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EXPERIMENTAL INVESTIGATION OF THE VORTEX STRUCTURE
OF A PROTOTYPE V-22 BLADE TIP

The members of the Committee approve the masters
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EXPERIMENTAL INVESTIGATION OF THE VORTEX STRUCTURE
OF A PROTOTYPE V-22 BLADE TIP

by

JENNIFER LARUE PEEPLES

Presented to the faculty of the Graduate School of
The University of Texas at Arlington in Partial Fulfillment
of the Requirements
for the Degree of

MASTER OF SCIENCE IN AEROSPACE ENGINEERING

THE UNIVERSITY OF TEXAS AT ARLINGTON

August 1997
ACKNOWLEDGEMENTS

I owe a special debt of gratitude to Dr. Mullins for his patience and assistance in completing this research. Dr. Mullins was involved in this project from the very beginning, and under his mentorship my confidence in my abilities grew along with my intellectual curiosity.

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Finally, I would like to thank my family, who taught me never to quit. And I especially want to thank Melanie, who believed in me even when I sometimes didn't.

July 11, 1997
ABSTRACT

EXPERIMENTAL INVESTIGATION OF THE VORTEX STRUCTURE
OF A PROTOTYPE V-22 BLADE TIP

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Jennifer LaRue Peeples, M.S.
The University of Texas at Arlington, 1997

Supervising Professor: Donald R. Wilson

The aerodynamic characteristics of an advanced helicopter rotor blade tip were experimentally determined using test conditions typically encountered by rotor tips (M = 0.80 with Re = 9.8 * 10^6). In the first phase of the test, stagnation pressure surveys of the airfoil wake were performed at two downstream chord stations (0.1 chord and 1.0 chord) and two angles of attack (0 and 3 degrees) using a wake rake. This yielded preliminary qualitative data in the form of pressure deficits in the wake of the airfoil. Quantitative data about the flow field in the wake were then obtained using a five-hole probe. Data from the probe were used to determine the velocity components in the vicinity of the pressure deficits previously identified. Finally, the lift, drag, and pitching moment coefficients were calculated using data obtained with a five component sidewall balance.
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<td>Arnold Engineering Development Center</td>
</tr>
<tr>
<td>ARC</td>
<td>Aerodynamics Research Center at UTA</td>
</tr>
<tr>
<td>BHTI</td>
<td>Bell Helicopter Textron, Inc.</td>
</tr>
<tr>
<td>BVI</td>
<td>blade vortex interaction</td>
</tr>
<tr>
<td>c</td>
<td>chord length</td>
</tr>
<tr>
<td>E</td>
<td>error</td>
</tr>
<tr>
<td>HIRT</td>
<td>High Reynolds Number Transonic tunnel</td>
</tr>
<tr>
<td>M</td>
<td>Mach number</td>
</tr>
<tr>
<td>n</td>
<td>number of bits used by analog to digital converter</td>
</tr>
<tr>
<td>P</td>
<td>pressure</td>
</tr>
<tr>
<td>psia</td>
<td>pounds per square inch absolute</td>
</tr>
<tr>
<td>Re</td>
<td>Reynolds number</td>
</tr>
<tr>
<td>U</td>
<td>uncertainty</td>
</tr>
<tr>
<td>UTA</td>
<td>The University of Texas at Arlington</td>
</tr>
<tr>
<td>x</td>
<td>spanwise coordinate, origin at airfoil tip, positive away from airfoil root</td>
</tr>
<tr>
<td>y</td>
<td>vertical coordinate, origin at quarter chord point of airfoil root chord</td>
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<td>downstream coordinate, origin at quarter chord point of airfoil root chord</td>
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<tr>
<td>A</td>
<td>amplifier</td>
</tr>
<tr>
<td>AD</td>
<td>analog to digital converter</td>
</tr>
<tr>
<td>d</td>
<td>design stage</td>
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<td>N</td>
<td>Nth order</td>
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<td>T</td>
<td>transducer</td>
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CHAPTER 1

INTRODUCTION

The characteristic sound of a helicopter’s rotor blades is produced by a phenomenon known as blade vortex interaction, or BVI. Helicopter manufacturers have an ongoing interest in the BVI phenomenon since it occurs continuously in helicopters, with the most pronounced effects occurring when the aircraft is in a powered descent (1). Transonic blade tip speeds, which represent the typical operating conditions for rotorcraft, are also known to exacerbate BVI effects (2).

BVI results when a rotor blade passes very near or through the tip vortex shed by the blade preceding it. In addition to producing an impulsive noise at the same frequency as the blade passage frequency, this interaction of a rotor blade with a vortex can result in locally transonic flows and shock formation. Boundary layer separation due to vortex interaction is also common and results in unsteady loading on the blade. This, of course, can seriously degrade the blade’s performance.

The interactions that contribute to the BVI phenomenon can be idealized into two types, each representing an extreme condition encountered by helicopter rotor blades. Both extremes, as well as intermediate interactions, are present in the flow field at all times.
The first of these limiting cases is known as the parallel interaction, which is treated as a two-dimensional, unsteady problem. Most theoretical and experimental investigations of BVI have focused on the parallel interaction since the unsteady interaction is believed to make the largest contribution to the noise field (1).

At the other extreme is the perpendicular interaction, which treats the BVI phenomenon as a three-dimensional, steady problem. The steady interaction does not contribute as much to the noise problem as does the unsteady, two-dimensional case, but can produce large variations in the pressure distribution over the airfoil. Most of the early investigations of the perpendicular problem were done at low speeds and were not representative of the conditions encountered by a helicopter rotor blade. Kalkhoran conducted one of the first experimental investigations of the perpendicular phenomenon at transonic speeds. His research documented substantial changes in the pressure distribution over an airfoil due to perpendicular vortex interaction at supercritical Mach numbers (3). Kalkhoran's work demonstrated the value of using The University of Texas at Arlington's High Reynolds Number Transonic (HIRT) facility to conduct BVI experiments in the transonic regime (3).

Two critical factors affect the severity of the BVI problem. These are the proximity of the rotor blade to the vortex core and the strength of the vortex. The possibility of altering these characteristics with the tip shape of a rotor blade was investigated by Rath (4) and Mullins, et al. (5). These studies involved a series of experiments con-
duced in The University of Texas at Arlington's (UTA) HIRT facility and documented substantial changes in the tip vortex shape and strength due to modifications in blade tip shape.

The noise generated by BVI is a concern for both military and civilian helicopters. Bell Helicopter Textron, Incorporated (BHTI) commissioned this study to investigate the vortex structure generated by an experimental blade tip developed for use on the V-22 Osprey Tiltrotor aircraft. The V-22 is a relatively quiet aircraft in "airplane" mode, but is extremely noisy when the prop-rotors are tilted up in "helicopter" mode. It is hoped that the experimental blade tip will modify the tip vortex sufficiently to reduce the BVI noise by 50%. Along with its military applications, this will open new possibilities for tiltrotor aircraft as civilian transports.
CHAPTER 2

THE TEST FACILITY

HIRT Operation and Upgrades

The UTA HIRT facility was originally placed into operation and its operational parameters defined at Arnold Engineering Development Center (AEDC) as a prototype for the National Transonic Facility. It was donated to UTA in 1978, where development of the required mechanical, pneumatic, and electrical subsystems continued until it was returned to service in 1984 (10).

The HIRT is a Ludwieg tube based on the work done by H. Ludwieg of Germany in 1957. The basic operational concept involves accelerating high pressure air to transonic speeds through an unsteady expansion process. Refer to figure 1 for a diagram of the HIRT facility.

The HIRT is unique in its capability to vary Mach and Reynolds number independently of each other. Mach and Reynolds numbers are dimensionless similarity parameters often used in modeling fluid flows. Mach number is the ratio of the free stream fluid velocity to the local speed of sound. Reynolds number is the ratio of inertial to viscous forces in the fluid.
To begin a HIRT run, high pressure air is used to charge the 33.8 m (111 ft) charge tube to the desired pressure level. The pressure used is dictated by the Reynolds number desired. Once the desired pressure level is reached, the flow of high pressure air is shut off and the pressures are allowed to stabilize. The expansion wave is initiated by opening a sliding sleeve valve. The expansion wave propagates upstream through the diffuser, test section, and nozzle.

![Diagram of HIRT Facility.](image)

A secondary flow through the plenum chamber is initiated by rupturing a mylar diaphragm, which results in a second expansion wave. During tunnel calibration, the time delay for the diaphragm cutter is set to ensure that both expansion waves reach the test section simultaneously. Proper coordination of the SSV opening and diaphragm rupture is critical to achieving the optimum steady flow period. Once the expansion waves reach the test section, they propagate upstream as a single wave. A period of steady flow
begins when the expansion wave clears the convergent nozzle and lasts for the time re-
quired for the wave to travel the length of the charge tube, reflect and return to the test 
section.

The theoretical maximum test window in 185 msec; however, this is reduced to 
about 120 msec due to the time required to open the SSV and the unsteady flow during 
initiation of the secondary expansion wave in the plenum chamber. Additional wave re-
flections result in one or more secondary test windows, each at approximately half the 
Reynolds number of the one preceding it and at the same Mach number. These subse-
quent reflections must be evaluated carefully before use, because the flow quality tends 
to deteriorate rapidly after the initial steady flow period.

The test section measures 18.5 cm (7.28 in) x 23.2 cm (9.15 in) x 64 cm (25.4 in) 
long. Its porous walls can be manually adjusted to provide a porosity of 3.5 to 10 per-
cent. This capability helps minimize shock wave reflections from the tunnel walls and 
also alleviate tunnel wall interference effects (11). Air is drawn through the walls and out 
of the plenum cavity by 8 hoses, each with an internal diameter of 2 inches. The hoses 
are connected to a manifold located just upstream of a Grayloc clamp, which holds the 
mylar diaphragm material. The diaphragm is ruptured by a cruciform cutter. The timing 
of the diaphragm rupture is determined during calibration.

The flow through the plenum chamber provides an additional means of mass re-
moval from the test section, which in turn allows acceleration of the flow to a maximum
Mach number of 1.2. Without this additional mass removal, normal operation of the HIRT would be limited to Mach 1.0 with the constant area nozzle. Figure 2 illustrates the flow through the plenum cavity. The Flex-flo valve shown in the figure can be used to shorten the starting transient in the tunnel, but was not used in this study.

![Diagram of Flow Through Test Section and Plenum Cavity](image)

Figure 2. Diagram of Flow Through Test Section and Plenum Cavity.

Downstream of the diaphragm is a variable orifice ball valve. The ball valve opening determines the Mach number. Thus, Mach and Reynolds numbers can be
independently varied throughout the operating range of the HIRT. Figure 3 illustrates the
HIRT Mach and Reynolds number capabilities.

![Graph showing Mach number vs. Reynolds number with various pressure levels indicated]

Figure 3. HIRT Facility Mach and Reynolds Number Capabilities.

Two sliding sleeve valves (SSVs) are available for use with the HIRT. The 12-
inch diameter valve was damaged in 1995, so the 16-inch diameter valve was used
throughout this study. The 16-inch diameter SSV has 27 ports, each consisting of a
short, threaded pipe nipple. The ports can be capped or left open to adjust the mass flow
through the SSV. Flow is exhausted through the SSV into an exhaust sphere and silencer. Figure 4 shows the key components of the test section, as well as the SSVs.
Figure 4. Test Section and Key Components.

The current study required use of a Reynolds number approximately twice that of previous tests performed in the HIRT. The use of such a high Reynolds number exacerbated the pressure measurement time lag problems noted previously (4). In fact, the time lag in the plenum pressure measurement system was so severe that the steady flow period could not be reliably identified, nor could accurate Mach numbers be determined. Figure 5 illustrates the pressure trace of the original system. Note that the pressure used in this trace was that for a much lower Reynolds number than the one used in this investigation. This pressure level was selected as the benchmark for trouble-shooting since a large database exists at this pressure.
Figure 5. System Response Before Modification.

The initial plan for improving the pressure measurement system response time called for shortening the connecting tubing between pressure probes and transducers as much as possible. For the total pressure probe, the tubing was shortened from 37.5 to 28 inches, while the static pressure probe tubing was shortened from 25.75 to 9.75 inches. In both cases, the internal diameter of the tubing was 1/16 inch.

The plenum pressure system was much more complicated. The original pressure probes were 14 inches long with an inside diameter of 0.180 inch and contained four 0.063 inch diameter static ports arrayed around the probe. The replacement probes were the same length, but the static port diameter was increased to 0.083 inch. The tubing connecting the probes to a single transducer was 0.042 inch I.D. tubing and was not
changed. This greatly improved the response time as figure 6 illustrates; however, the results were not repeatable. Examination of the probes revealed that debris from the clay used to fill the model attachment holes had partially blocked the connecting tubing.

![Pressure Plot](image)

**Figure 6. System Response With Shortened Tubing.**

The 0.042 inch I.D. tubing was then replaced with 1/16 inch I.D. tubing and each plenum pressure probe was connected to a separate transducer. This dramatically improved the response time of the system, and the uncertainty analysis contained in chapter 4 shows that time lag is now a minor contributor to uncertainty in plenum pressure measurement. Figure 7 illustrates the response of the system with all modifications and at the nominal charge tube pressure used in this investigation.
Figure 7. System Response After All Modifications.

As a final note, the increase in the plenum pressure beginning at approximately 240 msec and continuing to the end of the test window was a mystery until it was discovered that the rubber stops that control the rearward travel of the SSV had deteriorated. This allowed the SSV to open beyond the fully open position, thus partially blocking the orifices again. As a result, mass flow through the SSV was reduced and produced a corresponding rise in plenum cavity pressure. The reduced mass flow also resulted in a lower than predicted Mach number. The rubber stops were replaced, after which the plenum cavity pressure trace remained stable throughout the test window.
HIRT Calibration

The HIRT facility calibration involved an experimental determination of the time delays necessary to maximize the steady flow period in the test section. Improper, or less than optimal, setting of the time delays in the starting sub-systems can significantly shorten the testing window.

The test conditions specified for this experiment represent full-scale operational conditions of the V-22 blade tip. The HIRT facility was calibrated using these conditions. Table 1 shows the specified conditions.

<table>
<thead>
<tr>
<th>ITEM</th>
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<tbody>
<tr>
<td>Mach Number</td>
<td>0.78</td>
</tr>
<tr>
<td>Reynolds Number</td>
<td>9,810,000</td>
</tr>
<tr>
<td>Free Stream Pressure</td>
<td>107 psi (7.27 atm)</td>
</tr>
</tbody>
</table>

Once the subsystem time delays were set, the axial Mach number distribution within the test section was measured using a centerline probe. The centerline probe calibration is performed at the beginning of a test program and need not be performed again unless the tunnel operating conditions change substantially. The probe was mounted using removable mounts in the upstream end of the HIRT and in the model support section. Pressure taps on the centerline probe allowed accurate determination of the axial Mach number variation throughout the test section. Eleven pressure ports were selected
in the area where the airfoil model was to be mounted. A Mach number was computed for each of these static pressure readings and results are shown in figure 8.

![Test Section Mach Number Variation](image)

**Figure 8. Test Section Mach Number Variation.**

Note that the Mach number was easily varied by adjusting the plenum exhaust system ball valve setting. As was previously described, changing the Mach number has no effect on other HIRT parameters. The data scatter for the Mach number distribution is within ±0.50 percent. This is comparable to the data published by Starr and Schueler in the Arnold Engineering Development Center calibration (6).

In conjunction with the centerline probe calibration, a daily calibration of the pressure transducers was done using a Kulite XTS-1-190-200 pressure transducer as the master transducer. This is a temperature compensated pressure transducer and exhibits
very little drift over time. It is used as a reference signal to calibrate the other transducers in the system during the tunnel charging cycle. This procedure minimizes errors due to transducer drift.
CHAPTER 3

AIRFOIL MODEL STRUCTURAL ANALYSIS

Previous tip vortex studies done in the HIRT used 2-inch chord airfoil models; however, the complexity of the new tip shape investigated in this study necessitated the use of a 3-inch chord model. Since this airfoil would be subjected to Reynolds numbers nearly twice that of the models used in previous tests, a detailed analysis of the airfoil loading was completed before the airfoil model was fabricated. The model was fabricated by Raymond Mort of AVT Engineering. The technical drawings of the model are included in appendix B.

The initial data for the structural analysis was provided by BHTI as a series of chordwise pressure distributions at several spanwise locations (3). This information was generated with a commercial CFD code using the test conditions specified in table 2.

<table>
<thead>
<tr>
<th>ITEM</th>
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<tr>
<td>Mach</td>
<td>0.81</td>
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<tr>
<td>Reynolds Number</td>
<td>10,541,000</td>
</tr>
<tr>
<td>Free Stream Pressure</td>
<td>107 psi</td>
</tr>
<tr>
<td>Angle of Attack</td>
<td>4 degrees</td>
</tr>
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</table>
Note that these conditions are slightly different than the actual test conditions used (see table 1). Figure 9 shows the pressure distribution over the airfoil generated by the CFD code using the conditions in table 2.

![Tip Shape Pressure Distribution](image)

Figure 9. Tip Shape Pressure Distribution.

The stress analysis was based on a wind tunnel model constructed of H-1100 heat treated, 17-4PH stainless steel. This material is capable of withstanding stresses of up to 140 ksi. A finite element model was then constructed and analyzed using ALGOR, a commercial finite element analysis code. Several load cases were run, each representing a different pressure distribution from 1 to 10 atmospheres. Figure 10 shows the von Mises stress, or maximum stress in any direction, at the nominal load case of 7 atmospheres.
Figure 10. von Mises Stress at 7 Atmospheres.

Figure 11 shows the model displacement at the 7 atmosphere nominal test case.

Note that the scale factor for this figure is one.

Figure 11. Model Displacement at 7 Atmospheres.
Based on the stress analysis for the nominal 7 atmosphere condition, the factor of safety for the model was 1.6. Stress analysis at the maximum load case of 10 atmospheres was also done. This analysis showed a maximum von Mises stress of 128 ksi, which is well within the maximum allowable stress of 140 ksi.
CHAPTER 4

THE EXPERIMENTAL PROCEDURE

This program was funded by a grant from BHTI. The goal of the program was to test an advanced rotor tip shape planform in UTA's HIRT wind tunnel. The test program involved three phases: wake pressure deficit measurement at two downstream locations and two angles of attack using a 13 port wake rake, examination of the tip vortex core(s) at the same downstream locations and angles of attack using a calibrated five-hole probe, and measurement of lift, drag, and pitching moment using a five component sidewall balance.

Wake Pressure Deficit Measurements

Originally, BHTI asked for a series of runs at an angle of attack of 4° with pressure deficit maps at 1, 3, and 6 chord lengths downstream of the airfoil. The plan shown in table 1 represents a modification to the original plan to allow direct comparison with CFD data for this airfoil. The new plan deleted the requirement to obtain pressure deficit maps at 3 and 6 chord lengths downstream and added the requirement to map the flow field at 0.1 chord downstream. Again, this facilitated direct comparison with CFD data. Additionally, measurements of the HIRT test section revealed that the maximum separation possible between the airfoil trailing edge and the wake-rake was 15 inches, which is
only 5 chords. Thus, the 6 chord length measurements would have been impossible without a modification to the wake-rake mount.

Wake pressure measurements were obtained using a total pressure wake-rake. The wake-rake consists of 13 tubes, each separated by 0.125 inch. The wake-rake was positioned the required distance aft of the airfoil using mounting hardware and a traversing mechanism designed for this purpose. The traversing mechanism allowed rapid and accurate placement of the wake-rake during the test. The author then traversed the rake 0.125 inch for each run. This resulted in a 0.125 x 0.125 inch grid of total pressure measurements for each wake pressure deficit map. This yielded a relatively coarse grid, but prior work done by Rath (2) indicates that this is sufficient to obtain accurate information about vortex location and strength. The contract from BHTI also provided for additional runs using a smaller grid size to enhance data quality in any area of interest.

Five-Hole Probe Measurements

Total pressure surveys using a wake-rake are invaluable in determining the location and strength of vortices, but provide no detailed information about the flow beyond that. A five-hole cone probe provides much more detailed information about a flow field. The five-hole probe is a highly accurate pressure measurement device which can measure not only total pressure, but flow angularity as well. A five-hole probe requires a detailed calibration procedure before it can be used. The procedure used to calibrate the five-hole
probe used in this test as well as a detailed description of the probe is included in appendix A.

The five-hole probe surveys were conducted by mounting the probe in the wake of the airfoil at the same chord lengths downstream as in the wake-rake surveys. Wind tunnel tests were then performed with the probe traversed to a new location for each tunnel run. Use of the five-hole probe is more time consuming than the initial flow field mapping using the wake-rake. This is because the wake-rake can measure pressure in 13 locations during each tunnel run, whereas the five-hole probe can measure pressures in only one location. The probe was horizontally traversed using the traversing mechanism, but each change in the vertical position of the probe required a complete tunnel break. The BHTI contract provided for 60 tunnel runs to complete the five-hole probe surveys.

Five Component Sidewall Balance Test

The final phase of the test program required the use of a five component sidewall balance to measure the lift, drag, and pitching moment of the tip shape. The design loads of the five component sidewall balance are summarized in the table 3.

<table>
<thead>
<tr>
<th>COMPONENT</th>
<th>DESIGN LOAD</th>
</tr>
</thead>
<tbody>
<tr>
<td>Normal Force (NF)</td>
<td>700 lbs</td>
</tr>
<tr>
<td>Chord Force (CF)</td>
<td>75 lbs</td>
</tr>
<tr>
<td>Moment Due to NF (RBM-X)</td>
<td>1400 in-lbs</td>
</tr>
<tr>
<td>Moment Due to CF (RBM-Z)</td>
<td>150 in-lbs</td>
</tr>
<tr>
<td>Pitching Moment</td>
<td>140 in-lbs</td>
</tr>
</tbody>
</table>
The test plan called for the airfoil to be mounted on the five component sidewall balance, which can be pitched to place the airfoil at the desired angle of attack. A precision digital level was required to accurately pitch the airfoil in $1^\circ$ increments. The force-balance test required that the data from each run be reduced before proceeding to the next run to ensure that the balance limits were not exceeded. The initial plan called for a five component sidewall balance test at $-2^\circ$ to start, with the airfoil pitched up in $1^\circ$ increments. The test was to be terminated if the design load for any of the five components was exceeded during a run.
CHAPTER 5

EXPERIMENTAL RESULTS

The results of the wake rake surveys provided the gross analysis of the tip vortex strength and location for each angle of attack and chord position downstream of the trailing edge. Tip vortex strength is defined as the pressure deficit in the wake of the airfoil. This pressure deficit is expressed as the ratio of the stagnation pressure measured by the rake to the total pressure measured by the tunnel pressure measurement system. As Kalkhoran noted previously (3), the information obtained with the wake-rake is more qualitative than quantitative. Thus, a five-hole probe was used to obtain the qualitative data about the vortices. Once the pressure information was collected, the aerodynamic characteristics of $C_L$, $C_D$, and $C_M$ were determined using the five-component sidewall balance.

Figure 12 illustrates the coordinate system used in this study. The $z$-axis points in the same direction as the free stream flow, positive downstream. Measurements were nondimensionalized by dividing them by the chord length of 3 inches. Figure 13 shows a schematic of the model mounted in the tunnel test section with the test area identified.
Figure 12. Coordinate system axes.

Figure 13. Model mounted in wind tunnel with test area defined.
Wake-Rake Survey Results

The first wake-rake survey was conducted with the rake positioned at 0.1 chord downstream of the trailing edge and the airfoil pitched at 0°. Figure 14 shows the results of this survey. For this and all subsequent maps the trailing edge of the airfoil is located at \( y/c = 0.0 \), and \( x/c = 0.0 \) is the location of the tip of the airfoil.

![Pressure Distribution @ 0.1 Chord](image)

Figure 14. Pressure map at 0° pitch, 0.1 chord.

For this configuration, the vortex map shows no tip vortex. There are minor pressure deficits in the flow field, especially from \( x/c = -0.75 \) inboard. These deficits are representative of the wake formation.
The next configuration was also at 0° pitch, but this time the rake was positioned at 1.0 chord lengths downstream of the trailing edge. Figure 15 illustrates the results of this survey.

![Pressure Distribution at 1.0 Chord](image)

**Figure 15.** Pressure map at 0° pitch, 1.0 chord.

This map shows a well developed wake, as well as a strong tip vortex whose core is located at approximately x/c = -0.1. The pressure ratio in the vortex core is 0.79, which represents the pressure deficit and, hence, the strength of the vortex. Although BHTI did not make all of the CFD data for this airfoil available for the experimental phase, the author was able to confirm that a vortex of this strength was predicted by the CFD analysis (7).
Note that the wake appears to jump from its position on the trailing edge to approximately y/c = -0.5 between x/c = -0.9 and x/c = -0.8. This is where the geometry of the airfoil changes from a curved to a straight section. See appendix B for technical drawings of the airfoil.

The next survey was conducted with the airfoil pitched at 3° and the wake-rake fixed at the 0.1 chord position downstream of the trailing edge. Figure 16 shows the results of this survey.

Figure 16. Pressure map at 3° pitch, 0.1 chord.
This pressure map shows the formation of the wake, although this is less well-defined than the 0.1 chord map at 0°. The region between x/c = -0.3 and x/c = 0.0 is interesting because of the absence of a tip vortex. There are some weak pressure deficits, but none are sufficient to indicate the presence of a vortex.

The last pressure map was completed with the airfoil again pitched at 3° and with the wake-rake positioned at 1.0 chord lengths downstream. Figure 17 shows the results of this survey.

![Pressure Distribution at 1.0 Chord](image)

Figure 17. Pressure map at 3°, 1.0 chord.

Angle of Attack = 3 degree
Mach Number = 0.78
Reynolds Number = 9.81E6
The wake in this map is very well-defined; however, the tip vortex was much weaker than predicted by the CFD analysis (7). The minimum pressure ratio in the vortex core is only 0.98 and is far less than the 0.79 pressure ratio mapped at 0° pitch. This lead to questions about the accuracy of the pressure measurements taken during this survey. To answer these questions as well as gather more detailed information about the tip vortex at this pitch angle, a series of measurements was taken at points between the original data points. This provided an enhanced grid of pressure measurements in the vicinity of the vortex. Figure 18 shows the new data points superimposed on a vortex map of the original data points.

**Figure 18. Pressure map at 3°, 1.0 chord with grid points shown.**
Each small circle represents the location of a data point. The normal grid size is 0.125" by 0.125"; however, the additional data points provided a grid size of half that in the vicinity of the vortex. As a result, the vortex shape was modified slightly, but the strength was unchanged. At this time there is no satisfactory explanation for a weaker vortex at 3° than at 0°.

Five-Hole Probe Surveys

A calibrated five-hole probe was used to obtain quantitative data in the vicinity of the tip vortices. A detailed five-hole probe study was also completed near the inboard limits of the pressure map to capture the vortex sheet. Five-hole probe surveys were done only at the 1.0 downstream chord position for both angles of attack.

The survey indicates that the freestream velocity dominates the flow, especially at 0° angle of attack. For this condition, there were only three locations in the vicinity of the tip pressure deficit at which the probe measured velocity components in the x and y directions.

The results of the five-hole probe survey done at 3° angle of attack were quite different. Figure 19 shows the results of this survey. The velocity components were calculated in feet per second (fps) and scaled to 50 fps per inch.
Figure 19. Five-hole Probe Results at 3° Angle of Attack.

Again, the freestream velocity component dominates the flow; however, the circulation in the vortex core was more pronounced than in the previous case. Note that the downwash is evident in the wake. In this figure, the dark horizontal line represents the airfoil, and velocity vectors are represented by the arrows. The length of each arrow corresponds to the local velocity at that point. The velocity vector with the largest magnitude was measured at 105.76 fps.
Five-Component Sidewall Balance Results

The final step in this investigation was determination of forces and moments acting on the blade through the use of a five-component sidewall balance. In this phase of the study, the airfoil was mounted on the balance and runs were completed at pitch angles from +2° to -2° in 1° increments. One run was made at a pitch angle of +3°, but the results were so erratic that the data reduction program could not converge. Table 4 shows the results of these runs.

<table>
<thead>
<tr>
<th>AOA</th>
<th>NF</th>
<th>MX</th>
<th>CF</th>
<th>MZ</th>
<th>PM</th>
</tr>
</thead>
<tbody>
<tr>
<td>2</td>
<td>151.81</td>
<td>445.27</td>
<td>31.65</td>
<td>141.51</td>
<td>-50.8</td>
</tr>
<tr>
<td>1</td>
<td>96.37</td>
<td>260.54</td>
<td>32.44</td>
<td>146.85</td>
<td>-48.81</td>
</tr>
<tr>
<td>0</td>
<td>43.97</td>
<td>100.1</td>
<td>26.13</td>
<td>131.41</td>
<td>-31.03</td>
</tr>
<tr>
<td>-1</td>
<td>-3.88</td>
<td>-72.37</td>
<td>31.53</td>
<td>144.48</td>
<td>-48.15</td>
</tr>
<tr>
<td>-2</td>
<td>-65.73</td>
<td>-272.22</td>
<td>22.65</td>
<td>120.86</td>
<td>-46.71</td>
</tr>
</tbody>
</table>

The lift and drag coefficients, $C_L$ and $C_D$, were calculated from this data. Of particular interest was the fact that $C_D$ changed very little with angle of attack, while large differences in $C_L$ were noted. MX, or pitching moment about the x-axis was used to calculate $C_M$. MZ refers to the moment about the z-axis, while PM refers to pitching moment about the airfoil leading edge. The unit for all moments is lb-in. The x-axis passes through the 0.25 chord position on the airfoil. Figures 20, 21, and 22 show plots of $C_L$, $C_D$, and $C_M$ versus angle of attack.
Figure 20. $C_l$ versus Angle of Attack.

Figure 21. $C_D$ versus Angle of Attack.
Figure 22. $C_M$ versus Angle of Attack.

The angle of zero lift was determined by interpolation and was found to be at an angle of attack of $-0.83^\circ$. This does not agree with BHTI calculations of an angle of zero lift at $-2.0^\circ$. According to the technical drawings shown in appendix B, the angle of attack is accurate to $\pm 1^\circ$. This is a significant error for a wind tunnel model and could account for the difference in the predicted angle of zero lift versus the experimentally determined angle of zero lift.

Table 5 shows the results of BHTI CFD analysis. The wind tunnel tests of the airfoil do not agree with the BHTI preliminary data. This is possibly due to the angular error discussed in the previous paragraph.
Table 5. BHTI CFD Analysis.

<table>
<thead>
<tr>
<th>Description</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pressure Ratio</td>
<td>7.28</td>
</tr>
<tr>
<td>Dynamic Pressure</td>
<td>7073.9 lb/ft²</td>
</tr>
<tr>
<td>Area, S</td>
<td>12.95 in²</td>
</tr>
<tr>
<td>Average Chord</td>
<td>2.33 in</td>
</tr>
<tr>
<td>Normal Force</td>
<td>341.71 lbs</td>
</tr>
<tr>
<td>Chord Force</td>
<td>11.61 lbs</td>
</tr>
<tr>
<td>Moment about LE</td>
<td>-24.41 ft-lbs</td>
</tr>
<tr>
<td>Moment at Wall</td>
<td>62.50 ft-lbs</td>
</tr>
<tr>
<td>Xcp w.r.t Wing LE</td>
<td>23.58 %c</td>
</tr>
</tbody>
</table>

Finally, the lift to drag ratio was calculated for each angle of attack. The results of these calculations are shown in figure 23.

Figure 23. Lift to Drag Ratio versus Angle of Attack.
CHAPTER 6
ERROR ANALYSIS

This error analysis addresses three main error sources. These are calibration errors, data acquisition errors, and data reduction errors. Elemental errors are contained within each group, and the contribution made by individual elements were quantified whenever possible. Since most of this investigation involved collecting information about pressures in the wake of an airfoil, the error analysis focuses on the pressure measurement equipment. Finally, the calculated errors were used to determine the uncertainty in the measured data.

Calibration Errors

Transducer drift is a type of bias error, which can be a source of significant error in an experimental investigation. The calibration procedure used was originally developed at Arnold Engineering Development Center (AEDC) to minimize this type of error. It involves calibrating the pressure transducers used for data acquisition against an accurate, temperature compensated transducer at three points during the charging cycle. This was done during the first run of each day. Although all data acquisition transducers were calibrated using this procedure, the transducers used to measure static pressures were of primary concern. These transducers, like the others in the system, are Kulite ITQS-500F-500SG transducers and are not temperature compensated. The static transducers,
however, are used to calculate Mach number in the test section. Any calibration errors in these transducers could result in inaccurate Mach number calculations. The temperature compensated transducer, a Kulite XTS-1-190-200, provided this accurate standard for calibration. Data on static transducer performance from AEDC, as well as subsequent work done at UTA, show that use of this calibration procedure results in linearity errors of less than ± 0.30 percent (3).

Data Acquisition Errors

Data acquisition errors arise from transducer errors, amplifier errors, and analog to digital converter errors. The analysis of these errors follows that of Kalkhoran (3) and Rath (4). Kulite lists a combined error of 0.5 percent of full scale for its ITQS-500F-500SG transducers. This includes both linearity and hysteresis. Thus, the following equation yields the combined transducer error, $E_T$.

$$E_T = (0.005)(FS)/P$$

The full scale range of the output is denoted by FS, and the pressure is denoted by P.

The transducer contribution to the total error can be significantly reduced by use of the variable gain amplifiers incorporated in the data acquisition cart. The amplifiers in this experiment were set to a gain of 128, which optimized each transducer for the
pressure range of interest. Previous work by Kalkhoran (3) showed that use of amplifiers
can reduce the transducer error to 0.5 psi, compared to an error of 2.5 psi without the
amplifiers. Kalkhoran's research was done using much lower pressures, however, so the
transducer error for this study is correspondingly higher.

The amplifiers' error contribution is 0.1 percent of full scale, according to the
manufacturer's specifications. Thus, the amplifier total error, $E_A$, is expressed by the fol-
lowing equation.

$$E_A = (0.001)(FS)/P$$

The DSP technology data acquisition system is capable of a 100 kHz sampling
rate and simultaneous sampling of all channels. Each of its 48 channels is equipped with
it's own amplifier and analog to digital converter. The analog to digital converter error,
$E_{AD}$, is calculated using the following equation.

$$E_{AD} = (FS/ 2^{(n-1)})/P$$

The "n" in the above equation is 12 for the 12 bit converter system.

Data Reduction Errors

Data reduction was accomplished using batched FORTRAN routines written
specifically for the HIRT facility. The computer used for reduction is a 486-DX, 33
MHz, IBM-compatible PC, which is connected to the data acquisition system GPIB 488 bus. The FORTRAN programs use the following equation for calculating Mach number.

\[
M = \left\{ \frac{2}{\gamma - 1} \left[ \left( \frac{P_r}{P} \right)^{\frac{(\gamma-1)}{\gamma}} - 1 \right] \right\}^{\frac{1}{2}}
\]

Since this is the governing equation for Mach number using an isentropic assumption, no errors are introduced during the data reduction process due to an iterative scheme.

Uncertainty Analysis

The error contributions discussed in the previous section are propagated using the Root-Sum-Square (RSS) method to arrive at the total error of the system, E. Since errors due to calibration and data reduction were negligible, the only error sources used were those due to transducers, amplifiers, and analog to digital converters. This yields

\[
E = [ E_I^2 + E_A^2 + E_AD^2 ]^{0.5} = 0.7 \text{ percent} = 1.15 \text{ psi}
\]

which represents the design stage uncertainty, or \(U_d\), in the system.

Higher order uncertainty analysis attempts to quantify the effects of extraneous variables on the measured data. One extraneous variable of particular interest was the effect of time lag on the accuracy of the pressure measurements. That time lag was a problem was noted early in this project, and the modifications used to reduce it's effects
are detailed in chapter 2. The modifications to the pressure measurement system focused on improving the response time of the plenum cavity pressure systems, since these pressures are critical to an accurate determination of Mach number. The modifications resulted in a marked improvement, but did not address the problem of time lag in the wake rake or five-hole probe. Note that a time lag is still present in the plenum pressure system. A pitot-static probe with a differential pressure transducer mounted in the test section would further improve the accuracy of the pressure measurement system.

Time lag in the wake rake resulted in some free stream total pressure measurements that exceeded the tunnel total pressure. This was also reported by Rath (4).

The first order uncertainty, or U₁, refers to uncertainty due to time lag errors, while the second order uncertainty, or U₂, refers to run to run repeatability errors. The time lag errors were quantified by calculating the mean and standard deviation of the measurements from the rake positioned in the free stream. These were then used to calculate higher order uncertainties. If time lag were not a factor, the pressures measured by the rake would equal the total pressure measured by the master transducer. In fact, the pressures measured by the wake rake were consistently higher than those measured by the tunnel pressure measurement system. This yielded a wake rake bias error of 0.1854 psi. Pooled statistical analysis incorporated free stream wake rake measurements from several runs and resulted in a pooled sample standard deviation, \( S_x \), of 0.7183 psi and a pooled standard deviation of the means, \( S_{xbar} \), of 0.0704 psi. Using these results,
plus the unpoled statistics, the first and second order uncertainties are calculated as follows:

\[ U_1 = \pm t_{v,95} S_x = 0.85 \text{ percent} = 1.41 \text{ psi} \]

\[ U_2 = \pm t_{v,95} <S_{	ext{bar}} > = 0.08 \text{ percent} = 0.138 \text{ psi} \]

The t estimator was obtained from a table of the Student-t distribution, which is used to when calculating precision intervals for finite data sets. The Student-t distribution is available in any statistics book. The subscript \( v \) refers to the degrees of freedom, and the subscript 95 refers to a 95 percent confidence interval. For simplicity, the t estimator was used throughout this analysis, even when the sample size was large enough to justify the use of infinite statistics. This did not compromise the accuracy of the uncertainty analysis since the t value approaches the z value used in infinite statistics when the sample size gets large (9).

The Nth-order uncertainty allows direct comparison between results of similar tests. It is calculated as follows:

\[ U_N = \left[ U_c^2 + U_t^2 + U_2^2 \right]^{0.5} = 1.1 \text{ percent} = 1.82 \text{ psi} \]

where \( U_c \) is the instrument uncertainty. Note that for this system the resolution error, \( U_0 \), was taken to be zero. Thus, \( U_c = U_d \) in this case. As expected, the time lag errors in the
pressure measurement system contributed the most to the total error. The low value of $U_2$ indicates that this aspect of the experiment was well-controlled.

Uncertainty analysis of the five-hole probe results was performed in the same manner and yielded the following results. Note that the design stage uncertainty is the same for both the wake rake and the five-hole probe.

\[
(U_d)_{probe} = 0.70 \text{ percent} = 1.15 \text{ psi}
\]
\[
(U_i)_{probe} = 0.30 \text{ percent} = 0.53 \text{ psi}
\]
\[
(U_2)_{probe} = 0.07 \text{ percent} = 0.11 \text{ psi}
\]
\[
(U_N)_{probe} = 0.76 \text{ percent} = 1.27 \text{ psi}
\]

The results of Nth-order uncertainty analysis indicate that the five-hole probe total pressure measurement is more accurate than that measured by the wake-rake. This reduction in uncertainty was the result of a significant reduction in first order uncertainty. The first order uncertainty is affected principally by the diameter of the total pressure orifice and the length of the tubing in the pressure measurement system. NAVORD Report 1488 (8) lists matching orifice diameter to tubing inside diameter and minimizing tubing length as two criteria for minimum response time. During the design of the probe, the orifice diameter was matched to the inside diameter of the connecting tubing to eliminate this as a source of error. The connecting tubing was also shortened as much as
possible. As a result, time lag proved to be a secondary contributor to uncertainty in the five-hole probe tests.

The uncertainty in the five-hole probe's static port measurements cannot be determined at this time. The design stage uncertainty is the same as for the total pressure measurements; however, there is insufficient data to calculate the higher order uncertainties for the static ports. It is estimated that the static pressure uncertainty for the probe would be approximately the same as that of the probe total pressure uncertainty.

Other factors affecting the probe's ability to determine flow angularity were errors in pitch and roll positioning during the calibration procedure described in appendix A. Pitch angles were set using fixed mounting points on a bracket specifically designed for probe calibration. Thus, errors in pitch angle were assumed to be negligible. Roll errors posed a much more serious problem. The method used to set the roll angle during the calibration involved using a radial scale attached to the back of the probe. This method was crude at best, and no means of determining the roll angle error currently exists. It is estimated to be on the order of ± 2.5°, in spite of the fact that great care was used in positioning the probe to ensure accurate roll angle placement. This error is not sufficient to significantly affect the calibration at the pitch angles used in this study; however, if the probe is to be used at higher pitch angles, a better method of setting the roll angle during calibration must be developed.
The documentation provided with the five-component sidewall balance contains instructions for performing a detailed calibration prior to its use. This procedure was carefully followed, and the uncertainty in the measured values is assumed to conform to the manufacturers estimate of ±0.03 percent.

The airfoil was rigidly mounted to the sidewall balance using an adapter plate. This allowed the airfoil to be pitched about the 0.25 chord point. A precision digital level was used to do this, and the uncertainty in pitch angle is estimated to be within ±0.05°.

Finally, the uncertainty in Mach number calculation was an item of interest. The Mach number uncertainty is a function of the uncertainty in both the total pressure measurement and the plenum pressure measurement. The design stage uncertainty is the same as that previously calculated. Calculations of higher order uncertainty for the plenum and total pressure measurement systems yielded the following results:

\[
(U_1)_{\text{plenum}} = \pm 0.90 \text{ percent} = \pm 1.02 \text{ psi}
\]
\[
(U_2)_{\text{plenum}} = \pm 0.04 \text{ percent} = \pm 0.047 \text{ psi}
\]
\[
U_{\text{plenum}} = \pm 1.14 \text{ percent} = \pm 1.29 \text{ psi}
\]
\[
(U_1)_{\text{Total}} = \pm 0.76 \text{ percent} = \pm 1.27 \text{ psi}
\]
\[
(U_2)_{\text{Total}} = \pm 0.035 \text{ percent} = \pm 0.059 \text{ psi}
\]
\[
U_{\text{Total}} = \pm 1.03 \text{ percent} = \pm 1.73 \text{ psi}
\]
The uncertainty in Mach number was calculated to be ± 1.5 percent using the equation below.

\[
\frac{U_{\text{Mach}}}{M} = \left[ \left( \frac{U_{\text{plenum}}}{P_{\text{plenum}}} \right)^2 + \left( \frac{U_{\text{total}}}{P_{\text{total}}} \right)^2 \right]^{\frac{1}{2}}
\]

For a Mach number of 0.80, this yields an uncertainty of ± 0.01. In a similar manner, uncertainty in Reynolds number was calculated to be ± 1.15 percent, or ± 112,815.

Uncertainty in velocity components determined using the five-hole probe could not be calculated. This requires knowledge of the probe static pressure measurement uncertainty, which has been previously discussed. The uncertainty in velocity components measured by the probe is assumed at this time to be approximately the same as the Mach number uncertainty, or ± 1.5 percent.
CHAPTER 7

CONCLUSIONS AND RECOMMENDATIONS

Conclusions

One of the major stumbling blocks in developing blade tips for reduced BVI has been the difficulty in correctly modeling the flow conditions at the tip of a helicopter rotor blade. Such conditions typically involve transonic Mach numbers and very high Reynolds numbers. The UTA High Reynolds Number Transonic facility proved fully capable of accurately modeling such conditions.

The 13 port wake-rake was used in this study to obtain the initial qualitative data about the pressure distribution in the wake of the airfoil model. The wake-rake had some accuracy problems, most notably a time lag error, which contributed significantly to the overall uncertainty in the wake-rake measurements. Even with this lag in the system, the measurements were sufficiently accurate to allow construction of detailed vortex maps.

The vortex maps were crucial to the next phase of the study, which entailed using a calibrated five-hole cone probe to gather information about flow angularity in the tip vortices and in the vicinity of the vortex sheets. The results confirmed the wake-rake
predictions of the vortex core locations. The resulting plots provided a visual representation of the velocity vectors within the tip vortices and the vortex sheets.

The experimental results obtained at 3° angle of attack do not agree with BHTI predictions. The tip vortex measured in the wind tunnel tests was significantly weaker than expected. No satisfactory explanation for this anomaly exists at this time.

The time lag errors noted in the wake-rake were significantly reduced in the five-hole probe total pressure measurements due primarily to reduction in connecting tube length and matching the size of the pressure orifice to the inside diameter of the connecting tubing. The uncertainty in the static pressure measurements was not calculated due to the small number of available data points.

The final phase of this study involved determination of aerodynamic forces acting on the blade tip using a five-component sidewall balance. A precision digital level was used to set the pitch angle, and measurements were taken at pitch angles from +2° to -2° in 1° increments. The resulting data were used to determine $C_L$, $C_D$, and $C_M$, which were then plotted as a function of pitch angle. The data obtained in the wind tunnel tests did not agree with BHTI predictions. One possible explanation currently being investigated is presence of an angular error in the airfoil model. This error, according to the manufacturer's drawings in appendix B, is ±1°.
Uncertainties in pressure measurements were calculated for the total pressure system, the plenum pressure system, the wake-rake, and the five-hole probe. Mach number uncertainty was also calculated, and in all cases, uncertainties were determined to be in the acceptable range. Repeatability was a major concern during this study, since consistent wind tunnel performance is critical to obtaining reliable data. The uncertainty analysis clearly showed that the test conditions were highly repeatable for all phases of the investigation.

Recommendations

Although the uncertainties in the experimentally obtained data were acceptable, the following recommendations would further improve the accuracy of the measurements:

1. Use of smaller transducers would reduce the response time and improve the design stage uncertainty. This would improve the accuracy of every pressure measurement system used on the HIRT.

2. Response time of the 13 port wake-rake could be significantly reduced by shortening the connecting tubing. Also, use of smaller transducers and mounting the transducers inside the wind tunnel would further reduce the response time and improve accuracy.
3. Use of a differential pressure pressure transducer in conjunction with a pitot-static probe mounted in the test section would provide the best possible pressure measurement. This would reduce the Mach number uncertainty, possibly to negligible levels.

4. The traversing mechanism used with the wake-rake and the five-hole probe causes some interference effects within the tunnel. This often resulted in a Mach number lower than that predicted. More research needs to be done on this phenomenon so that a correction chart can be constructed.

5. A better means of setting the roll angle on the five-hole probe during calibration needs to be developed. The current method does not lend itself to accurate determination of roll angle error.

6. The modifications to the wind tunnel described in appendix A improved the wind tunnel response time, especially the plenum pressure response time. Further improvement could be realized if the plenum pressure transducers were mounted inside the plenum cavity instead of outside the cavity.

This investigation provided crucial data to BHTI for the development of an experimental blade tip for reduced BVI on the V-22 Osprey. It also produced sufficient data to allow determination of Nth-order uncertainties in the HIRT pressure measurement systems, thereby providing a database for comparison with future test results.
APPENDIX A

FIVE-HOLE PROBE CALIBRATION PROCEDURE
FIVE-HOLE PROBE CALIBRATION PROCEDURE

Multiport pressure probes are highly sensitive pressure measurement tools used to gather detailed information about a flow field. A five-hole pressure port was designed and manufactured for this study. Figure 24 shows the details of the probe. The probe was calibrated at the Mach and Reynolds number specified in the experimental model. The calibration required a total of 25 runs in the HIRT facility.

![Diagram of probe dimensions](image)

Figure 24. Three view diagram of probe, dimensions in inches.

The new probe was fabricated as a replacement for the university's 3/8 inch diameter probe, which was damaged during a wind tunnel test in 1994. A smaller diameter probe was desired as a replacement to increase the accuracy of the flow measurements. The new probe was made of 5/16 inch outside diameter (O.D.) stainless steel tubing with
a brass tip mounted to the end of it. The brass tip was milled into the shape of a 30° cone (± 5°), just as its predecessor had been. The cone angle selection is important because it influences the sensitivity of the probe. Large cone angles provide for greater separation between static pressure ports, and thus increase sensitivity. Small cone angles minimize probe interference effects on the flow field (12). The 30° angle is large enough to provide good pressure response while minimizing interference effects in the flow field.

Five pressure ports, each with a diameter of 0.052 inches, were milled into the brass cone tip. This diameter was selected because it matched the inside diameter of the tubing used to connect the pressure transducers to the probe. Matching the diameters of the orifices and connecting tubing in this way minimizes time lag in the measurement system.

The pressure ports were numbered from one to five, with Port 1 being the total pressure port and centered on the cone tip. Four equally spaced static pressure ports of the same diameter were arrayed around the tip. The pressure ports on the back side of the tip were soldered to 0.052 inch inside diameter (I.D.) steel tubing prior to the stainless steel tube’s attachment to the brass tip. The free ends of these tubes were then soldered to brass fittings which allowed easy connection to flexible tubing attached to pressure transducers. Finally, the probe was reinforced with a 3/8 inch O.D. stainless steel shroud attached to the aft 10 inches of the probe. The forward 5 inches of the probe
were left unaltered by this procedure, while the tube wall in the aft section of the probe was doubled.

Due to the need to complete the calibration of the new probe quickly, an abbreviated calibration procedure was developed. A full calibration would require tunnel runs at pitch angles from -12° to +12° in 2° increments with roll angles from -90° to +90° in 10° increments for a total of 143 runs. A tunnel break is necessary each time the pitch or roll angle is changed, which would have required far more time than was available. The abbreviated calibration procedure used required a total of 25 runs and involved calibration runs at -0°, -4°, and -8° pitch, and roll angles from -90° to +90° in 30° increments. This procedure proved adequate to obtain accurate data for generating the required calibration chart.

To properly calibrate the probe, it was necessary to mount the probe in the tunnel so that the tip was pitched about the a single point. A collar was attached to the probe to facilitate accurate placement of the probe tip. The probe was attached to a mount using a stainless steel bracket and secured to the mount using cap screws. The mount was designed with screw holes drilled in precise 2° intervals. Thus, the error in the pitch angle was negligible. The collar used to position the probe had a scale on it which was used to set the roll angle. This was done manually, making the roll angle error difficult to precisely determine; however, it was estimated to be ±2.5°.
Pressure data were collected from each calibration run and were used to determine calibrated downwash and sideslip coefficients, or $C_\alpha$ and $C_\beta$, respectively. These pressure coefficients were formed as follows:

$$C_\alpha = \frac{P_2 - P_4}{q}$$

$$C_\beta = \frac{P_3 - P_5}{q}$$

Figure 25 shows the plot of $C_\alpha$ vs. roll angle for $0^\circ$, $4^\circ$, and $8^\circ$. Figure 26 shows the plot of $C_\beta$ vs. roll angle for $0^\circ$, $4^\circ$, and $8^\circ$.

Figure 25. Plot of $C_\alpha$ vs. Roll Angle.
Figure 26. Plot of $C_\beta$ vs. Roll Angle.

A curve fit of the data was necessary to generate the calibration charts. A linear regression was attempted, but failed to yield satisfactory results. Closer examination of the data revealed its highly non-linear nature. Because of this, a non-linear curve fit using the Marquardt-Levenberg algorithm contained in Jandel’s SigmaPlot software package was selected. The Marquardt-Levenberg algorithm yielded acceptable results, and the fitted data were plotted in a calibration chart using SigmaPlot. The calibration chart correlates the dimensionless pressure coefficients to downwash and sideslip angles in the flow. Figure 27 shows the completed calibration chart generated as a result of this procedure.
The calibrated probe was then mounted in the wake of the experimental airfoil model and a series of wind tunnel tests were performed to define the flow in selected locations. The dimensionless pressure coefficients were formed from the pressure data collected and used to enter the calibration chart. The calibrated downwash angle was obtained from the radial scale, while the roll angle was obtained from the angular scale. These angles were then used to define velocity components in the x, y, and z-directions using the following equations:
\[ u = V \sin \theta \sin \phi \]
\[ v = V \sin \theta \cos \phi \]
\[ w = V \cos \theta \]

where \( \theta \) refers to the pitch angle and \( \phi \) is the roll angle. The velocity is denoted by \( V \).
APPENDIX B

AIRFOIL MODEL TECHNICAL DRAWINGS
Figure 28. Model Assembly.
Figure 29. End Plate.
Figure 31  Machining Blank.
REFERENCES


