the 'k^2' term and only the cooling effect of the normal velocity components was considered. Substitution of equations (3-4) and (3-6) into (3-7) and expansion results in:

\[ R_{\text{eff}} = (\frac{\overline{U} + 2\overline{U}u' + u'^2}{2}) \cos^2 \alpha + 2(\overline{UV} + \overline{U}v' + u'\overline{V} + u'v') \sin \alpha \cos \alpha + \frac{2}{(\overline{V} + 2\overline{V}v' + v'^2)} \sin^2 \alpha + (\overline{W} + 2\overline{W}w' + w'^2))^{1/2} \]  

(3-8)

Similar to CHAMPAGNE [33] and KLATT [34] the second order turbulence terms were assumed to be small with respect to the first order terms and in 2-D flow \( W = 0 \).

So

\[ R_{\text{eff}} = ([\overline{U} \cos^2 \alpha + 2\overline{UV} \sin \alpha \cos \alpha + \overline{V} \sin^2 \alpha]^2 + [2\overline{U}u' \cos^2 \alpha + 2(\overline{U}v' + u'\overline{V}) \sin \alpha \cos \alpha + 2\overline{V}v' \sin^2 \alpha])^{1/2} \]  

(3-9)
The collection of terms into time averaged and fluctuating components allow for a binomial series expansion.

\[ R_{\text{eff}} = R_{\text{eff}} \left( 1 + \left[ Uu'\cos^2\alpha + (Uv' + u'V)\sin\alpha\cos\alpha + Vv'\sin^2\alpha \right] \right)^{-1} \]  

\[ \text{where } R_{\text{eff}} = \left[ U^2 \cos^2\alpha + 2UV \sin\alpha\cos\alpha + V^2 \sin^2\alpha \right] \]  

Equation (3-10) is a general equation usable for either wire in a cross configuration where \( \alpha \) becomes \( \alpha_1 = 45^\circ \) and \( \alpha_2 = 135^\circ \). Further, equation (3-8) has a mean and fluctuating component. Performing a time average results in an expression for the mean effective velocity:

\[ R_{\text{eff}} = \left[ U^2 \cos^2\alpha + 2UV \sin\alpha\cos\alpha + V^2 \sin^2\alpha \right]^{1/2} \]  

Substitution of \( \alpha_1 = 45^\circ \) and \( \alpha_2 = 135^\circ \) results in the form

\[ \bar{R}_{\text{eff}}_1 = \left[ \frac{1}{\sqrt{2}} U + \frac{1}{\sqrt{2}} V \right] \quad \text{and} \quad \bar{R}_{\text{eff}}_2 = \left[ \frac{1}{\sqrt{2}} U - \frac{1}{\sqrt{2}} V \right] \]  

For convenience each equation was normalized with respect to a free stream position where only one velocity component was present thus

\[ \bar{R}_{\text{eff}}_{\text{norm}}_1 = \frac{1}{\sqrt{2}} U \quad \text{and} \quad \bar{R}_{\text{eff}}_{\text{norm}}_2 = \frac{1}{\sqrt{2}} U \]  

Normalizing, adding, subtracting and rearranging results in:
\[
\frac{U}{U_0} = \frac{1}{2} \left[ \frac{r_{\text{eff}}}{R_{\text{eff}_1}} + \frac{r_{\text{eff}}}{R_{\text{eff}_2}} \right] \quad (3-17)
\]

\[
\frac{V}{U_0} = \frac{1}{2} \left[ \frac{r_{\text{eff}}}{R_{\text{eff}_1}} - \frac{r_{\text{eff}}}{R_{\text{eff}_2}} \right] \quad (3-18)
\]

where the \( R_{\text{eff}} \) terms are replaced by an appropriate calibration relation converting the hot wire voltage output to velocity units.

The turbulence quantities were determined as follows:

Since
\[
R_{\text{eff}}(t) = \overline{R_{\text{eff}}} + r'_{\text{eff}}(t) \quad (3-19)
\]

then
\[
r'_{\text{eff}}(t) = \overline{R_{\text{eff}}} - R_{\text{eff}}(t)
\]

So examination of equations (3-10) and (3-12) reveals

\[
r_{\text{eff}}' = \frac{\overline{u'u'} \cos^2 \alpha + (\overline{u'v'} + u'v) \sin \alpha \cos \alpha + \overline{v'v'} \sin^2 \alpha}{R_{\text{eff}}} \quad (3-20)
\]

if \( \alpha \) becomes \( \alpha_1 = 45^\circ \) and \( \alpha_2 = 135^\circ \)

\[
r_{\text{eff}}' = \frac{1}{\sqrt{2}} (u' + v') \quad r_{\text{eff}}' = \frac{1}{\sqrt{2}} (u' - v') \quad (3-21)
\]

Normalizing with equations (3-15) and (3-16), adding, subtracting and rearranging:

\[
\frac{u'}{U_0} = \frac{1}{2} \left[ \frac{r_{\text{eff}}}{R_{\text{eff}_1}} + \frac{r_{\text{eff}}}{R_{\text{eff}_2}} \right] \quad (3-23)
\]
As before the $R$ and $r$ terms are replaced with suitable calibration relations. The turbulent flow fluctuations were measured by means of the voltage fluctuations of the hot wire output voltage, therefore, the $r$ relation is determined from the derivative of the calibration relation. Thus with $(u'/U_\infty)$ and $(v'/U_\infty)$ known the root mean square values may be calculated. Finally, the Reynolds shear stress term is found through a time average of the product $(u'/U_\infty)(v'/U_\infty)$.

Hot-wire measurements are susceptible to errors due to ambient temperature fluctuations. According to BEARMAN [17], a change in flow temperature of 1°C can result in an error of about 1% in the mean velocity while the turbulence intensity measurements are minimally affected. However, the correction software was not utilized as the temperature variation over any data run was less than 1°C. However, corrections were applied to the $x$-wire data as required to account for any misalignment of the probe with the mean flow direction or airfoil surface according to the cosine law. Wall proximity corrections were also considered for the hot wire boundary layer measurements. The presence of a solid boundary will increase the cooling of the wire which produces an erroneously high output voltage. According to the correction method of HEBBAR [19] for turbulent flow, the laminar sublayer (especially if $y^+ \leq 5$) requires voltage corrections. For this experiment it is doubtful that any

\[
v'/U_\infty = (1/2) \left[ \frac{r'_{eff_1}}{R_{eff_1}} - \frac{r'_{eff_2}}{R_{eff_2}} \right]
\]

(3-24)
measurements could be made in the region because of the geometry of the x-wire probes and the Reynolds number of the test.
4.0 THEORY AND ANALYSIS

4.1 SURFACE PRESSURE INTEGRATION

Upon collection of the data as described in Section 2.3 the pressure forces were resolved into their normal and tangential components with respect to the airfoil chord line. The determination of the lift, pressure drag, and quarter chord moment with respect to the chord line of the airfoil was carried out by a computer program employing a cubic spline during integration of the distribution. To resolve these components into free stream referenced values the following equations were used.

The lift coefficient was evaluated from:

\[ C_L = C_N \cos \alpha_c - C_C \sin \alpha_c \]  \hspace{1cm} (4-1)

where \( \alpha_c \) = corrected angle of incidence.

The pressure drag was calculated from:

\[ C_{Dp} = C_N \sin \alpha_c + C_C \cos \alpha_c \]  \hspace{1cm} (4-2)

The quarter chord moment was determined by including the factor of distance from the resolved forces to the quarter-chord position.
From a private conversation with NELSON [31], it was known that the unmodified airfoil exhibited a trailing edge pressure of approximately half of the sum of the furthest aft upper and lower surface pressure tap pressure values. As previously stated, the final few percent of the model was not instrumented with pressure taps so the average of the actual last two taps was used to approximate the trailing edge pressure. No further corrections were made to the raw data prior to integration but wind tunnel wall corrections were employed, as detailed in Section 4.2.

4.2 WIND TUNNEL BOUNDARY CORRECTIONS

Ideally, a wind tunnel test should simulate the conditions experienced by the model in free flight. However, the presence of the tunnel walls constraining the flow will contribute to incorrect values for the angle of attack, lift, drag, and moment. Physically mounting a model in a closed section results in the distortion of the flow field around the model leading to an effect known as solid blocking. Solid blocking causes an increase in the magnitude of lift and drag over the true values. Further, the wake is prohibited from expanding as it would in free, unconstrained conditions. This effectively increases the drag measured for a model.

The method of correction detailed by RAE/POPE [13] is based on the method of images which mathematically models the flow pattern
about the model by a system of vortices. The key equations for this theory are summarized below:

\[ \alpha = \alpha_u + \left(\frac{57.3\sigma}{2\pi}\right) \left(C_{Lu} + 4C_{Mc/4a}\right) \]  
(4-3)

\[ C_L = C_{Lu} \left(1-\sigma-2\varepsilon\right) \]  
(4-4)

\[ C_{Mc/4} = C_{Mc/4u} \left(1-2\varepsilon\right) + \sigma C_L/4 \]  
(4-5)

\[ C_{Do} = C_{Du} \left(1-3e_{sb}-2e_{wb}\right) \]  
(4-6)

\[ q = q_u \left(1+2\varepsilon\right) \]  
(4-7)

\[ Re = Re \left(1+\varepsilon\right) \]  
(4-8)

\[ \varepsilon = e_{sb} + e_{wb} \quad e_{sb} = \Lambda\sigma \quad e_{wb} = \left(c/2h\right) C_{Du} \]  
(4-9)

\[ \sigma = \left(\frac{\pi^2}{48}\right) \left(c/h\right)^2 \]  
(4-10)

A solid blockage shape factor of \(\Lambda=0.33\) was used for the corrections, together with a wind tunnel height of 54", and an airfoil chord of 19.518". The corrected values may be found in Table 1. The correction to the lift coefficient was less than 5% as was the correction to the quarter-chord moment coefficient while the
angle of attack was corrected from zero degrees to 0.002 degrees. Further, the total drag was altered by less than 3%.

4.3 DRAG CALCULATION

The total drag on a body consists of both skin friction drag and form drag. Form drag or pressure drag may be found through the integration of the surface pressure distribution of the body. The skin friction drag is considerably more difficult to assess. As a result, the model's total drag was determined through the methods developed by Betz and Jones as described by SCHLICHTING [12]. Both methods are based on the loss of momentum of the mean flow after passing the airfoil. The methods essentially produced equivalent values for $C_d$. Jones' defining equation is:

$$C_d = 2\int \frac{g_2-p_2}{q_0} (1 - \frac{g_2-p_0}{q_0}) \, d(y/c) \quad (4-13)$$

While the integral used by Betz is more involved:

$$C_d = \int \frac{g_2-p_2}{q_0} \, d(y/c) + \int (\frac{g_2-p_2}{q_0} - \frac{g_2-p_2}{q_0})(\frac{g_2-p_2}{q_0} + \frac{g_2-p_2}{q_0} - 2) \, d(y/c) \quad (4-14)$$

Both integrations extend across the wake width. For explanation of the subscripts please refer to the List of Symbols on page (x1).
4.4 BOUNDARY LAYER INTEGRAL PARAMETER CALCULATION

The incompressible boundary layer integral parameters, namely, the displacement thickness and momentum thickness were determined from the uncorrected boundary layer profiles through integration of a cubic spline. The parameters are defined as follows:

\[
\frac{\delta^*}{c} = \int_0^{\delta/c} (1 - \frac{U}{U_e}) \, d\left(\frac{y}{c}\right) \quad (4-15)
\]

\[
\frac{\theta}{c} = \int_0^{\delta/c} \frac{U}{U_e} \left(1 - \frac{U}{U_e}\right) \, d\left(\frac{y}{c}\right) \quad (4-16)
\]

\[H = \frac{\delta^*}{\theta} \quad (4-17)\]

Each integration was performed from the airfoil surface to the edge of the boundary layer (y=\delta) where the edge velocity, U_e, was taken to be the value of U at y=\delta.

4.5 ESTIMATION OF SKIN FRICTION

Skin friction was evaluated by least squares fitting the inner portion of the boundary layer profiles to Coles' Law of the Wall [20].

\[u^+ = (1/0.41) \ln y^+ + 5.0 \quad (4-18)\]

where
The skin friction coefficient is related to the skin friction velocity by:

\[
uy^+ = U/u_* \quad y^+ = u_y/v \tag{4-19}
\]
\[
C_f = 2u_w/(\rho U_e^2) = 2(u_*/U_e)^2 \tag{4-20}
\]

An iterative least squares fit was performed over the region \(50 \leq y^+ \leq 250\). Any data points outside this region were neglected during any subsequent fit. An expanded discussion may be found in Section 5.4.

4.6 CEBECI/SMITH EDDY-VISCOSITY MODEL

With the advent of powerful computers, numerical turbulence models are of great interest in recent attempts to accurately predict the flow characteristics about airfoils, such as, the mean velocity or turbulence intensity distribution across a turbulent boundary layer. The more popular mean flow models describe turbulence by relating the Reynolds stresses to the local mean velocity gradient. Two of these theories include Boussinesq's 1877 eddy-viscosity model and Prandtl's 1925 mixing-length concept (SCHLICHTING [12]). Boussinesq assumed that the turbulent stresses are proportional to the velocity gradient. In particular,
where \( \varepsilon_m = \text{eddy viscosity} \) (units: length \( \times \) velocity).

Prandtl first proposed:

\[
-\rho u'v' = \rho \varepsilon_m \frac{\partial u}{\partial y} \tag{4-22}
\]

thus, the eddy-viscosity and mixing-length theories are related by:

\[
\varepsilon_m = l^2 \left| \frac{\partial u}{\partial y} \right| \tag{4-24}
\]

In a fully turbulent boundary layer, the mixing length, \( l \), is defined over two regions, namely,

\[
l_1 = \kappa y = 0.4y \quad y_0 \leq y \leq y_c \tag{4-25}
\]

\[
l_0 = \alpha_1 \delta = 0.075\delta \quad y_c \leq y \leq \delta \tag{4-26}
\]

where \( y_0 \) is a small distance from the wall and \( y_c \) is obtained from the continuity of \( l \). Similarly, \( \varepsilon_m \) is defined over 2 regions:

\[
(\varepsilon_m)_1 = \frac{l^2}{l_1} \left| \frac{\partial u}{\partial y} \right| \quad y_0 \leq y \leq y_c \tag{4-27}
\]

\[
(\varepsilon_m)_0 = \alpha u_0 \delta^* \quad y_c \leq y \leq \delta \tag{4-28}
\]

where

\[
\delta^*_k = \int_{y_0}^{\infty} [1 - (u/u_e)] \, dy
\]
Equations (4-27) and (4-28) are valid outside the boundary sublayer and buffer-layer near the wall. VAN DRIEST [29] proposed a modified mixing length defined as:

$$L = l_i \left[ 1 - \exp(-y/A) \right] \tag{4-29}$$

where $A = 26v (r_w/\rho)^{-1/2}$ = damping length constant which extends the previous expressions over the complete turbulent boundary layer. Combining equations (4-25) and (4-29) and replacing the $l_i^2$ term in equation (4-27) with the modified mixing length of equation (4-29):

$$\left( e_m \right)_i = (\kappa y)^2 \left[ 1 - \exp(-y/A) \right]^2 \left| \partial u/\partial y \right| \tag{4-30}$$

again over the range $y_o \leq y \leq y_c$. For the outer region equation (4-28) still applies.

Equation (4-30) was obtained for a flat-plate flow experiencing no mass transfer, heat transfer or strong pressure gradient. CEBECI [30] has extended the eddy-viscosity model to flows experiencing these phenomena. The theory outlined in this section excludes the effect of mass transfer since the airfoil under study is nonporous.

The key non-dimensionalized equations of the theory found in CEBECI [30] are summarized below for the Cebeci/Smith eddy-viscosity model. Introducing the following non-dimensional variables:
\[ \hat{u} = U/U_\infty \quad \hat{v} = V/U_\infty \quad \hat{y} = y/c \quad \hat{x} = x/c \quad \hat{e}_m = e_m/(U_\infty c) \quad \hat{u}_\tau = u_\tau/U_\infty \]

\[ u'v'/U_\infty^2 = -\hat{e}_m \frac{\partial U/U_\infty}{\partial (y/c)} \quad (4-31) \]

\[ (\hat{e}_m)_1 = L^2 \gamma \frac{\partial U/U_\infty}{\partial (y/c)} \quad (e_m)_{\text{inner}} \leq (e_m)_{\text{outer}} \quad (4-32) \]

\[ (\hat{e}_m)_o = \alpha \gamma \int_{y/c}^{\infty} (U_\infty/U_\infty - U/U_\infty) \, d(y/c) \quad (4-33) \]

where \[ \alpha = 0.0168 \quad \text{since} \quad R_\theta \geq 5000 \]

\[ L = 0.4 \, (y/c) \left[ 1 - \exp\left(-\frac{(y/c)}{A}\right) \right] \quad (4-34) \]

\[ A = \frac{26}{N} \, \text{Re}^{-1} \, \hat{u}_\tau^{-1} \quad (4-35) \]

\[ \hat{u}_\tau = \left[ \left( \frac{\partial U/U_\infty}{\partial (y/c)} \right)_{w} \text{Re}^{-1} \right]^{-1/2} \quad (4-36) \]

\[ N = \left( 1 - 11.8 \, p^+ \right)^{1/2} \quad (4-37) \]

\[ p^+ = -\left( \frac{1}{(\text{Re} \, \hat{u}_\tau^3)} \right) (-1/2) \left( \frac{\partial C_p}{\partial (x/c)} \right) \quad (4-38) \]

The intermittency expression, \( \gamma \), is based on KLEBANOFF'S [35] results in a constant pressure boundary layer. It is defined by

\[ \gamma = 0.5 \left[ 1 - \text{erf} \, 5(y/\delta-0.78) \right] \quad (4-39) \]
Equation (4-36) was replaced by the following for ease of calculation and consistency of results:

\[ \hat{u}_r = \left[ \frac{u_s}{U_\infty} \right] \quad (4-40) \]

where \( u_s \) = friction velocity as determined from the Coles' Law of the Wall fit as described in Section 4.5.

A fourth order polynomial least squares fit to the measured mean \( U \) data was utilized in the application of this theory. The required integrals or derivatives were easily calculated once the coefficients of the velocity profile were known. The term \( \partial(C_p)/\partial(x/c) \) was calculated from the AADP9 surface pressure prediction as a result of the absence of pressure taps over the last few percent chord of the model. The validity of this choice is discussed in Section 5.6. The results of the eddy-viscosity analysis may be found in Section 5.5.
5.0 DISCUSSION OF RESULTS

5.1 SURFACE PRESSURE MEASUREMENTS

This investigation was performed at a Mach number of 0.14 and Reynolds number of 1.7 million based on chord. The airfoil characteristics were determined for a corrected angle of incidence of 0.002 degrees. The boundary layer was artificially tripped at 0.075c in order to enhance the development of the turbulent boundary layer. The location of the tripping device was chosen to maintain its effectiveness in future tests at other angles of incidence. Figure 10 details a typical surface pressure profile of the airfoil. The effect of the wind tunnel wall corrections as described in Section 4.2 is also shown in Figure 10. At this angle of incidence the airfoil is found to be strongly aft-loaded with an adverse pressure gradient starting around the 0.53c location. This was also predicted by a computer code (AADP9) developed at UTIAS as described in Section 5.6.
5.2 LIFT, MOMENT AND DRAG

The lift, moment, and drag for the airfoil were calculated through integrating a cubic spline fit of the data. For this configuration \( C_L = 0.517 \), \( C_{\mu/4} = -0.128 \) and \( C_{DP} = 0.00879 \) and \( C_{D_{total}} = 0.01245 \). Compared to the results of NELSON [31], the coefficients of lift and quarter chord moment are lower than those with free transition by approximately 0.05 and 0.0012 respectively. The drag coefficient increased by close to 50%. As a matter of interest, the experimental profile drag measurement was compared to the Squire-Young law drag calculation at \( \alpha = 0.002^\circ \). The equation is applicable in the near wake and is defined as:

\[
C_d = 2 \left( \frac{H_{te} + 5/2}{H_{te} + 5} \right) + 2 \left( \frac{H_{te} + 5/2}{H_{te} + 5} \right)
\]

(5-56)

\[
\hat{\theta}_{te} = \frac{\theta_{te}}{c} \quad U_{te} = U_{te}/U_{\infty} \quad H = \theta/\delta^n
\]

(5-57) (5-58) (5-59)

Since no lower surface boundary layer profiles were examined the effect of using the last known profile on the upper surface and the nearest wake profile for the lower profile was compared to values calculated at two near wake positions. The values calculated with x-wire profiles are listed in Table 2. The value calculated for the 0.998c upper surface boundary layer combined with the lower portion of the 1.005c wake profile compares favorably with the measured profile drag value of 0.01245. The effect of the strong pressure
gradient near the trailing edge can be seen to inflate the Squire-Young calculated drag. Further downstream at 1.015c the values tend to approach the measured value.

5.3 WAKE PROFILES

The x-wire mean streamwise velocity profiles over the region X/C = 1.005 to X/C = 1.30 are found in Figures 11-13. The values are an average of 10-12 D.C. voltage samples of the anemometer output taken at each point in the wake traverse. The upper limit of 12 samples is a result of the capability of the data acquisition system. Each traverse was taken normal to the chord line of the airfoil and normalized by a reference velocity as opposed to an edge velocity like that of the boundary layer profiles as discussed in Section 5.4. Further, the rotation of the coordinate system from tangential to the airfoil surface to tangential to the chord line makes direct comparison of the wake profiles and boundary layer profiles difficult.

Interestingly, not only does the asymmetry of the wake essentially disappear beyond X/C = 1.075 but the wake width remains approximately constant up until X/C = 1.075. Figure 14 reveals the development of the minimum wake velocity. Results from HAH [6], NAKAYAMA [8], NELSON [31] and YU [7] are included along with the current results. It can be seen that the current results and those
of NAKAYAMA [8] and NELSON [31] are effectively parallel over the region $0.03 \leq (X-C)/C \leq 0.20$. Further, the modified deHavilland airfoil results indicate a slower minimum wake velocity development than NELSON'S [31] results. The results of NELSON [31] detail the wake characteristics of free transition as opposed to artificial transition of the boundary layer. Therefore, the development of the minimum wake velocity for the present study would take longer due to the thicker boundary layer. It is also interesting to see that the minimum wake velocity develops logarithmically in the region bounded by the very turbulent, asymmetric near wake and the symmetric, far wake. In the very near wake the upper and lower surface boundary layers are combining in a highly energetic and complex manner, thus, a log relationship describing the development of the minimum wake velocity is not valid. In the far wake, where the streamwise velocity profiles become symmetric, the flow characteristics are asymptotically approaching the free stream conditions and, again, the log relationship is not valid.

Detailed in Figure 15 and 16 are the transverse mean velocity profiles over the range $1.002 \leq X/C \leq 1.30$. In the near wake at $X/C=1.002$ the upper wake profile exhibits the same basic shape of the V boundary layer profiles (see Figures 27-28) except that the direction of flow is reversed. The fact that the mean streamwise profile ($U$) are seen to slowly translate below the airfoil chord line supports the small mean transverse component shown in Figures 15-16. It is interesting to note that the basic profile shape does not change
substantially up to $X/C = 1.05$ at which point the asymmetry becomes less pronounced as the $V$ component approaches zero in the far wake ($X/C = 1.30$).

The streamwise turbulence intensity wake profiles found in Figures 17 to 19 reveal some interesting characteristics of the turbulence structures. In the near wake up to $X/C = 1.015$ the profiles contain two local minima on either side of a narrow peak. Further, the artificial tripping of the boundary layer has created a rather well developed or uniform distribution across the central portions of the upper and lower wake profiles. The upper minimum corresponds to the location of the maximum wake defect of $U$. In other words, the streamwise r.m.s. velocity ($u'$) is a minimum at $y/c = 0$ where $\partial U/\partial y = 0$. While the lower minimum is associated with the mean trailing edge streamline. The peak shown in the near wake is a consequence of the large wall shear at trailing edge transported to the wake. As the wake progresses downstream to $X/C = 1.05$, the minimum associated with the trailing edge streamline ceases to be an important factor in the flow and the peak has widened and lost the sharp definition of earlier profiles. By the time the wake reaches $X/C = 1.10$ the turbulence intensity profile is essentially symmetric. As of $X/C = 1.30$ the overall magnitude of the profile decreases slightly and the location of the minimum $u'$ value is difficult to detect as evidenced by the difficulty in determining the location of $\partial U/\partial y = 0$ in the mean velocity profile.
The presence of the wind tunnel walls prevents the turbulence structure from being completely homogeneous as evident from the cross-stream turbulence intensity profiles of Figures 20-21. In the near wake up to $X/C = 1.015$ the transverse r.m.s. velocity, $(v')$, exhibits approximately same basic shape as that of the $u'$ distribution but is approximately $1/3$ the magnitude. Again, two local minima and a maximum are present up to $X/C = 1.015$. The upper minimum is associated with the $\partial U/\partial y = 0$ point and progresses downstream up to $X/C = 1.30$. The peak evident in the near wake quickly disappears and the profile becomes essentially symmetric by $X/C = 1.10$.

The near wake Reynolds shear stress profiles of Figure 22 reveal, as expected, a region of high shear stress in the region where the upper and lower boundary layers combine. By $X/C = 1.05$ the mixing region has expanded and become less severe. The familiar 'S' profile is evident by $X/C = 1.30$.

5.4 BOUNDARY LAYER PROFILES

Cross-wire anemometer measurements of the mean velocity components ($U,V$), turbulence intensity, and shear stress profiles are exhibited in Figures 24-37. As detailed in Section 4.6 the mean velocity was determined from the voltage fluctuation about the mean voltage output. Each boundary layer profile was taken normal to the
surface as described in Sections 2.4 and 3.2 and normalized by an edge velocity. The mean streamwise velocity profiles for the last ten percent of the upper surface boundary layer are shown in Figures 24-26. The linear trend of the profile is believed to be a result of the artificial tripping of the boundary layer. It is expected that this type of profile is indicative of the type to be seen on longer chord models. The boundary layer is seen to be fully developed as shown by the uniformity or flatness of the turbulence intensity profiles of Figures 29-31. YU [7] tripped the boundary layer on a symmetric airfoil but the same linear trend is present. The mean streamwise velocity profiles of NAKAYAMA'S [8] results also exhibit the same linear trend.

On the airfoil currently under study, the boundary layer can be seen to grow by approximately 50% in thickness by the X/C =0.998 location over that X/C =0.97 due to the surface curvature and adverse pressure gradient. (See Figure 38 for displacement thickness vs X/C). Recalling the coordinate system rotation upon departure from the trailing edge, the near wake U upper wake profile can be seen to be similar to the X/C = 0.998 boundary layer profile which implies that the mixing region of the upper and lower boundary layer is constrained to a small region as evidenced by the center line peak. The cross-stream mean velocity, (V/U_e), in Figures 27-28 exhibit the same linear trend of the (U/U_e) component. From X/C= 0.96 to X/C =0.998 the V velocity increases from 0.10 to approximately 0.25 of the edge velocity.
Presented in Figures 30-31 is the streamwise turbulence intensity over the range $0.90 \leq X/C \leq 0.998$. These profiles reveal the steady growth of the boundary layer and that the peak turbulence intensity is relatively constant at $0.10 \bar{U}_e$ over $0.90 \leq X/C \leq 0.98$. After $X/C = 0.98$ the peak intensity increases slightly but it should be noted that $U_{edge}$ is also decreasing by approximately 3% over the last 0.1c. The transverse turbulence intensity is plotted as against $(y/c)$ in Figures 33-35. The basic shape of the $u'$ profile is evident in the $v'$ profiles but the peak values are no more than 25% of the streamwise turbulence intensity peaks. The Reynolds shear stress profiles of Figures 35-37 reveal that the peak of $u'v'$ at $\alpha = 0.002^\circ$ remains essentially constant until $X/C = 0.98$ after which it gradually increases to $-0.004 \bar{U}_e^2$ at $X/C = 0.998$. This increase is a result of the strong adverse pressure gradient at the trailing edge.

Displacement thickness and momentum thickness were integrated directly from the boundary layer profiles as described in Section 4.4. The values obtained, along with those from the wake, are found in Figure 38. The upper surface displacement thickness, $\delta^*$, increases very rapidly in the last 0.1c as does the momentum thickness, $\theta$. While in the wake the values decay even more rapidly. Nakayama's model B [8] shows a similar growth and decay trend but his peak values for $\delta^*$, $\theta$ and shape factor, $H$, are all greater than the present values of 0.0106, 0.00438 and 2.43 respectively. Possible reasons for the discrepancy include
Nakayama's higher angle of incidence (4° vs 0.002°) and lower Reynolds Number promoting a thicker boundary layer. It should be repeated that in the wake the upper and lower \( \delta^* \) and \( \theta \) were calculated individually by dividing the wake at maximum defect point and using the respective edge velocities. The lower wake \( \delta^* \) and \( \theta \) exhibit rapid growth and decay between \( 1.0 \leq X/C \leq 1.10 \).

Far downstream the wake must become symmetric so that the upper or lower \( \delta^* \) and \( \theta \) should become identical. Also, the shape factor \( H \) must approach 1.0 far downstream. This is just what happens with \( H \) decreasing from 1.26 to 1.15 over \( 1.2 \leq X/C \leq 1.45 \). To determine the asymptotic thickness value the measured profile pitot-static drag was used in:

\[
C_D = 2 \frac{\theta_\infty}{c}
\]

from momentum theory where \( \theta_\infty \) is the momentum thickness far downstream of the trailing edge and \( \theta_\infty = 2\theta_u = 2\theta_1 \). A \( C_D = 0.01245 \) leads to \( \theta_u/c = \theta_1/c = 0.00311 \) which is shown as a solid line in Figure 38.

Estimation of the skin friction coefficient \( C_f \) was accomplished by a least squares fit of the cross-wire anemometry results over the inner boundary layer (50 \( \leq y^+ \leq 250 \)) to Coles' Law of the Wall [20] followed by an evaluation of equation (4-21). The boundary layer profiles are presented in wall coordinates, \( u^+ \) and \( y^+ \), in Figure 39. The accuracy of the data fit to the law of the wall equation,
\[ u^+ = \left(\frac{1}{0.41}\right) \ln y^+ + 5.0 \]  

as shown as the solid line in Figure 39 and described in Section 4.5, may be identified as reasonable up to \( X/C = 0.98 \). As the flow approaches the trailing edge the agreement between theory and experiment decreases to the extent that doubt is cast upon the accuracy of the skin friction coefficients reported in Table 3 for \( X/C > 0.98 \). The fit of the law of the current test is similar to NAKAYAMA's [8] model 'B' results which also show that the logarithmic portion of the boundary layer fit decreases as the flow approaches the trailing edge. This could be a result of the law of the wall failing to account for the effects of a substantial adverse trailing edge pressure gradient combined with considerable surface curvature.

5.5 COMPARISON TO EDDY-VISCOSITY MODEL

The theory for the CEBECI/SMITH [30] eddy-viscosity model is outlined and reviewed in Section 4.7. Figure 40 presents the predicted \( u'v' \) profiles versus \( (y/c) \) for \( X/C = 0.97, 0.98 \) and 0.99 up to \( y/c = \delta/c \). The eddy-viscosity theory is applicable to many flows including those with mass transfer or pressure gradients. However, predictions of the experimental Reynolds shear stress are not as accurate for flows experiencing strong adverse pressure gradients (such as a boundary layer close to separation) or the opposite, that
is, a strong favorable pressure gradient. The theoretical profiles of Figure 40 are shown up to \( y = \delta \) because above this position the intermittency factor \( \gamma \) is approaching zero very rapidly resulting in \( u'v' \) equaling essentially zero. The eddy viscosity theory was found to predict values of \( u'v' \) approximately 2.4-2.6 times greater than the measured experimental values. This does not imply that the CEBECI/SMITH eddy-viscosity model is inaccurate. Except for the difference in magnitude, the theory accurately predicts the Reynolds shear stress in the upper 70% of the boundary layer. A discrepancy occurs below \( y/c = 0.005 \). This discrepancy is expected to be a result of the inability of the theory to account for the pressure gradient at the trailing edge. Further, although not evidenced by the mean boundary layer profiles the instantaneous velocity could be reversing in direction since in the region close to the airfoil surface the streamwise turbulence intensity is \( \geq 0.3U_e \). This implies that instantaneous separation and re-attachment may be occurring near the trailing edge thus the eddy-viscosity model would not accurately predict the \( u'v' \) profiles.

NAKAYAMA's [9] boundary layer profiles also show a region where the streamwise turbulence intensity is \( \geq 0.3U_e \), however, the mean streamwise velocity boundary layer profiles do not exhibit a profile indicative of separation. The over prediction by the eddy-viscosity theory is a result of the use of an eddy-viscosity coefficient that is assumed to be constant (\( \alpha = 0.0168 \)) and independent of chord location. NAKAYAMA's conventional airfoil (model A) results
essentially support the use of a constant coefficient. However, model A upper and lower surfaces were almost symmetric so that the boundary layer and wake flow experienced very mild streamline curvature or pressure gradients especially at the angle of incidence of zero degrees. In the current case and NAKAYAMA's model B the airfoil upper and lower surfaces near the trailing are not symmetric. NAKAYAMA found that the extreme flow curvature and pressure gradients caused the eddy-viscosity coefficient to decrease from 0.0168 far upstream of the trailing edge to 0.003 at $x/c = 0.99$. This tendency to decrease is also shown in the current test results. The results between the two tests can not be expected to quantitatively agree because of the differences in the models, lower angle of incidence, and higher Reynolds number of this investigation. As was said, except for the difference in magnitude the eddy-viscosity model accurately predicts the general shape of the Reynolds shear stress in the boundary layer. However, for flows experiencing large streamline curvature or pressure gradients perhaps a variable empirical eddy-viscosity constant is in order.

5.6 COMPARISON TO AADP9 COMPUTER CODE

The computer code AADP9 [11] was developed at UTIAS over a period of years. This proven code incorporates Thwaites' laminar boundary layer calculation and two different turbulent boundary layer calculation methods, namely, Green's integral method and the
Keller/Cebeci finite difference method. For the purpose of the comparison both methods were used. The various program tolerances were set to their default values and both programs began with the same starting conditions as determined by the experimental conditions. The airfoil coordinates together with angle of incidence, Reynolds number, Mach number and viscosity were all set to their respective experimental values. Curvature corrections were employed for the turbulent boundary layer calculation but wake modelling was omitted.

Table 4 details the comparison between the experimental and theoretical parameters. The accuracy of Green's method in predicting $C_L$, $C_D$, and $C_{Mc/A}$ is reasonable. The Keller-Cebeci finite difference method predicted separation of the boundary layer a few percent before the trailing edge. As a result, the accuracy of the predicted $C_L$, $C_D$ and $C_{Mc/A}$ is not as high as the Green's boundary layer method. The Keller-Cebeci results are not presented since the mean boundary layer profiles of Section 5.4 do not seem to indicate the predicted separation. The surface pressure distribution as predicted by AADP9 provides a reasonable representation of the actual distribution. As shown in Figure 41 the most favourable agreement occurs at the leading edge and last 0.4c on the upper surface. The experimental displacement thickness and momentum thickness are compared to the predicted values in Figure 42. The theoretical predictions reveal a similar trend to the experimental results, however, the predictions are greater than the measured results for both thicknesses. Over the
last 0.06c, the code predicted skin friction coefficients very close to the experimental values as seen in Figure 44. However, the experimental coefficients are believed to be in error, that is, too high as a result of the poor Law of the Wall fit past $X/C = 0.98$.

Possessing the knowledge of where the boundary layer transition occurs improves the overall accuracy of AADP9 in predicting actual parameters as based on the results of Table 4 and Figures 41-43.

5.7 ESTIMATION OF UNCERTAINTY

For accurate evaluation of computer model predictions an estimation of the uncertainties of any experimental data must be known. In this section consideration will be given to the characteristics of the equipment and analysis methods.

The wind tunnel facility at UTIAS exhibits a free stream root mean square turbulence intensity of less than 0.1%. The variation in velocity across the test section was generally less than 0.5% of the center line velocity. Mild surging of less than 0.5% of the dynamic pressure was present. Finally, the temperature variation of the flow stream was less than 1 °C as a result of three factors, namely, the time of day the tests were conducted, evacuation and replenishment of the quiescent air in the intake plenum before beginning a test and the open circuit design of the tunnel.
As described in Section 2.2 the geometric angle of attack of the airfoil was accurately determined with respect to the mean flow through the use of a cylinder possessing a common mounting block with the airfoil. Due to their physical dimensions the airfoil surface pressure taps occasionally became blocked or began to leak slowly. To correct this problem periodic examination of all connections and lines was carried out and blockages were removed or connections adjusted as needed. Finally, the fact that the airfoil was not instrumented with pressure taps over the last few percent chord contributed to the uncertainty of the lift and moment coefficients. An estimation of the uncertainty would be 0.01 for $C_L$ and 0.002 for $C_{Mq}$. 

The electronic equipment also contributed some uncertainty to the results gathered. In the case of the SD375 analog to digital converter the uncertainty was small at a standard deviation of 0.014% over the range of voltages converted. The difference between the two channels was less than 0.05% of the measured or mean voltage. The pressure transducer used for wake and surface pressure measurements possessed excellent linearity with an uncertainty of 0.08%. Due to the previously mentioned mild surging a number of samples were taken at any tap location and the results averaged. The typical standard deviation of the ensemble average was 0.4% of $q$.

Constant temperature anemometer measurements are also susceptible of possessing uncertainty from various sources. The
equipment was allowed ample warm-up time prior to the start of any test so as to prevent the uncertainties of thermal drift. The 55P63 x-wires were replaced and calibrated often to minimize the effects of probe contamination or degradation. The changes in ambient temperature of the flow was less than 1 °C which according to BEARMAN [17] maintains the errors in the mean velocity to about 1% and even less than 1% in the turbulent quantities. Due to the magnitude of the turbulence intensity over that of the empty tunnel, the signals were not filtered in order to prevent colouring of the data. Each channel of the x-wire system was appropriately tuned thus preventing electronic oscillations from occurring. The occasional frequency spectrums taken close to the wake center line, did not show any spikes, thus, indicating eddy shedding off the probe and probe vibration were not present.

An estimation of the uncertainty for the hot wire measurements would be <2% on the mean normal component of either wire and 6% on the turbulence quantities. This leads to an estimate of uncertainty of ≤12% on U'V'. The hot wire signal was digitized at 50 KHz. Typically 10-12 data blocks of 1024 bytes were taken at any point. Averaging was completed within each block and then an ensemble average of the blocks was calculated. These uncertainties could be reduced by taking longer time samples for each data block and increasing the number of blocks taken. Although the boundary layer mean velocity profiles do not indicate boundary layer separation periodic instantaneous separation may be occurring due to the u'
components being $\geq 0.3$ of $U_e$ in some regions of the boundary layer profile.

Finally, analysis of the drag coefficient, $C_D$, displacement thicknesses, $\delta^+$, and momentum thickness, $\theta$, possess some uncertainty especially in the wake. This is a result of the difficulty in determining the edge of the wake or the center line of the wake, due to the step resolution. An estimation of the uncertainty in $\delta/c$ and $\theta/c$ was 10% as determined by varying the location of the boundary layer edge or wake center line. This error of 10% is not insurmountable and neither are the previous errors. A step that could be taken to reduce the uncertainties would be the use of an improved data collection scheme. The use of an A/D converter with a higher sampling rate over that of the current converter would allow for an increased number of samples to be taken at any location in the flow, as well as, increase the number of locations investigated in the flow.
6.0 CONCLUSIONS AND RECOMMENDATIONS

With the ever escalating costs involved with experimental investigations of flight tests, the use of computers to numerically simulate flows is becoming more attractive to industry. In order to achieve accurate numerical models the nature of the phenomena must be fully understood. This investigation has attempted to provide insight into the flow characteristics about the trailing edge of an aft-loaded airfoil. The model was tested at a Reynolds number of 1.7 million, a Mach number of 0.14 and restricted to zero incidence. Artificial tripping of the boundary layer was employed at 0.075c to provide a well defined transition location for numerical analysis over that of free transition. A lift coefficient of 0.517 was determined through the integration of the surface pressure profile. The drag coefficient was found to be 0.01245 upon integration of the momentum loss of the mean flow across the wake width.

Measurements of the mean velocity profiles in the upper surface boundary layer and near/far wake were collected through the use of an x-wire anemometer system. The profiles were integrated to provide the momentum and displacement thickness. Over the last 3% chord of the airfoil the boundary layer was seen to grow by approximately 50% in thickness. This is due to the surface curvature and adverse pressure gradient. The asymmetric near and far wake streamwise
turbulence intensity profiles essentially become symmetric by X/C=1.10. The streamwise and transverse turbulence intensity exhibited similar profile shapes in both the wake and boundary layer. The transverse turbulence intensity was discovered to be approximately one-third and one-quarter the magnitude of the streamwise turbulence intensity profiles in the wake and boundary layer respectively.

This study has revealed only a fraction of the total area of interest surrounding trailing edge flows. So with great interest the following recommendations concerning further study are presented.

Consideration should be given to investigating the effect of the angle of incidence on the boundary layer structure due to free transition and artificial transition. Further, the effectiveness of the tripping device with regards to the size and location and its consequences on the drag of the airfoil warrant investigation. Boundary layer traverses on the lower surface of the airfoil would also contribute to a greater understanding of the flow development. Each of these avenues of study could be carried out for various flow speeds and Reynolds numbers. It may be seen that a tremendous amount of data could be presented upon completion of the above studies.

It is suggested that the model be instrumented with pressure taps over the last few percent chord. This would allow for a more accurate determination of lift and pitching moment coefficients. The
drag of the airfoil could be determined through the use of a wake rake as opposed to a single center span traverse. The two dimensionality of the flow could be exhaustively determined through the use of an x-wire located in the y-z plane or by flow visualization. Improved spatial resolution could be achieved through the use of a smaller x-wire probe or a non-intrusive laser Doppler anemometry system.

Positioning accuracy of the hot wires in the boundary layer with respect to the airfoil surface could improved by utilizing an automated rotation device over manual wedges. More stringent hot-wire calibrations could be achieved through the use of a multi-channel A/D converter to record the linear and non-linear signals in combination with the flow temperature. Further, the possible errors associated with a limited number of signal samples could be effectively eliminated through the use of a more sophisticated data collection scheme. With the use of a rapid sampling rate hot-wire output voltage samples could be taken until the mean of the total ensembles varies by less than a specific tolerance over the (N-1) population sample. Finally, although computationally expensive, an exhaustive utilization of the AADP9 code and other codes for comparison of theoretical predictions to the experimental results of the suggested studies is of interest.
7.0 LIST OF WORKS CONSULTED


### TABLE 1: EFFECT OF WIND TUNNEL WALL CORRECTIONS

<table>
<thead>
<tr>
<th>EXPERIMENT</th>
<th>$\alpha$</th>
<th>$\text{Rec} \times 10^6$</th>
<th>$C_L$</th>
<th>$C_M c/4$</th>
<th>$C_L$</th>
<th>$C_M c/4$</th>
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</thead>
<tbody>
<tr>
<td>A</td>
<td>0.002$^\circ$</td>
<td>1.73</td>
<td>0.5432</td>
<td>-0.1338</td>
<td>0.5179</td>
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<tr>
<td>B</td>
<td>0.001$^\circ$</td>
<td>1.73</td>
<td>0.5342</td>
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### TABLE 2: EFFECT OF LOCATION ON EVALUATION OF SQUIRE-YOUNG DRAG

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<th>LOCATION (x/c)</th>
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<td>$C_D$</td>
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<td>0.01331</td>
<td>0.01302</td>
<td>0.01245</td>
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</table>

TABLE 2: EFFECT OF LOCATION ON EVALUATION OF SQUIRE-YOUNG DRAG
### Table 3: Boundary Layer Parameters

<table>
<thead>
<tr>
<th>Location (x/c)</th>
<th>( \delta^*/c )</th>
<th>( \theta/c )</th>
<th>( H )</th>
<th>( C_f )</th>
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<tr>
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<td>0.01060</td>
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<td>2.426</td>
<td>0.00061</td>
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</table>

### Table 4: Comparison of Experimental \( C_L \), \( C_D \), \( C_{M/L4} \) to AADP9 Predicted Coefficients

<table>
<thead>
<tr>
<th>Type</th>
<th>( \alpha )</th>
<th>( C_L )</th>
<th>( C_D )</th>
<th>( C_{M/L4} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>Experimental</td>
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<td>0.0119</td>
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<td>0.528</td>
<td>0.0120</td>
<td>-0.132</td>
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<td>AADP9 Keller-Cebeci</td>
<td>0.002°</td>
<td>0.544</td>
<td>0.0102</td>
<td>-0.135</td>
</tr>
</tbody>
</table>
Figure 1 Wind Tunnel Configuration
Figure 2  Honeycomb and Screen Geometry
Figure 4  Pressure Data Acquisition System Block Diagram
FIGURE 5  GENERAL TRAVERSE GEOMETRY FOR DATA COLLECTION
Figure 6  Hot Wire Data Acquisition System Block Diagram
Figure 7  Typical Calibration for the A/D Converter and Pressure Transducer
Figure 8  Typical Hot Wire Calibration
GENERAL 3 DIMENSIONAL FLOW FIELD

FIGURE 9 GEOMETRY FOR X-WIRE THEORY
Figure 10  Effect of Wind Tunnel Boundary Corrections on the Experimental Surface Pressure Profile
Figure 11  Mean Streamwise Velocity Profiles in the Near Wake
Figure 12  Mean Streamwise Velocity Profiles in the Wake
Figure 13  Mean Streamwise Velocity Profiles in the Wake
Figure 14  Development of Minimum Wake Velocity
Figure 15  Mean Transverse Velocity Profile in the Near Wake
Figure 16  Mean Transverse Velocity Profile in the Wake
Figure 17 Streamwise Turbulence Intensity Profiles in the Near Wake
Figure 18  Streamwise Turbulence Intensity Profiles in the Wake
Figure 19  Streamwise Turbulence Intensity Profiles in the Wake
Figure 20  Transverse Turbulence Intensity Profiles in the Near Wake
Figure 21  Transverse Turbulence Intensity Profiles in the Wake
Figure 22  Reynolds' Shear Stress Profiles in the Near Wake
Figure 23  Reynolds' Shear Stress Profiles in the Wake
Figure 26  Boundary Layer Streamwise Velocity Profiles
Figure 30  Boundary Layer Streamwise Turbulence Intensity Profiles
Figure 31  Boundary Layer Streamwise Turbulence Intensity Profiles
Figure 33  Boundary Layer Transverse Turbulence Intensity Profiles
Figure 34

Boundary Layer Transverse Turbulence Intensity Profiles
Figure 35  Reynolds' Shear Stress Profile in the Boundary Layer
Figure 36  Reynolds' Shear Stress Profile in the Boundary Layer
Figure 38  Measured Displacement and Momentum Thickness
Figure 39  Boundary Layer Profiles in Wall Coordinates
Figure 41 Comparison of Experimental Surface Pressure Distribution at $\alpha = 0.002$ with AADP9
Figure 42  Comparison of Experimental Displacement and Momentum Thicknesses with AADP9
Figure 43  Comparison of Experimental Skin Friction Coefficient on the Upper Surface with AADP9
The mandate of this report was to determine the mean and turbulent flow quantities of an attached, fully-developed boundary layer of an aft-loaded airfoil. The steady, two dimensional flow characteristics of the near and far wake were also studied. This was done to gain an understanding of the viscous effects near the trailing edge of the airfoil in the hope of providing accurate data for airfoil performance models.

Tests were conducted at a Reynolds number of 1.7 million based on chord and a Mach number of 0.14. The airfoil model on loan from Boeing of Canada's deHavilland Division was restricted to zero incidence and the boundary layer was artificially tripped at 0.75c. Artificial tripping was employed to provide a well defined transition location over that of free transition for numerical analysis.

The hot-wire anemometer measurements taken in the boundary layer and wake near the trailing edge reveal that a complex interaction takes place just aft of the airfoil where the upper and lower surface flows combine. The boundary layer thickness was found to increase quickly over the last few percent chord with a corresponding increase in turbulence intensity. The extreme asymmetry in the near wake profile was seen to transform into an essentially symmetric profile by approximately X/c = 1.075. Parameters such as skin friction coefficients, displacement and momentum thicknesses were determined from the x-wire profiles. Numerical predictions by the computer code ADAP of these parameters, together with the coefficients of lift, drag, and quarter chord moment, compared favourably with the experimental values.

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