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An experimental investigation of the perpendicular vortex-airfoil interaction at transonic speeds

Kalkhoran, Iraj Masbooghi, Ph.D.
The University of Texas at Arlington, 1987

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AN EXPERIMENTAL INVESTIGATION OF THE PERPENDICULAR VORTEX-AIRFOIL INTERACTION AT TRANSONIC SPEEDS

by

Iraj M. Kalkhoran

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

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THE UNIVERSITY OF TEXAS AT ARLINGTON

December 1987
AN EXPERIMENTAL INVESTIGATION OF THE PERPENDICULAR
VORTEX-AIRFOIL INTERACTION AT TRANSONIC SPEEDS

APPROVED:

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DEDICATION

This work is dedicated to my parents who made my education possible.
ACKNOWLEDGMENTS

I am indebted to Dr. D. R. Wilson, the chairman of graduate supervisory committee, for his helpful suggestions and guidance during this research program.

I acknowledge many useful suggestions given by Dr. D. D. Seath, which provided a deeper understanding of the vortex problem. I wish to thank Drs. D. A. Anderson, J. E. Fairchild, and I. C. Dragan, for many useful suggestions.

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I am deeply indebted to my wife and children for their patience and understanding.

November 11, 1987
ABSTRACT

AN EXPERIMENTAL INVESTIGATION OF THE PERPENDICULAR
VORTEX-AIRFOIL INTERACTION AT TRANSONIC SPEEDS

Publication No.——

Iraj M. Kalkhoran, Ph.D.
The University of Texas at Arlington, 1987

Supervising Professor: Donald R. Wilson

Extensive experimental studies were conducted in the UTA High Reynolds Number Transonic Wind Tunnel facility. The experiments included a complete calibration of the wind tunnel over a wide range of transonic Mach numbers and at Reynolds numbers representing the normal operating condition of interest to aerospace vehicles. The calibration tests incorporated the effects of the secondary flow on the test section flow quality and its uniformity.

Experimental investigations of the compressible trailing vortex system were performed by means of a thorough probing of the viscous core of the vortex system. Detailed flow field
measurements and contour plots of the total pressure field in the neighborhood of the vortex core are presented.

Comprehensive experimental studies were conducted to investigate the pressure distribution on a C-141 and a NACA 0012 airfoil sections for a range of transonic flow conditions. Transonic vortex airfoil interaction tests at Mach numbers ranging from 0.68-0.90 and airfoil chord Reynolds numbers of 3.5-6 million were conducted to simulate the helicopter rotor blade vortex interaction (BVI) problem. The interaction scheme involves positioning a lifting wing (vortex generator) upstream of an airfoil so that the trailing vortex interacts with the downstream airfoil. The interacting airfoils included a symmetric NACA 0012 and a two-dimensional C-141 (NACA 65 series) airfoil section. The effects of vortex strength, vortex height above the downstream airfoil, Mach number and Reynolds number on the pressure distribution of the downstream airfoil, and the normal force coefficients were studied and the quantitative results are presented.

The results obtained from these experiments indicate a substantial change in the pressure distribution of the downstream airfoil, a spanwise drift of the vortex core after interacting with the airfoil, and a high degree of unsteadiness in the vicinity of the vortex viscous core.

It is felt that the particular experimental configuration used avoids the complexity involved with a
rotating blade, yet it preserves the important physical characteristics of the BVI problem. Due to the geometric simplicity of the present investigations, the data of present analysis are believed to be of significant importance in validating the computational codes.
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<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
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<tbody>
<tr>
<td>C</td>
<td>Chord length</td>
</tr>
<tr>
<td>$C_p$</td>
<td>Pressure coefficient</td>
</tr>
<tr>
<td>$\overline{C_p}$</td>
<td>Average integrated pressure coefficient</td>
</tr>
<tr>
<td>$C_{p*}$</td>
<td>Critical pressure coefficient</td>
</tr>
<tr>
<td>$C_L$</td>
<td>Lift coefficient</td>
</tr>
<tr>
<td>d</td>
<td>Vortex core diameter</td>
</tr>
<tr>
<td>E</td>
<td>Error</td>
</tr>
<tr>
<td>h</td>
<td>Vortex height</td>
</tr>
<tr>
<td>M</td>
<td>Mach number</td>
</tr>
<tr>
<td>P</td>
<td>Pressure</td>
</tr>
<tr>
<td>$P_t$</td>
<td>Total pressure</td>
</tr>
<tr>
<td>Q</td>
<td>Velocity</td>
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<tr>
<td>Re</td>
<td>Reynolds number</td>
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<tr>
<td>S</td>
<td>Spanwise drift</td>
</tr>
<tr>
<td>T</td>
<td>Temperature</td>
</tr>
<tr>
<td>t</td>
<td>Time</td>
</tr>
<tr>
<td>$V_i$</td>
<td>Induced velocity</td>
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<tr>
<td>X,Y,Z</td>
<td>Cartesian coordinates</td>
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<tr>
<td>W</td>
<td>Uncertainty</td>
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<tr>
<td>W</td>
<td>Mass flow rate</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>Angle of attack</td>
</tr>
<tr>
<td>$\delta_f$</td>
<td>Ejector flap opening</td>
</tr>
<tr>
<td>$\gamma$</td>
<td>Gas constant</td>
</tr>
<tr>
<td>$\tau_W$</td>
<td>Wall porosity</td>
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</table>
\[ \mu \quad \text{Coefficient of viscosity} \]
\[ \Gamma \quad \text{Vortex circulation} \]
\[ \Omega \quad \text{Angular velocity} \]
\[ \omega \quad \text{Vorticity} \]

Subscripts

\[ A \quad \text{Amplifier} \]
\[ AD \quad \text{A-D Converter} \]
\[ pc \quad \text{Plenum chamber} \]
\[ t \quad \text{Total} \]
\[ ts \quad \text{Test section} \]
\[ V \quad \text{Vortex} \]
\[ VG \quad \text{Vortex generator} \]
\[ \infty \quad \text{Free stream} \]

Superscripts

\[ (-) \quad \text{Time averaged} \]
\[ . \quad \text{Time rate} \]

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CHAPTER I

INTRODUCTION

The study of Vortex-Airfoil Interaction, or more commonly Blade-Vortex Interaction (BVI), in its present form is a relatively new subject. These studies find their importance in several aspects of science and engineering, and have always been a challenging problem for the theoreticians. Although the physics of the interaction problem is poorly understood, the interaction of vortices with lifting surfaces can have a significant influence on the aerodynamics, aeroelastic behavior and aeroacoustics of maneuvering vehicles, and more critically so in transonic flows. This is due to sensitivity of the shock strength and position to small disturbances in transonic flows. With respect to the rotorcraft industry, this problem has been of particular interest since the interaction problem occurs continuously in a helicopter rotor flow field. It is well established that the interaction of a trailing vortex wake in such a flow field with the oncoming rotor blades can induce unsteady blade loading and aerodynamic noise. The concentrated trailing vortices generated by the blade tips undergo a continuous formation and deformation in space, leading to a variety of possible Blade-Vortex
Interactions, which depend on the exact geometry of the interaction. As shown in Fig. 1.1, the Blade-Vortex Interaction can be viewed as an unsteady, three dimensional interaction of a curved-line vortex with a lifting surface at an arbitrary interaction angle. The limiting cases for such encounters are parallel (vortex axis parallel to the blade span) and perpendicular (vortex axis perpendicular to the blade span) interactions. Principally the former interaction is essentially two-dimensional but unsteady, while the later encounter can be considered as a steady but highly three-dimensional problem (Fig. 1.2).

Under certain flight conditions, helicopter rotors can produce impulsive noise at a regular frequency corresponding to the blade passage frequency. The two important mechanisms that are known to be responsible for the BVI noise, which is known as "Blade-Slap" or "Blade-Bang", are:

1. The rapid load variations caused by a rotor blade passing close to or through a tip vortex trailing from the same or another blade, which is known to be responsible for impulsive sounds, and

2. Shock formation and fluctuations due to existence of local transonic flows caused by interaction of tip vortices with blades.

The discussion of Blade-Vortex Interactions and their aeroacoustic implications are covered in excellent papers by
George (Ref. 1), George and Chang (Ref. 2) and very recently by Leverton (3).

The noise produced by the interaction of vortices with lifting surfaces is not limited to the main rotor. In fact it has been shown experimentally that the tail rotor BVI mechanism can be subjectively annoying, and in certain flight conditions can dominate over the main rotor BVI and that of the engine noise (Ref. 4). The tail rotor blade-vortex interaction mechanism is considered to be potentially important under forward flight or vertical descent conditions. In exact analogy to the main rotor BVI, the vortices shed by the main rotor pass through the tail rotor which results in noise generation. In principal, the strength of the noise generated by this mechanism strongly depends on the position of the tail rotor relative to the main rotor and it is possible to reduce the noise level by changing the design of the vehicle. For example, altering the relative position of the tail rotor and main rotor is a possible solution. In addition it has been suggested that particular main rotor blade tip shapes may be used to reduce the vortex strength and diffuse the vorticity field in a manner which reduces interaction noise. These studies are an active research area and extensive experimental and theoretical investigations are still being conducted.

Although the aerodynamic parameters are the essential
input data for determining the acoustic characteristics, the role of aerodynamics has been somewhat overlooked in BVI investigations. The interaction between the air flow associated with several components of the vehicle has been known to be a major source of noise excitations. (Ref. 5-6)

It is also known that the strong BVI at close encounters is responsible for extensive boundary layer separation at the blade surface. This vortex induced separation causes a significant alteration of the pressure distribution and is in fact a form of dynamic stall which can produce large unsteady loading.

The Vortex-Airfoil Interaction problem is not limited to rotorcraft applications, and it has traditionally been studied in aircraft applications. For example, the response of small aircraft encountering the trailing vortex system of large aircraft has been extensively studied in the past (Refs. 7-8). In addition the study of the BVI problem is important to propellers and turbo-machinery.

The problem of vortex dynamics has been the subject of many studies in last few decades. Parameters such as vortex-core size, vortex strength and vortex position are of significant importance in solution of acoustic problems and need to be accurately modeled in order to obtain useful acoustic data. Although several vortex structure models have been suggested and extensively used with various complexities,
our knowledge of the physics of this phenomenon is not complete.

Background

Theoretical and experimental study of Blade-Vortex Interaction (BVI) has been subject to intensive researches over past 25 years, and the interaction problem continues to be studied in order to gain a better understanding of this problem. The future applications of helicopters may be limited by the noise generated by the rotors, both from environmental concern and also military detectability and maneuverability considerations. The lack of arrival at a solution to this problem has been argued by some to be due to the limited availability of reliable aerodynamic input data, while there are researchers who question the accuracy of the acoustic theories for predicting the interaction problem. Complexity of the experimental setup has also been highly debated to be a major drawback since it produces several complicated interactions within the BVI.

Historically, some basic studies of this subject were made as early as 1935 (Ref. 9) and 1936 (Ref. 10), where the sound generation due to steady lift and drag forces was analyzed by Guti. In these studies the mechanism for rotor noise was considered to be the same as that of the propeller noise. These approaches severely underestimated the intensity of
noise generation.

The current status of the BVI investigation is in a relatively advanced stage. Most of the experimental and computational schemes dealing with the BVI problem have focused on the parallel (2D, unsteady) problem, and very few investigations have considered the perpendicular interaction problem (3D, steady). Because the perpendicular interaction problem is thought to be steady in nature, the noise problem is believed to be less severe. However, the effect of the vortex on the aerodynamic field surrounding the blade is known to be quiet pronounced (Ref. 11).

The first notable theoretical investigation, perhaps, is due to Widnall (Ref. 12), who studied the sound generated by the interaction of a two dimensional airfoil with an obliquely oriented vortex. Widnall and Wolf (Ref. 13) considered a three dimensional interaction which included the two dimensional problem as a special case. In their investigation the vortex was assumed to remain stationary and the vortex induced velocity field was decomposed into fourier components. These components were then used as inputs for the airfoil sinusoidal gust response such that the fluctuating pressure field could be determined which resulted in calculations of sound. Similar treatment of the BVI problem has been used extensively in the past by several investigators (Refs. 14-15).

Another approach to study the BVI problem, which is
widely used in the literature, is composed of interaction of airfoils with a rectilinear vortex filament. The most notable of these are the study made by Hardin and Masson (Ref. 16), and more recently by Poling and Dadone (Ref. 17) who used a conformal transformation and discrete vortex dynamics to calculate the interaction of a blade with vortices drifting with the free stream.

The computational study of two dimensional transonic BVI was first considered by George and Chang (Refs. 2,18), who used the transonic small disturbance theory, including regions of convected vorticity. Computational schemes with various complexities have been developed incorporating both inviscid and full Navier Stokes equations (Refs. 19-20). Steinhoff et. al. (Ref. 21) incorporated line vortices into the three dimensional compressible potential flow equation, and used a modified Biot-Savart law to compute the vortical velocity field. The most recent 2D computational schemes utilizing full Navier Stokes equations (Ref. 22) and Euler algorithms (Ref. 23), were reported to have good agreement with the experimental data. The numerical solution of the BVI problem using Euler equations is thought to have a lack of true physical basis in inviscid creation of the tip vortex by such schemes, despite the fact that the Euler equations have the ability to capture and convect vorticity. On the other hand the solution of vortical flow problems using full Navier Stokes codes are debated to be too costly to be used in
routine design analysis (Ref. 24).

On the experimental side, a number of investigations have focused on the BVI problem. The excellent experimental data of Caradonna et. al. (Ref. 25), is being extensively used for validating new theoretical and numerical schemes. In their work they considered a rotor downstream of a vortex generator such that a two dimensional interaction of blade and vortex occurs (parallel interaction), and effects of the free stream Mach number and vortex proximity were studied. The flow visualization studies of Ref. 26 and laser velocimeter (LV) measurements of Ref. 11 are also of significant importance in understanding the BVI phenomenon. Schlinker and Amiet (Ref. 14) have also considered the three dimensional interaction problem, in which hot wire measurements in the vortex core are used to predict the acoustic signature. They conclude that ingestion of the vortex by the rotor generates harmonic noise and impulsive waveforms. Among the other relevant investigations to the present report are those of Ham (Ref. 27), who conducted perpendicular interaction studies similar to those of the present investigation. His measurements indicated $\Delta C_L$ values of 0.2 to 0.3, and discovered the possibility of flow separation induced by intensive vortex loading. The simulation of the perpendicular blade vortex interaction of Ref. 28 indicated a 40 percent increase in drag and a loss of lift for a NACA 0012 airfoil at $M=0.6$. Recent low speed experimental investigations of Ref. 29, also
indicated a significant change in the pressure distribution of the downstream airfoil as well as a spanwise drift of the vortex when interacting with the airfoil in such encounters. All of the experimental studies discussed above have been conducted in low speed wind tunnels, except those of Ref. 25 in which parallel interaction data for $M=0.714$ is reported.

Objectives of the Current Research Program

Apparently very little research has been conducted on the vortex airfoil interaction at transonic speeds, even though this is the common flight environment for the advancing rotor blade at forward flight conditions. Moreover, simulation of the interaction at appropriate Reynolds numbers does not appear to have been attempted in previous experiments. Both of these deficiencies justify the present experimental investigation of Vortex-Airfoil Interaction phenomenon.

Although the current experimental study emphasizes the perpendicular vortex airfoil interaction at transonic speeds and at Reynolds numbers representative of modern rotor operation, some relevant studies common to both transonic aerodynamics and wind tunnel testing are also covered in this report.

The previous wind tunnel calibrations conducted in the PILOT HIRT wind tunnel facility (Ref. 30), the effect of secondary flow through the ejector flap system on the test
section Mach number distribution and its uniformity was not considered. In the present research program the test section calibration experiments were performed for four ejector flap opening and the test section Mach number uniformity for the aforementioned ejector flap settings was studied. The experimental investigation of the present study also included development of a scheme for calculating the test section Mach number from that of the plenum chamber. The scheme consisted of obtaining a relationship between the test section and plenum chamber Mach numbers such that the test section Mach number could be determined by applying a correction factor to the plenum chamber Mach number. Such a procedure will eliminate the need for a separate probe for obtaining the test section Mach number while an experiment is being conducted in the test section, thus eliminating an excessive tunnel blockage.

It is believed that the experimental data of Ref. 30 concerning the airfoil pressure distribution in the PILOT HIRT wind tunnel facility is not sufficient to prove the feasibility of conducting such experiments in the small scale Ludwig tube wind tunnel facility. The experimental data of Ref. 30 indicated good agreement with the results of AEDC transonic wind tunnel facilities for a C-141 airfoil at an airfoil angle of attack of 0, however their results for an airfoil angle of attack of 2 degrees were very poor. In the present study the airfoil pressure distribution was
investigated using both a C-141 airfoil section and a NACA 0012 airfoil section for which extensive experimental data from different transonic wind tunnel facilities are available.

The UTA high Reynolds number wind tunnel facility is discussed in Chapter II. The calibration of the wind tunnel and test section uniformity studies utilizing effects of secondary flow on the test section flow quality are covered in Chapter III. Chapter IV contains the results of vortex structure studies based on total pressure rake measurements. The calibration results of the airfoil sections used for the interaction studies are presented in Chapter V, which are then used as the experimental data base for the interaction tests of Chapter VI.
Fig. 1.1 Illustration of the Blade-Vortex Interaction
(a) VORTEX PARALLEL TO LEADING EDGE

(b) VORTEX PARALLEL TO FREE STREAM

Fig. 1.2 Parallel and perpendicular Blade-Vortex Interactions
CHAPTER II

FACILITY DESCRIPTION

History

The UTA Transonic Wind Tunnel facility (Fig. 2.1) is a Ludwig tube wind tunnel which can provide high Reynolds number transonic flows. The facility is essentially a blowdown type wind tunnel with the air storage vessel or vessels replaced by a long supply tube (charge tube). The facility is a one-thirteenth scale model of a high Reynolds number transonic tunnel which was to be constructed at the Arnold Engineering Development Center (AEDC) as the National Transonic Facility. The University of Texas at Arlington acquired the facility after testing was discontinued at AEDC in 1976. After a complete renovation of the facility control and data acquisition system, the facility became operational in 1983.

UTA Transonic Wind Tunnel Facility

The Ludwig tube storage system (charge tube) has a circular cross section, 13.94 in. in diameter and 111 ft in length (Fig. 2.1). The tunnel can be charged to a pressure of
660 Psia (hydrostatically tested to 1150 Psig), which can produce a maximum stagnation pressure of about 500 Psia. Downstream of the charge tube is a convergent nozzle (Fig. 2.2, 2.3), 18.5 in. long, with contour design to provide a smooth transition of the flow from the circular charge tube to the rectangular test section. The contraction ratio (area ratio) of the nozzle is 2.27. The nozzle ideally chokes when the nozzle entrance Mach number reaches 0.269, but experimental data of Ref. 30 indicate choked flow for a nozzle entrance Mach number of approximately 0.30, which is due to the reduction of the effective area caused by the viscous boundary layer.

The test section (Fig. 2.2, 2.3, 2.4), has a rectangular cross section measuring 7.34 in. by 9.16 in. and is 25.4 in. long. The test section walls are based on a conventional porous wall design, with porosity available on all four walls. The porous walls consist of two stacked plates with 60-degree inclined holes having a tapered porosity pattern in the upstream one third of the test section. The wall porosity (ratio of the open area to the test section wall area) can be manually varied in the range from 3.5 to 10 percent by moving one plate relative to the other. The porosity adjustment capability is provided on all four walls. The variable porosity, coupled with the use of slanted holes, is used for canceling both the shock and expansion wave reflections from the tunnel walls which are common in transonic flows. The use
of such walls are also of practical importance in obtaining Mach numbers in the low supersonic range with a fixed area ratio convergent nozzle. Wall interference effects, which are critical at transonic Mach numbers, can be reduced by using porous wall test sections. Finally, by the proper use of variable porosity walls, transonic flows for a range of Mach numbers can be achieved.

The test section is surrounded by a plenum chamber with a volume of approximately 1.75 times that of the test section (neglecting the volume of the wall support structure). Flow in the plenum chamber is exhausted to the atmosphere through a system consisting of ten 2 in. I.D. flexible hoses that are attached around the downstream end of the plenum shell and run to a common manifold. Downstream of the manifold is a diaphragm holder, consisting of two plates held together by a clamp assembly. Mylar diaphragm material of the required thickness (depending on the wind tunnel operating pressure) is placed here. The diaphragm is ruptured at the desired time during a run by means of a plenum exhaust cutter (PEC), which is actuated by a pneumatic cylinder to initiate the secondary flow through the plenum chamber. The 6 in. diameter plenum exhaust line splits into two branches downstream of the diaphragm assembly (Fig. 2.5). One branch vents to the atmosphere through a 6 in. variable orifice ball valve (Fig. 2.6) for which the open area is set as required by the run Mach number prior to tunnel firing. The second branch contains
a Grove Flex-Flo Valve (FFV), which is normally open during the tunnel start, and is closed during the steady state portion of the run by inflating a rubber boot inside the valve body. The Flex-Flo Valve is used to reduce the starting time by relieving the chamber pressure during the starting process.

The main starting device in the UTA transonic wind tunnel is the Sliding-Sleeve Valve (SSV) which is located at the extreme downstream section of the facility inside the exhaust sphere (Fig. 2.1, 2.2). The Sliding-Sleeve Valve is driven by a pneumatic cylinder and is opened at the beginning of the run to initiate the primary flow through the test section. The proper opening of the main starting device is of critical significance in the overall performance of small scale wind tunnels, due to the relatively short duration of the steady state time for such wind tunnels.

All of the main tunnel flow, as well as that of the secondary system, consisting of both branches of the plenum chamber, empty into a large sphere, from which they are carried out of the building through a four ft. diameter duct.

Air and Nitrogen Supply System

The air pressure system is composed of a low-pressure compressor and desiccant-type dryer. This system is capable of providing low pressure air (less than 150 Psi) to charge the wind tunnel. Facility control subsystems, consisting of
the main starting valve (SSV), secondary flow diaphragm cutter (PEC) and Flex-Flo Valve (FFV) are all operated using high pressure nitrogen, and they are individually pressurized to the required pressure prior to the tunnel firing.

Control System

The control board (Fig. 2.7) is composed of several pressure regulators and gauges, which are used to monitor the pressure in the wind tunnel as well as those in the subsystems. Each subsystem is equipped with a separate accumulator operated by actuators and also several dump and vent valves which are used to relieve the pressure in case of tunnel over pressurization.

Data Acquisition and Facility Control Computer

Facility operation and control, data acquisition, and data processing are accomplished by a custom-designed micro-computer system utilizing two Xerox model 820 circuit boards. A schematic of the system is shown in Fig. 2.8. The Data acquisition/control computer (DAC) controls the operation of the tunnel by sending electrical signals to solenoid valves to control the subsystem valves to initiate the main tunnel flow as well as the flow through the secondary system. The time delay between solenoid firings is set by storing them in their proper memory address prior to the tunnel firing. The DAC computer also collects data from a 24-channel Kulite high-
frequency pressure transducer system, in some pre-determined
time interval increments as required by the particular
experiment. The stored data is then transferred to the main
computer for data reduction and permanent storage on magnetic
disks.

Theory of Operation

The basic concept of a Ludwieg tube wind tunnel is shown
schematically in Fig. 2.9. Prior to the tunnel firing, the
wind tunnel (charge tube, nozzle, test section etc.) is
pressurized to a pre-determined pressure as required by the
desired run Reynolds number. When the main starting device
(SSV) is opened, the flow through the wind tunnel resembles
that of the flow in the driver section of a shock-tube. The
unsteady expansion wave generated at the starting valve moves
through the test section into the charge tube and sets the
stagnant high pressure air in motion. Such a flow cannot come
to a steady state condition until the flow through the plenum
chamber surrounding the test section reaches an equilibrium
value. When the main starting valve is opened, the total flow
rate ($\dot{W}_{T}$) consists of the combined flow rates entering the
test section from the plenum ($\dot{W}_{1a}$) and the charge tube ($\dot{W}_{2}$).
As the chamber pressure drops and the pressure ratio across
the porous walls decreases, the flow rate out of the charge
tube must increase to maintain a nearly fixed total flow rate.
Eventually the flow rates $\dot{W}_T$, $\dot{W}_{1a}$, and $\dot{W}_2$ all become steady. This transient nature of the Ludwieg tube tunnel is called the starting process. From the description of the unsteady starting process, it is evident that increasing the flow rate out of the plenum chamber can have a marked effect on the tunnel starting time. Theoretically, the tunnel steady state run time is limited by the charge tube length to about 180 milliseconds, which is the time required for the expansion wave to travel down the length of the charge tube, reflect at the charge tube end and return to the test section. During this period, the test section is ideally at a constant aerodynamic flow condition. Since the unsteady starting process is included in this time, it is highly desirable to reduce the starting time as much as possible, thus affording the longest possible useful testing time (Ref. 31). The minimization of the starting process can, for example, be accomplished by using an auxiliary bleed system with an opening device independent of that of the main starting valve. In the UTA transonic wind tunnel facility, such auxiliary flows are obtained by using three independent devices as follows:

1. A plenum exhaust system (PEC), which is opened by rupturing a diaphragm at a prescribed time during the starting process.

2. A controlable plenum exhaust system (FFV) which can provide an excessive plenum exhaust flow during the starting
process and be throttled to a closed position required by the steady state run.

3. An ejector flap system downstream of the test section, which can be opened to increase the effective flow area between the test section and the plenum chamber during the tunnel start while provisions are provided to close them to a lower value as required by the steady state run time.

The plenum exhaust manifold, which contains the plenum exhaust diaphragm, is connected by a 6 inch I.D. line approximately 5 feet long to a ball valve which controls the quantity of secondary mass flow during the steady portion of the run. This valve together with the Flex-Flo Valve allow for an excessive mass flow in order to reduce the pressure inside the plenum chamber, and if properly used, the excess exhaust greatly reduces the tunnel start time.

The ideal use of the ejector flap system, can be accomplished by opening the flap height to its maximum opening (0.9 in.) during the tunnel start time, and reducing the flap height opening to a lower value during the steady run time for low transonic Mach numbers, while it can be left open during the steady state time to obtain higher transonic Mach numbers. The effect of ejector flap setting on the attainable Mach numbers is investigated to a limited extent, and the results will be presented in Chapter III.

Once the tunnel is started, the steady state run
conditions are defined by the interaction of the ball valve opening when the plenum diaphragm is ruptured (secondary mass flow), the ejector flap opening, test section wall porosity, the main valve characteristics and to a limited extent to the tunnel operating pressure. It has been experimentally shown (Ref. 30) that the effect of wall porosity setting in determining the mean test section flow is of secondary importance.

Ideally, it is highly desirable to arrange the time delay settings of the different valves so that the expansion waves generated by the valve openings reach the test section simultaneously (Fig. 2.10). This will result in the longest possible useful running time. If either the expansion wave generated by the main starting valve (SSV) or that produced by the diaphragm rupture arrives earlier, the result is a decrease in the useful steady state run time.
*NOTE: All Dimensions In Inches

Fig. 2.2 Cross sectional view of the wind tunnel
Fig. 2.3 Photograph of the nozzle section

Fig. 2.4 Photograph of the test section
Fig. 2.5 Schematic of plenum exhaust system.
Fig. 2.6 Schematic of plenum exhaust ball valve orifice
Fig. 2.7 Schematic of the control board
Fig. 2.8 Facility control and data acquisition system
Fig. 2.9 Illustration of the Ludwieg tube starting process

a. $\Delta t_p < \text{Optimum}$

b. $\Delta t_p > \text{Optimum}$

c. $\Delta t_p = \text{Optimum}$

Fig. 2.10 Illustration of the optimum starting process (from Ref. 30)
CHAPTER III

WIND TUNNEL CALIBRATION

The experiments conducted for the purpose of calibrating the wind tunnel included test section pressure (Mach number) measurements by means of a centerline pipe equipped with several pressure taps spanning the charge tube, nozzle, test section and the model support section. The objectives of these experiments were to develop a relationship between the test section and plenum chamber Mach numbers, and to examine the uniformity of the test section Mach number distribution. Also attempts were made to obtain an optimum time delay and subsystem pressure settings of various valves, which results in maximum steady run time in the tunnel. These attempts resulted in the achievement of longer steady state run time than those of the previous experiments at AEDC. All of the experimental investigations, were conducted for a fixed test section wall porosity setting of approximately 7 percent. The effect of secondary flow on the test section Mach number distribution and its axial uniformity was studied by means of operating the wind tunnel at different ejector flap openings.

The centerline pipe (CLP) used during these experiments
contained several static pressure orifices at various locations along the nozzle and test section (Fig. 3.1). Total pressure and temperature measurements in the charge tube combined with the static pressure measurements from the centerline pipe results in a direct calculation of the test section Mach number at several axial locations according to the following equation for isentropic flow of a perfect gas

$$M_\infty = \left( \frac{2}{\gamma - 1} \right) \left[ \frac{P_t}{P_\infty} \left( \frac{\gamma - 1}{\gamma} \right) \right]^{1/2}$$

(3.1)

The front portion of the centerline pipe is secured in the charge tube, while the rear part mounts in the model support section, just downstream of the test section. The pressure tubing from the centerline pipe orifices is connected to the outside of the tunnel by means of an instrumentation access port on the model support section. Two total pressure probes and one static pressure orifice are located in front of the nozzle section to yield the charge tube total pressure and Mach number. Two total pressure probes are used to help account for any irregularities that might exist in the flow. These two total pressure measurements are mechanically averaged by joining the tubing together before again splitting to two separate transducers. One transducer serves as the total pressure reading during the actual run, while the other is connected to a digital voltmeter, located in the control room, serving as an indicator for monitoring the charge tube
pressure during the tunnel charge up procedure. The pressure transducer used to measure the stagnation pressure is also used as a reference transducer against which all the other transducers are calibrated during the wind tunnel charge up. Due to the extreme importance of this transducer in obtaining experimental data, it is constantly calibrated to account for any possible drift that might occur. A typical calibration plot of this transducer is shown in Fig. 3.2.

During the tunnel charge up, the transducer output from each orifice and the DVM reading from the master transducer are recorded for several charge tube pressures. The DVM reading from the master transducer is then used to obtain the pressure from a calibration plot similar to that of Fig. 3.2. These pressures and the individual pressure transducer outputs on the model are then used to construct a linear least square curve fitting of pressure vs. transducer output voltage. The linear equations resulting from the above procedure for each transducer yields the experimental pressures during the steady state run time. The accuracy of this procedure is determined for each individual pressure transducer in use during the data reduction and typical linearity errors are generally less than 0.3 percent.

The time delay settings and the optimum operating pressures of various control valves were investigated in the course of this program and each time some improvements were made. As already mentioned, the duration of the useful run
time for the UTA transonic wind tunnel facility can be increased by a careful arrangement of time delays such that the expansion waves generated from both the main starting valve and that of the plenum exhaust cutter arrive at the test section at the same time. The nearly equal distances from the main valve and the plenum exhaust cutter to the test section, suggests that both valves should be actuated simultaneously and therefore no time delay is required between their openings, but practically since the sliding-sleeve valve opens slower than the plenum exhaust cutter, some 20 - 25 milliseconds time delay is required between their openings. Determining an optimum time delay is usually an iterative procedure and no systematic procedure is known for obtaining the proper time delay settings.

A similar approach, but a rather less complicated procedure, was used to determine the proper pressure settings required to actuate different subsystem valves. Usually this task is accomplished in a few runs and once they are set, they need not to be altered unless major modifications to the tunnel are performed. Table 3.1 indicate a typical pressure and time delay settings.

The axial uniformity of the test section Mach number is a prime measure of the test section flow quality in transonic wind tunnels. The static pressure measurements using the centerline pipe is especially well suited for such
investigations since it contains pressure orifices at several axial locations along the test section. The relatively small blockage of this model (about 0.6 percent) allows fairly accurate transonic flow measurements with minimum interference effects.

Repeatability of the static pressure measurements on different models was considered to be of vital importance throughout the course of these investigations and they were studied extensively. The wind tunnel repeatability tests were performed by operating the facility at several identical conditions (i.e., ball valve setting, operating pressure etc.). A limited number of runs at different charge tube pressures were conducted in order to determine data for various Reynolds numbers, and their effects on the test section Mach number and uniformity.

The effect of the secondary flow through the ejector flap system, and the range of test section Mach numbers attainable for different flap opening heights, was investigated for four flap settings of 0, 0.2, 0.45 and 0.9 inches, where a 0 flap opening corresponds to no flow through the flaps and flap opening height of 0.9 in. results in the maximum attainable auxiliary flow. The primary purpose of these flaps are to reduce the starting time by allowing a portion of the flow to be exhausted through the openings, thus reducing the chamber pressure. However as will be discussed later, flap openings in excess of about 0.2 in. could be used
if low supersonic Mach number flows are desired. But since the proper use of ball valve opening area can produce a sufficient auxiliary flow for most transonic Mach numbers of interest. Ideally it is desirable to open the flaps only during the tunnel start and then close them to a lower opening area during the steady state portion of the experiment.

A relationship between the test section and plenum chamber Mach numbers was determined in this phase of the experimental investigation. This relationship was determined for a range of ball valve settings (Mach numbers), ejector flap openings and two different Reynolds numbers. The results of these studies are used in subsequent experiments to determine the free stream Mach number. Determination of the test section Mach number in this manner will eliminate the use of a secondary probe to determine the test section Mach number and avoid the possibility of an excessive blockage which is very critical at transonic flows. The plenum chamber Mach number was determined by using a similar equation to that for calculating the test section Mach number (i.e., equation 3.1) as follows:

\[ M_{pc} = \left( \frac{2}{\gamma - 1} \right) \left[ \left( \frac{P_t}{P_{pc}} \right)^{\left( \frac{\gamma - 1}{\gamma} \right)} - 1 \right]^{1/2} \]  (3.2)

where the subscript pc represents the plenum chamber.
Test Results

The effects of proper time delay and pressure settings of the various control valves on the starting characteristics of the facility can be summarized in time history plots of the total pressure measurements in the charge tube. These traces are illustrated in Figs. 3.3 - 3.5 for the entire run (including the unsteady starting portion), which clearly indicates the importance of achieving a proper wind tunnel start on the wind tunnel useful run time. Fig. 3.3 shows the effect of low subsystem pressure on the main starting valve. The wind tunnel is fired at a charge tube pressure of 135 psia at time $t=0$, but the valve is not fully opened until $t=120$ ms, at which time the unsteady expansion begins. Unfortunately, before steady state flow can be established in the test section, the reflected expansion wave has already reached the charge tube, resulting in termination of the steady state flow in the test section. Figure 3.4 on the other hand indicates a constant total pressure from 80 to 160 ms, with the starting process lasting about 80 ms. Figure 3.5 indicates the results of a successful run with the useful run time in excess of 130 ms and the starting time of about 40 ms. This in fact is the result of both ideal time delay setting and also using adequate pressure level in order to open the main starting valve within a minimum opening time.

A summary of the test section calibration results are
presented in Figs. 3.6 - 3.9 for several ball valve settings and four flap openings. Figure 3.6 shows a Mach number variation in the range of 0.68 - 0.88, corresponding to ball valve openings of 9 - 16. The ball valve position varies from 1 to 23, where position 1 corresponds to the maximum opening area and a ball valve setting of 23 represents a completely closed secondary exhaust area. The test section static pressure measurements were made at five axial positions spanning a distance from 14 to 20 in. downstream of the test section entrance. The axial uniformity of the test section Mach number for a flap setting of 0 in., as shown in Fig. 3.6, is fairly good with a total deviation of less than 1 percent. Similar results are seen in Figs. 3.7 - 3.9 for flap openings of 0.2, 0.45, and 0.9 inches, respectively. The results of the above figures also indicate the difficulties of achieving low transonic Mach numbers for ejector flap openings in excess of 0.45 in. This behavior is clarified in Figs. 3.8 and 3.9 for flap settings of 0.45 and 0.90 in, where a ball valve setting of 18 (almost no auxiliary flow through the ball valve opening), produces Mach numbers in excess of 0.90. Such high Mach numbers cannot be reduced by any other means. This in fact verifies the earlier discussion in using the ejector flaps only to start the tunnel.

The relationship between the plenum chamber and test section Mach numbers is shown in Figs. 3.10 - 3.13. For flap openings in excess of 0.45 in, almost no correction is
required and the test section and plenum chamber Mach numbers are identical for a given ball valve setting; but as discussed earlier, difficulties in achieving Mach numbers below 0.9 for such flap openings limits the use of the high flap openings unless flows in the high transonic and low supersonic Mach number range are desired. On the other hand a range of "true" transonic Mach numbers are attainable for the case of no auxiliary flow through the ejector flaps. The correction factor to obtain the test section Mach number was found to slightly depend on the ball valve position (i.e., free stream Mach number). The difference in Mach numbers of the test section and that of the plenum chamber is seen to vary from 0.04 for a test section Mach number of 0.68 and to a low value of 0.01 at a free stream Mach number of 0.9.

Before leaving this subject some comments concerning the pressure stabilization and system response-time lags are in order. Due to the system time lag problems, the data obtained from any wind tunnel facility has limitations below the theoretical predictions based on the characteristics of the wind tunnel itself. Study of the system time lag shows that response time in a pressure measuring system incorporating capillaries is a function of the orifice pressure, the initial pressure differential, and the system volume (Ref. 32). It is directly proportional to the capillary length, and is inversely proportional to the capillary inside diameter according to the following equation,
\[ t = \frac{128 \mu t_e}{\pi d^4} \left[ \frac{V_1}{p_1} \ln \left( \frac{p_0 - p_1}{p - p_1} \right) \frac{(p_0 + p_1)}{(p + p_1)} + \frac{3V_d}{(p_0 - p_1)} \ln \left( \frac{p_0}{p - p_1} \right) \right] \]

where

\( p_0 = \text{initial pressure, lb/ft}^2 \)

\( p_1 = \text{final pressure, lb/ft}^2 \)

\( V_1 = \text{entire air volume of system from orifice to and including measuring device at } p = p_1 \text{ (i.e., } t = \infty) \text{ ft}^3 \)

\( V_d = \text{volume displaced by deflection of diaphragm, ft}^3 \)

\( d = \text{common diameter for a set of connected capillaries, ft} \)

\( l_e = \text{length of capillary of diameter } d \text{ which is equivalent in flow resistance to total resistance of all series-connected capillaries in the system, ft} \)

\( \mu = \text{coefficient of viscosity of air in system, lb-sec/ft}^2 \)

The pressure time lag problem is even more critical if a short duration facility is being used, since the steady run time can be greatly reduced by a slow response system. Study of the charge tube total pressure time history (Fig. 3.5), indicates a constant stagnation pressure for a period of about 130 millisecond. On the other hand the experimental data indicate that the pressure response time for the centerline pipe is of longer duration. Both the tubing length and the pressure differentials of the initial and final stages are partially responsible for this problem. Since the charge tube total and static pressure transducers are located just outside of the
tunnel, on the surface of the charge tube, a relatively short tubing length is used. Also for a tunnel starting pressure of 135 psia, the total pressure drop of the stagnation pressure as a result of the unsteady expansion is on the order of 30 psi, while the test section flow can be subject to static pressure drops in excess of 70 - 80 psi, depending on the free stream Mach number.

The system time lag response problem is even more pronounced in transonic flows, since relatively small model sizes are required in order to reduce the blockage effects. The size limitations of the testing model directly results in small orifice diameters on the model, which is known to have a marked effect on the system response time. In fact the experimental studies of the present investigation indicates that this problem is more serious than the combined effects of the tubing length and pressure differential problems discussed earlier. The model response lag problem will be discussed in more detail in Chapter IV.
Fig. 3.2 Calibration of the master transducer

Fig. 3.3 Illustration of a poor wind tunnel start
Fig. 3.4 Illustration of an average wind tunnel start

Fig. 3.5 Illustration of optimum wind tunnel start
Fig. 3.6 Uniformity of test section Mach number for $\delta_x=0$ in.

Fig. 3.7 Uniformity of test section Mach number for $\delta_x=0.2$ in.
Fig. 3.8 Uniformity of test section Mach number for $\delta_f = 0.45$ in.

Fig. 3.9 Uniformity of test section Mach number for $\delta_f = 0.9$ in.
Fig. 3.10 Test section calibration for $\delta_x=0$ in.

Fig. 3.11 Test section calibration for $\delta_x=0.2$ in.
Fig. 3.12 Test section calibration for $\delta_f=0.45$ in.

Fig. 3.13 Test section calibration for $\delta_f=0.9$ in.
<table>
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<th>SSV Pressure (Psig)</th>
<th>PEC Pressure (Psig)</th>
<th>FFV Pressure (Psig)</th>
<th>Time delay SSV-PEC (msec)</th>
<th>Time delay PEC-FFV (msec)</th>
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CHAPTER IV

AIRFOIL CALIBRATION

Prior to performing the vortex airfoil interaction tests, several experiments were conducted to obtain the "clean" airfoil pressure distribution as a base test for the future interaction problems. A two-dimensional model of the C-141 wing at 38.9 percent of the semi span and a NACA 0012 airfoil section, both with chord length of 2 in., were used. Pressure measurements were performed on the aforementioned airfoils in order to obtain the chordwise pressure distribution on the upper surfaces of the airfoils. Attempts were made to obtain the data for chord Reynolds numbers of 3.8 and 5.5 million, and test section free stream Mach numbers ranging from 0.68 to 0.90.

The airfoils were mounted vertically at the tunnel CMR (Fig 2.2) spanning the entire height of the wind tunnel, approximately 9 chord lengths downstream of the test section entrance. The models had a 3.2 percent blockage based on the projected frontal area in the test section, which is a substantial increase over that of the centerline pipe used for the test section calibration tests. This blockage is
comparatively large but it can not be reduced since the models are already of minimum practical size. The factors involved in limiting the model size are the number of pressure orifices required, the wind tunnel blockage and the model allowable stress level. To alleviate some of the model blockage and the test section wall interference effects, the two dimensional airfoil sections were mounted approximately 3/4 in. off centerline with the instrumented side (upper surface) of the airfoil further away from the test section wall.

Test Results

Calibration of the C-141 Airfoil

Only a limited number of runs were conducted utilizing this airfoil, and all the runs performed were for an airfoil angle of attack of 0. The C-141 airfoil (Fig 4.1) was equipped with 15 static pressure ports. The chordwise location of the orifices are shown in Fig. 4.1 and are tabulated in Table 4.1. The pressure tubes were 0.045 in. I.D. with six 0.02 in. diameter feeder holes each, which results in averaging the chordwise pressure distribution along a 1 in. span of the airfoil at the mid span position of the airfoil.

It was reported by Starr (Ref. 33), that some disagreement between the pressure distribution of the present airfoil section and a similar airfoil section tested in the
AEDC, Propulsion Wind Tunnel (FWT) (Ref. 34) existed for the leading 20-25 percent chord. This discrepancy was thought to be as a result of scale differences between the two airfoil sections. To clarify the scale differences, careful inspection of the existing C-141 airfoil and that of the FWT airfoil were performed and are magnified in Fig. 4.2, as taken directly from Ref. 33. The present airfoil measurements indicated to be approximately 1 to 5 thousands of an inch thicker than the one used at FWT for the leading 1/3 of the upper and lower surfaces. These scale differences are equivalent to about 1 to 4 percent of the local airfoil thickness. The FWT airfoil shape was reported to be more nearly dimensionally correct. As the test progressed and appropriate instrumentation swaps and checks were made, the results of these investigations indicated that port number 3, corresponding to the X/C of 20 percent, to be partially blocked, and the pressure readings from the above port were discarded, thus limiting the pressure measurements to 14 ports along the airfoil chord.

The experimental pressure distribution data are presented in terms of pressure coefficient \(C_p\) plots throughout this report. The procedure used to determine the \(C_p\) values are similar to the Mach number calculations of the previous chapter, where the wind tunnel calibration was discussed. The test section free stream Mach number is determined by introducing a correction factor to the plenum chamber Mach number, and the pressure coefficients are then obtained using
the following equation

\[ C_p = \frac{2}{\gamma M^2} \left( \frac{P}{P_\infty} - 1 \right) \]  \hspace{1cm} (4.1)

All of the calculations needed to produce the results, are performed using a fortran program "AIRFOIL", a listing of which is given in Appendix A.

The experiments conducted on the C-141 airfoil section were limited to pressure measurements on the upper surface of the model and an airfoil angle of attack of 0. In the course of the airfoil calibration experiments, free stream Mach numbers were varied to simulate subcritical, supersonic, and to the highest attainable degree a critical flow over the airfoil. The free stream Mach number which produces such flow over the airfoil in general is not known a priori, and it must be determined experimentally. Wind tunnel repeatability was also of concern and attempts were made to reproduce runs at nearly identical operating conditions. Throughout the present investigation, the upper limit of the wind tunnel operating pressure was limited to about 135 psia, by the low pressure compressor available. The aforementioned pressure results in a free stream stagnation pressure of approximately 100 psia, and a chord Reynolds number of about 5.5 million. The lower range of operating pressure was restricted to 100 psia \((P_0=80 \text{ psia})\), due to the noise level of the Kulite pressure transducers.
used. The former operating pressure produces Reynolds numbers in the range of 3.5-3.8 million based on a 2 in. chord airfoil. The exact value of attainable Reynolds number depends on free stream Mach number and the wind tunnel operating temperature.

The pressure distribution on the upper surface of the airfoil for a test section Mach number of 0.73, which corresponds to a subcritical flow over the airfoil, is shown in Fig 4.3. This pressure distribution confirms our earlier discussion concerning the unusual behavior near $X/C = 0.25$. The pressure distribution on the upper surface of the C-141 airfoil for a free stream Mach number of 0.75 corresponding to a critical flow over the airfoil is illustrated in Fig 4.4. A series of experiments were conducted for free stream Mach numbers corresponding to supercritical flows over the airfoil, and typical pressure distributions for such flows are presented in Figs. 4.5 and 4.6 for free stream Mach numbers of 0.77 and 0.80, respectively.

Historically, the pressure distribution of the C-141 airfoil section has played an important role in developments and advances of transonic wind tunnel testing in general. Earlier some difficulties regarding the position of the shock wave and the Reynolds number scaling effects were detected in "free flight" tests that were not predicted by the wind tunnel data. A trailing edge separation was observed, which was
thought to be a possible "Reynolds number" effect by several investigators. The transonic flow measurements of Refs. 34 and 35 are free of such trailing edge separation and they attribute the discrepancy at the proximity of engine nacelles to the wings, and concluded that the trailing edge separation on the C-141 airfoil may have been dominated by some other factors other than the airfoil shape itself.

The reason for discussing the C-141 pressure distribution is the fact that there already exists a controversial behavior in the pressure distribution of the C-141 airfoil, and since it is intended to validate the experimental data of the present analysis, the use of such airfoil will not in general be of significant importance to the results of these studies. The present investigations are not intended to attack the C-141 pressure distribution nor the experiments were carried in that direction. The only experimental data available for the C-141 airfoil are those of Ref. 33, and a successful matching of the results of the present analysis with those of the aforementioned paper for supercritical flows was not achieved. In Ref. 33, the comparison of Ludwig tube data (PILOT HIRT), to those of the FWT-4T and PWT-16T tunnels at AEDC, for an airfoil angle of attack of zero, indicated a fairly good agreement except near the leading edge of the airfoil, where the scale differences were reported to be the reason. However, for an airfoil angle of attack of 2 degrees the comparisons between the data of
PILOT HIRT and both PWT-4T and PWT-16T were extremely poor.

Considering the above discussion, the C-141 airfoil pressure distributions for several Mach numbers are presented here without any comparison. The accuracy of the airfoil pressure distribution data from the Ludwieg tube tunnel will be considered in the following section, where the results for a NACA 0012 airfoil section, for which more reliable and accurate experimental and theoretical data are available, will be presented.

The Reynolds number effect on the airfoil pressure distribution is shown in Fig 4.7. The results of the present investigation indicates no appreciable Reynolds number effect in the range of 3.5-5.5 million, and only a slight difference in the "post shock" pressure distribution is seen, which is most likely caused by the small Mach number differences of the runs. The repeatability of the airfoil pressure distribution is shown in Fig. 4.8, where two runs of identical Mach and Reynolds numbers are compared with good agreement.

Calibration of the NACA 0012 Airfoil

More detailed airfoil pressure measurements were conducted using a NACA 0012 airfoil (Fig. 4.9 and Table 4.2). The use of a NACA 0012 airfoil section for the remainder of these investigations is justified by considering the extensive amount of experimental and computational data available for
this airfoil. As was the case for the C-141 airfoil section, pressure measurements for a range of Mach numbers representing the subcritical, critical, and supercritical flows over the airfoil were obtained. The airfoil was equipped with 13 static pressure ports on only the upper surface, and similar to the C-141 model, there were 6 orifices for each chord-wise station which were averaged prior to connecting to each individual pressure transducer.

The difficulties discussed earlier concerning the pressure response time on the model were even more serious in the airfoil calibration studies, since due to restricted model size one is limited to small orifice sizes on the airfoil, which drastically increases the pressure response time. With the limited wind tunnel steady state run time, the slow response may produce some difficulty in obtaining accurate pressure measurements if the starting time of the tunnel is not optimized. Subsequent improvements on maximizing the useful run time were performed by a combination of starting procedure, time delay settings, and optimization of the instrumentation. Fig. 4.10 illustrates the pressure distribution for a Mach number of 0.73, and a chord Reynolds number of 4.3 million, where each symbol represents a distinct time during the run, with a 5 millisecond time interval between each pressure trace. Fairly stabilized pressure readings are obtained for the last 25 ms of the run prior to termination of the steady state time by the reflected wave.
Repeatability and accuracy of the experimental data from the UTA transonic wind tunnel facility for a range of Mach numbers were compared with experimental data from Ref. 36, which are extensively used in the literature as the base data for validating new theoretical and computational schemes. The experimental data of the aforementioned report contains pressure distributions for the NACA 0012 airfoil section for a wide range of Mach numbers. These data are from experiments conducted at the S3 Modane (with a test section of 0.78 x 0.56 m) and NAE 5-ft x 5-ft transonic wind tunnels. These sources contain very accurate measurements for a range of transonic Mach numbers, and all of the comparisons are made against the corrected results rather than the actual measurements. It was also attempted to match the run Mach numbers as closely as possible. However, some of the experimental Reynolds numbers for the data of Ref. 36 are much higher than the experimental data of the present investigation.

Figure 4.11 indicates the comparison between the data of the present investigation and those of Ref. 36, for a free stream Mach number of 0.69. The corrected angle of attack taken from Ref. 36, is shown to be slightly different from the intended value of 0. In general the comparison is good, and the small differences downstream of the mid chord position is thought to be due to slight differences between the Mach numbers and the angles of attack. A somewhat better agreement between the results of the present study and those of Ref. 36
is shown in Fig. 4.12 for a free stream Mach number of 0.703. Acceptable comparisons are shown in Figs 4.13 and 4.14 for test section Mach numbers of 0.76 and 0.78 (corrected Mach numbers of 0.756 and 0.776 respectively), as reported in Ref. 36), representing critical and supercritical flows on the airfoil. For the supercritical case it may be seen that the shock wave at X/C = 0.30 is captured close to the position suggested by the data of Ref. 36. Although the data of the present experiments for the NACA 0012 airfoil section indicate reasonable accuracy in comparison with the published data for Mach numbers up to about 0.78, the trends are not acceptable for higher Mach numbers. This is illustrated in Fig. 4.15 where the pressure distributions for a test section Mach number of about 0.80 are compared. This figure indicates some strange behavior near X/C = 0.30, and in the neighborhood of the expected shock wave (X/C = 0.40). The profile suggests the existence of a weak compression wave near X/C = 0.30 followed by what appears to be a separation region and a secondary weak compression wave before following the distribution of the data of Ref. 36 near X/C = 0.60. The exact cause for this behavior is not known at present time, but after considering several possible reasons for such behavior, the model shape at the location of the pressure orifices is thought to be a major cause of the problem. Another point worth noting is the fact that downstream of X/C = 0.50, a fairly good matching can be seen, which suggests that the spurious compression wave is not
a result of downstream conditions, which is a major cause for dislocation of shock waves in transonic flows.

Similar results were observed for higher free stream Mach numbers, and for Mach numbers in excess of 0.85, the aft portion of the airfoil contains what appears to be a series of flow separation and reattachment regions (Fig. 4.16).

Since the results of the airfoil calibration data were to be used later for investigating the vortex airfoil interaction experiments, it was highly desirable to have repeatable runs under similar operating conditions. Figure 4.17 indicates the chordwise pressure coefficient on the NACA 0012 airfoil for 3 different runs with M=0.70, which indicates repeatability within 1 percent. Generally, repeatability was considered to be of significant importance and it was verified for any new test configuration.

Due to the problems encountered in investigating the high Mach number flows, and since a firm understanding of the above behavior and their exact causes were not determined, in the subsequent experiments the high Mach number runs (generally above 0.80) were not investigated in great detail, and the main emphasis was placed on the lower Mach number runs for which the results agree with those of the published data.

As was the case for the experiments of the previous
chapter, the highest possible Reynolds number was limited to about 6 million (based on the 2 in. airfoil chord), and the lowest Reynolds number tested was about 3.5 million, due to the limitations of the data acquisition system and the pressure transducers. Figure 4.18 indicates the pressure distribution for two Reynolds numbers of 3.8 and 5.2 million. Similar to the results of the C-141 airfoil test, there appears to be no appreciable Reynolds number effect on the airfoil pressure distribution.

The system time lag response problem discussed earlier was more critical in conducting pressure measurements on the airfoil calibration experiments, due to the small orifice sizes on the airfoil section. This was partially resolved by optimizing the wind tunnel start time. For steady state experimental studies, similar to those of the present investigation, the system time lag response problem is not as critical as for unsteady flow simulations for which different approaches must be taken in order to obtain acceptable data (Chapter VII). This is illustrated in Fig. 4.19 in the form of time history plots of charge tube total and static pressures, static pressure in the plenum chamber and also a typical port on the surface of the airfoil. Although the charge tube static and stagnation pressure traces indicate a relatively long steady state time, pressure measurements from the plenum cavity and airfoil surface result in perhaps the minimum
acceptable steady time for practical purposes.
Fig. 4.1 Photograph of the C-141 airfoil section

--- Prescribed
--- Pilot Hirt 2" Chord as machined

Typical orifice and tube size (15 places)

Fig. 4.2 Illustration of scale differences of C-141 airfoil
Fig. 4.3 Pressure coefficient for C-141 airfoil:

$M=0.73$, $Re=5 \times 10^6$

Fig. 4.4 Pressure coefficient for C-141 airfoil:

$M=0.75$, $Re=5.1 \times 10^6$
Fig. 4.5 Pressure coefficient for C-141 airfoil:

\[ M=0.78, \ Re=5.3 \times 10^6 \]

Fig. 4.6 Pressure coefficient for C-141 airfoil:

\[ M=0.80, \ Re=3.9 \times 10^6 \]
\( Re=3.3 \times 10^6 \quad M=0.805 \quad \Delta \quad Re=5.3 \times 10^6 \quad M=0.795 \quad \Diamond \quad Re=5.3 \times 10^6 \quad M=0.795 \)

Fig. 4.7 Effect of Reynolds number variation: \( M=0.80 \)

\( \square \quad \text{Run 127} \quad M=0.855 \quad \Delta \quad \text{Run 126} \quad M=0.86 \)

Fig. 4.8 Repeatability of pressure distribution:

\[ M=0.86, \quad Re=5.5 \times 10^6 \]
Fig. 4.9 Photograph of NACA 0012 airfoil section

![Photograph of NACA 0012 airfoil section]

Fig. 4.10 Illustration of pressure stabilization on the NACA 0012 airfoil: $M=0.72$, $Re=5.1 \times 10^6$
Fig. 4.11 Comparison of measured upper surface pressure distribution with data of Ref. 36 for M=0.69

Fig. 4.12 Comparison of measured upper surface pressure distribution with data of Ref. 36 for M=0.70
Fig. 4.13 Comparison of measured upper surface pressure distribution with data of Ref. 36 for $M=0.76$

Fig. 4.14 Comparison of measured upper surface pressure distribution with data of Ref. 36 for $M=0.78$
Fig. 4.15 Upper surface pressure distribution:

\[ M = 0.80, \ Re = 5.6 \times 10^6 \]

Fig. 4.16 Upper surface pressure distribution:

\[ M = 0.85, \ Re = 5.7 \times 10^6 \]
Fig. 4.17 Repeatability of pressure distribution:

\[ M = 0.72, \ Re = 5.1 \times 10^6 \]

Fig. 4.18 Effect of Reynolds number variation on the pressure distribution of NACA 0012 airfoil for \( M = 0.70 \).
Fig. 4.19 Illustration of the system time lag response
Table 4.1 Port locations for the C-141 airfoil section

<table>
<thead>
<tr>
<th>Port Number</th>
<th>X/C</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.05</td>
</tr>
<tr>
<td>2</td>
<td>0.10</td>
</tr>
<tr>
<td>3</td>
<td>0.15</td>
</tr>
<tr>
<td>4</td>
<td>0.20</td>
</tr>
<tr>
<td>5</td>
<td>0.25</td>
</tr>
<tr>
<td>6</td>
<td>0.35</td>
</tr>
<tr>
<td>7</td>
<td>0.45</td>
</tr>
<tr>
<td>8</td>
<td>0.50</td>
</tr>
<tr>
<td>9</td>
<td>0.55</td>
</tr>
<tr>
<td>10</td>
<td>0.60</td>
</tr>
<tr>
<td>11</td>
<td>0.65</td>
</tr>
<tr>
<td>12</td>
<td>0.75</td>
</tr>
<tr>
<td>13</td>
<td>0.80</td>
</tr>
<tr>
<td>14</td>
<td>0.90</td>
</tr>
</tbody>
</table>

Table 4.2 Port locations and airfoil thickness for the NACA 0012 airfoil section

<table>
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<tr>
<th>Port Number</th>
<th>X/C</th>
<th>Y/C</th>
</tr>
</thead>
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<tr>
<td>1</td>
<td>0.025</td>
<td>0.026</td>
</tr>
<tr>
<td>2</td>
<td>0.10</td>
<td>0.047</td>
</tr>
<tr>
<td>3</td>
<td>0.15</td>
<td>0.053</td>
</tr>
<tr>
<td>4</td>
<td>0.20</td>
<td>0.057</td>
</tr>
<tr>
<td>5</td>
<td>0.30</td>
<td>0.060</td>
</tr>
<tr>
<td>6</td>
<td>0.40</td>
<td>0.058</td>
</tr>
<tr>
<td>7</td>
<td>0.50</td>
<td>0.053</td>
</tr>
<tr>
<td>8</td>
<td>0.55</td>
<td>0.050</td>
</tr>
<tr>
<td>9</td>
<td>0.60</td>
<td>0.046</td>
</tr>
<tr>
<td>10</td>
<td>0.65</td>
<td>0.041</td>
</tr>
<tr>
<td>11</td>
<td>0.75</td>
<td>0.032</td>
</tr>
<tr>
<td>12</td>
<td>0.80</td>
<td>0.026</td>
</tr>
</tbody>
</table>
CHAPTER V

VORTEX SURVEY

A thorough understanding of the vortex airfoil interaction phenomenon requires an in depth knowledge of the shape and strength of trailing vortices. Most of the interaction studies in the literature consist of the effects of trailing vortices on the following blade or blades, and very few studies have focused on the inverse problem i.e., the effect of the blades on the vortex and the structure of the vortices during the interaction problem. It is known that the vortices are under continuous deformation and modification in shape and strength throughout the flow field, and depending on the exact geometry of the interaction problem they may be subject to severe changes in size and strength.

The lack of physical understanding of the vortex structure, especially in compressible transonic flows, has limited the applicability of some excellent computational schemes in the field of computational fluid dynamics (CFD), and the fact that theoretical and computational methods generally incorporate vortex theories of some 20 - 30 years ago raises serious questions about the validity of such
schemes in studying this problem. Flow visualization techniques are still being used in order to gain more physical understanding of the vortex structure, and until the structure of trailing vortices is firmly understood, general validity of any theoretical and computational study incorporating empirical vortex fitting schemes will be highly questionable.

The general scheme used in the vortex survey experiments involved setting a vortex generator at the interface of the nozzle and test section such that the trailing vortex convects downstream. Four vortex generators of different semi-spans were used in order to investigate the effect of vortex height. The vortex generators all resembled a NACA 0012 airfoil shape and had a 2 in. chord. The vortex generating mechanism consisted of shaft and bearing assembly connected to a pitch angle indicator device outside the wind tunnel, which was used to adjust the vortex generator pitch angle prior to the wind tunnel firing. Figure 5.1 shows the vortex generator mechanism and the generating airfoil section. The vortex generators used in these experiments measured 4.75, 5.1, 5.3 and 5.5 inches in span (Fig. 5.2) and were mounted horizontally at the test section entrance.

A total pressure rake equipped with 13 stagnation ports (Fig. 5.3) was placed approximately 9 chord lengths downstream of the vortex generator. The total pressure probes were 0.045 in. in diameter with 0.125 in. spacing between probe centerlines, and the rake spanned horizontally in the test
section. The total pressure rake was secured in the test section using a probe holder and the vertical position of the rake was adjusted by means of a traversing mechanism (Fig. 5.4) prior to the tunnel firing. The vertical position of the rake could be determined with high accuracy by means of a dial indicator which was carefully calibrated prior to its use. The horizontal traversing of the rake, however, required manual pre-setting prior to the wind tunnel assembly, and for subsequent alteration of the horizontal position of the rake, partial dis-assembly of the tunnel was required.

The vortex rake survey experiments were conducted for a limited range of Mach and Reynolds numbers, and generally for vortex generator angles of 0, -4, and -8 degrees. To study the airfoil thickness effect and the flow contraction due to the presence of the downstream airfoil on the trailing vortex structure and its vertical height above the airfoil, the interacting airfoil was removed and several runs identical to those with the airfoil installed were performed. The structure of the trailing vortex after interacting with the downstream airfoil was then examined by conducting rake pressure surveys at both the leading and trailing edges of the interacting airfoil for several horizontal and vertical rake settings.

Generally the total pressure measurements using the total pressure rake was repeated until the point of minimum pressure, corresponding to the vortex center was recorded.
This may be illustrated by considering the inviscid momentum equation of a rotational flow. Although the viscosity plays an important role in the total pressure deficit in the core of the vortex, it will be neglected here for simplicity. The inviscid momentum equation (i.e., Euler's equation) in the absence of body forces may be expressed in the following form

\[
\frac{\partial V}{\partial t} + \nabla p = 0 \tag{5.1}
\]

or

\[
\rho \left( \frac{\partial V}{\partial t} + (V \cdot \nabla)V \right) + \nabla p = 0 \tag{5.2}
\]

If one assumes a steady flow of an incompressible fluid, and uses the vector identity

\[
(V \cdot \nabla)V = V \frac{\nabla^2}{2} - V \times \omega \tag{5.3}
\]

the momentum equation then becomes

\[
V(\rho + 1/2 \rho V^2) = \rho (V \times \omega) \tag{5.4}
\]

where \(\omega\) is the vorticity vector i.e.,

\[
\omega = \nabla \times V
\]
equation 5.4 then becomes:

\[ \nabla (P_t) = \rho (V \times \omega) \]  \hspace{1cm} (5.5)

A simple representative velocity profile in the core of the vortex is that of a forced vortex. The magnitude of the tangential velocity of the fluid for such a vortex, denoted by \( V_t \), is given by:

\[ V_t = r \Omega \]  \hspace{1cm} (5.6)

where \( \Omega \) is the angular velocity of the forced vortex. Introducing this into equation 5.5 and integrating from \( r_1 \) to \( r_2 \) (\( r_1 < r_2 \)) yields

\[ (P_t)_2 - (P_t)_1 = 2 (r_2 - r_1) \rho \Omega^2 \]  \hspace{1cm} (5.7)

which indicates that the total pressure is minimum at the center of a forced vortex (i.e., \( r_1 = 0 \)).

Although qualitative results concerning vortex center, trajectory, and shape can be obtained using the total pressure rake measurements, due to the limited sensitivity of such devices and the high interference effects between the vortex and the total pressure rake, quantitative values must be obtained by some other means. The use of flow visualization techniques, LV measurements, and possibly hot wire anemometry should be considered for more accurate measurements. Such
schemes are being extensively used for conducting vortex measurements (Refs. 11 and 14). Directional probes have also been suggested to cause less interference and have been used in several studies (Refs. 38 and 39). A form of directional probe was fabricated to be used to obtain a semi-quantitative measure for the data of the present experimental analysis. A five-hole cone probe with a total pressure orifice at the nose and four equally spaced ports on the cone surface (Fig 5.5) was used primarily to verify the data obtained from the total pressure rake measurements, and to yield more accurate information about the vortex viscous core size and its symmetry.

Test Results

A schematic of the test setup for the total pressure rake measurements is illustrated in Fig. 5.6. To determine the vortex core geometric position, i.e., the location of the vortex above the airfoil \( Z_v \) and along the span \( Y_v \), the total pressure rake was placed downstream of the vortex generator, with the total pressure probes spanning a segment of the flow in the \( Z \) direction. Two-dimensional mapping of the vortex was then accomplished by positioning the rake at different spanwise positions for subsequent wind tunnel firings. The procedure was continued until both the point of minimum total pressure, corresponding to the center of the vortex, and also the free stream stagnation values were
recorded, at which point the flow field dominated by the vortex was completely captured. Typical rake survey results are shown by horizontal total pressure distribution in Figs. 5.7 - 5.9 for vortex generator angles of 0, -4, and -8 degrees for a constant vertical position. The above figures indicate the rake measurements at five different times with 20 millisecond intervals after the steady state flow has been established in the test section. The results of Fig 5.7 for a vortex generator angle of zero indicate a slight total pressure deficit representing the viscous wake of the vortex generating wing. Horizontal pressure surveys through the core of the vortex for vortex pitch angles of -4 and -8 degrees indicate two obvious points regarding the vortex. First, the vortex appears to have a well defined core, with total pressure deficits on the order of 20 to 25 percent of the free stream stagnation pressure. Second, the results of the rake surveys indicate a substantial amount of unsteadiness near the vortex center. To clarify the unsteadiness in the vicinity of the vortex core, selected pressure vs time traces are shown in Figs. 5.10 and 5.11 for vortex generator angles of -4 and -8 degrees respectively. These traces, which are obtained by scanning the pressure transducer output at 1 millisecond time intervals, show considerable pressure fluctuations near the vortex center, however these fluctuations tend to damp out at distances further away from the vortex center. The fluctuations in pressure are thought to be a result of
oscillation of the vortex core about a mean position, although the possibility of probe oscillation due to strong rotational loading should not be ruled out. This problem was further investigated using the five-hole cone probe with a higher natural frequency on the probe holder with similar unsteadiness in the vicinity of the vortex center. Comparison of pressure traces for a zero vortex generator pitch angle, similar to those of previous figures, are shown in Fig. 5.12. In contrast to the results of Figs. 5.10 and 5.11, no unsteadiness is observed which confirms the above discussion of suggesting the unsteadiness to be a result of vortex core oscillations. Vertical total pressure traces may also be constructed by using the measurements of several runs for the same horizontal location on the rake. The time history plot for a vortex generator angle of -8 degrees near the vortex core is shown in Fig. 5.13, which further illustrates the total pressure fluctuations in the vertical direction near the vortex center.

Due to high pressure fluctuations of the vortex core, a mean vortex core position was obtained by time averaging the horizontal and vertical pressure traces over the steady state portion of the wind tunnel run time. This gives a steady mean value for the vortex core position, and is shown in the contour map plot of Fig. 5.14 for a vortex generator angle of -8 degrees and a free stream Mach number of 0.78. The location of the minimum total pressure point as determined by contour
maps similar to those of Fig. 5.14 is used to define to the vortex core height above the airfoil in the remainder of the present investigation.

Although the experiments with the vortex generator semi span of 4.75 in. as presented above produced satisfactory results in understanding some of the important characteristics of the compressible trailing vortex structure in transonic flows, the use of the above vortex generator for vortex airfoil interaction studies was discontinued after a limited number of interaction experiments were conducted utilizing the C-141 airfoil section. This was in large due to the relatively large vortex core height above the downstream airfoil (h/C=0.48). After studying the interaction data of the C-141 airfoil, vortex generators with semi spans of 5.1, 5.3, and 5.5 inches were fabricated to be used in the subsequent interaction tests with the NACA 0012 airfoil, in order to position the vortex closer to the surface of the airfoil. This point will be further illustrated in detail when the vortex airfoil interaction data are discussed in Chapter VI.

The remainder of the vortex studies consisted of total pressure measurements using vortex generators with semi spans of 5.1, 5.3 and 5.5 inches. The structure of the trailing vortices after interacting with the downstream airfoil was examined by conducting rake pressure surveys at both the leading and trailing edges of the airfoil. During the rake
pressure measurements at the leading edge of the airfoil, the presence of the downstream airfoil introduced some difficulties in obtaining horizontal pressure traces as the downstream airfoil prevented the total pressure rake to be spanned in the horizontal direction. On the other hand the total pressure rake could be positioned freely behind the airfoil in order to conduct pressure surveys at the trailing edge of the airfoil.

The results of these investigations are presented in the form of horizontal pressure traces. Figure 5.15 and 5.16 illustrate such pressure surveys at the airfoil leading and trailing edges respectively, for a vortex height of h/C = 0.30 (measured at the leading edge of the downstream airfoil), corresponding to a generator semi span of 5.1 in. The above data further illustrates the unsteadiness near the vortex core. Also a well defined core as well as a small region with total pressure deficit on the order of 8-10 percent of the free stream value (Fig. 5.16), corresponding to the viscous wake of the downstream airfoil, may be identified. The mean vortex position is again determined by time averaging the total pressure traces over some 80 milliseconds during the steady state portion of the run time. Such time averaged pressure readings for several vertical rake positions are used to construct the total pressure contour maps at the airfoil leading and trailing edges and are shown in Figs. 5.17-5.19 for non dimensional vortex heights (h/C) of 0.3, 0.2, and 0.10
respectively. For the case of the closest encounter (h/C=0.10), the interaction between the vortex and the wake of downstream airfoil is quiet strong (Fig. 5.19), but due to the narrow size of the airfoil wake the vortex core is easily distinguished by its wider band.

An examination of the above contour maps at the leading and trailing edges indicate a spanwise displacement of the minimum total pressure point (i.e., center of the vortex). These results are in qualitative agreement with the low-speed test results of Ref. 29, for similar interaction problems. If the vortex generator (drawn as dashed lines in the above maps) is taken as a reference line, the leading edge contour maps indicate both an upwash and also an inboard roll up of the trailing vortex. While for the trailing edge contour maps the inboard roll up (horizontal location) is not altered from the leading to trailing edges, a downward movement of the vortex center even below the vortex generator can be seen. An examination of the contour maps at the trailing edge of the airfoil indicates an increase in the spanwise deflection of the vortex core, as the height of the vortex core above the airfoil is reduced. This effect is illustrated in Fig. 5.20, which presents the measured spanwise drift of the core of the vortex as a function of the vortex core height as measured at the leading edge of the downstream airfoil. For comparison the low speed wind tunnel data of Ref. 29 is shown for a
vortex generator angle of 10°. The low speed data indicate a very similar trend for the vortex spanwise drift to the transonic data of the present study up to h/C=0.1. For h/C less than 0.1, the low speed wind tunnel data indicates a reduction in the spanwise deflection as a result of decreased vortex height.

The spanwise deflection of the vortex center is explained in Ref. 29 based on an image vortex analogy. A free vortex moves in the flow field such that its axis is aligned with a stream line. Since the vortex passes close to a solid surface (airfoil surface in this case), an image vortex may be hypothized such that the velocity component perpendicular to the surface induced by the image vortex cancels the perpendicular component of the flow induced by the free vortex at the surface of the airfoil. This image vortex induces a lateral flow component resulting in deflection of the vortex core in a spanwise direction (Fig. 5.21).

The total pressure contour map for vortex height of 0.20 (Fig. 5.18), also indicates the appearance of two distinct minimum total pressure regions at the trailing edge separated by some distance. The presence of these low total pressure regions suggests the possibility of formation of a secondary vortex caused either by the break up of the primary vortex into two vortices or by formation of a secondary vortex caused
by the separation of the flow due to vortex airfoil interaction (separation vortex). The secondary vortex observed for a vortex height of 20 percent (h/C=0.20) is not seen for the closest encounter (h/C=0.10). The exact reason for this is not known at this time, but due to the strong interaction of the vortex with the wake of the downstream airfoil, the secondary vortex might be absorbed within the low total pressure region in the wake of the airfoil, and not detectable by total pressure measurements alone.

Further investigation of the trailing vortex structure included the five-hole cone probe measurements in the vicinity of the vortex core. The procedure was similar to the total pressure rake measurements i.e., vertical sweeps through the core were conducted until free stream total pressure was reached at the nose of the cone probe. The difference between the two opposing ports on the surface of the cone is clearly both an indication of the vortex symmetry and its viscous core size. Definition of the viscous core diameter size is somewhat arbitrary but it is generally defined as the "peak" to "peak" distance. Figure 5.22 and 5.23 are produced by differencing the averaged pressures of the two opposing ports on the surface of the cone probe (i.e., P₂₋P₄, and P₁₋P₃ respectively). The non-dimensional core diameter measures approximately 0.25 (d/C=0.25), which is identical for both pressure differences. Also a fairly symmetric vortex structure
with a well defined viscous core can be seen in these figures.
Fig. 5.1 Photograph of vortex generator mechanism

Fig. 5.2 Photograph of vortex generator wing sections
Fig. 5.3 Photograph of total pressure rake

Fig. 5.4 Photograph of traversing mechanism
Fig. 5.5 Schematic of five-hole cone probe

Fig. 5.6 Experimental setup for vortex survey studies
Fig. 5.7 Horizontal total pressure survey through the vortex core: $\alpha_{VG} = 0$, $M=0.78$, $Re=5.4 \times 10^6$

Fig. 5.8 Horizontal total pressure survey through the vortex core: $\alpha_{VG} = -4$, $M=0.78$, $Re=5.5 \times 10^6$
Fig. 5.9 Horizontal total pressure survey through vortex core: $\alpha_{VG} = -8$, $M=0.78$, $Re=5.5 \times 10^6$

Fig. 5.10 Time history of total pressure rake measurements:

$\alpha_{VG} = -4$, $M=0.78$, $Re=5.5 \times 10^6$
Fig. 5.11 Time history of total pressure rake measurements:

\[ \alpha_{VG} = -8, \ M = 0.78, \ Re = 5.4 \times 10^6 \]

Fig. 5.12 Time history of total pressure rake measurements:

\[ \alpha_{VG} = 0, \ M = 0.78, \ Re = 5.5 \times 10^6 \]
Fig. 5.13 Time history of total pressure rake measurements in vertical direction: $\alpha_{VG} = -8$, $M=0.78$, $Re=5.3 \times 10^6$

Fig. 5.14 Time averaged total pressure contour map:
$\alpha_{VG} = -8$, $h/C=0.48$, $M=0.78$, $Re=5.3 \times 10^6$
Fig. 5.15 Horizontal total pressure survey at the leading edge of the airfoil: $\alpha_{VG} = -8$, $h/C=0.20$, $M=0.76$, $Re=5.4 \times 10^6$

Fig. 5.16 Horizontal total pressure survey at the trailing edge of the airfoil: $\alpha_{VG} = -8$, $h/C=0.20$, $M=0.76$, $Re=5.4 \times 10^6$
Fig. 5.17 Time averaged total pressure contour maps at leading and trailing edges: $\alpha_{VG} = -8$, h/C=0.3, M=0.72, Re=5.1 x $10^6$
Fig. 5.18 Time averaged total pressure contour maps at leading and trailing edges: \( \alpha_{VG} = -8 \), \( h/C = 0.2 \), \( M = 0.72 \), \( Re = 5.1 \times 10^6 \)
Fig. 5.19 Time averaged total pressure contour maps at leading and trailing edges: $\alpha_{VG} = -8$, $h/C=0.1$, $M=0.72$, $Re=5.1 \times 10^6$
Fig. 5.20 Spanwise drift of the vortex center:

\[ \alpha_{VG} = -8, \quad M=0.72, \quad Re=5.1 \times 10^6 \]

Fig. 5.21 Illustration of the flow field for vortex passing over the airfoil surface (from Ref. 29)
Fig. 5.22 Five-hole cone probe survey of vortex viscous core:
\[ a_{V_G} = -8, \ M = 0.75, \ Re = 5.2 \times 10^6 \]

Fig. 5.23 Five-hole cone probe survey of vortex viscous core:
\[ a_{V_G} = -8, \ M = 0.75, \ Re = 5.2 \times 10^6 \]
CHAPTER VI

VORTEX AIRFOIL INTERACTION

The parameters influencing the vortex airfoil interaction problem at transonic speeds are the flow Mach number, Reynolds number, vortex geometric position relative to the interacting airfoil, \((i.e., X_v \text{ and } Y_v)\), and vortex circulation, \(\Gamma\). A simple dimensional analysis of the steady perpendicular vortex airfoil interaction problem results in the non-dimensional parameters \(\Gamma / V_\infty C\), where \(\Gamma\) is the vortex circulation, and \(C\) is the airfoil chord length. Among the above parameters the Mach and Reynolds number effects were explicitly investigated, while the circulation was varied by changing the vortex generator angle of attack.

The geometry of the vortex airfoil interaction studies resembled the set up of the vortex structure survey investigations (Fig. 6.1). The scheme involved positioning a two-dimensional airfoil downstream of a semi-span wing (vortex generator), such that the trailing vortex from the vortex generator interacts with the airfoil. A limited number of interaction measurements were conducted using the C-141 airfoil section for which the calibration data were discussed.
in Chapter IV. More detailed interaction experiments were performed using the NACA 0012 airfoil section with particular emphasis on such effects as the vortex geometric position, free stream Mach number, Reynolds number, and downstream airfoil angle of attack. The perpendicular vortex airfoil interaction study for the C-141 airfoil section included a vortex generator with a semi-span of 4.75 in., corresponding to a vortex height \((h/C)\) of 0.48 measured at the airfoil leading edge. The airfoil was set to a zero angle of attack and the vortex always passed over the instrumented surface of the airfoil (i.e., upper surface). On the other hand, the NACA 0012 airfoil was subject to non-dimensional vortex heights of 0.3, 0.2, and 0.1 as a result of vortex generator semi-spans of approximately 5.1, 5.3, and 5.5 inches, respectively.

The vortex generators were generally operated at pitch angles of 0, -4, and -8 degrees, in order to determine the vortex strength effects on the pressure distribution of the downstream airfoil. The experimental Mach and Reynolds number variations were similar to those of the airfoil calibration studies, such that a direct comparison between them would be possible. It was considered important to devise experiments in which only one parameter varied at a time. This was necessary in order to isolate the effects of individual variables.
Test Results

Vortex Airfoil Interaction Using C-141 Airfoil

The vortex airfoil interaction measurements were primarily in the form of pressure distribution changes in the upper surface of the airfoil. The vortex height effects were not studied during this phase of the investigation, however vortices of different strengths were simulated. Figure 6.2 illustrates the results of a typical interaction experiment where the data taken from the vortex-free flow are compared to the results of flows with vortex generator pitch angles of 0 and -8 degrees for a subcritical case of \( M = 0.73 \). The results of the above runs do not indicate a significant change in the pressure distribution of the airfoil as a result of the vortex, and the slight differences are well within the accuracy of the system, and perhaps the differences between the Mach numbers of the runs. Similar trends may be seen in Fig. 6.3 for a free stream Mach number of 0.75, representing a critical flow over the airfoil and a vortex generator angle of -8 degrees. For a free stream Mach number of 0.76, a slightly supercritical flow, however a rather stronger shock wave at \( X/C = 0.45 \) is evident. Although the shock induced separation for the vortex free flow is preserved, a region of what appears to be a secondary separation at \( X/C = 0.7 \) may be seen. Similar behavior is shown in Figs. 6.5 and 6.6 for test section Mach numbers of 0.83 and 0.85 respectively. Figure 6.5 indicates
that as a result of increasing the vortex generator angle from -4 to -8 degrees, a progressive increase in the strength of the shock wave is seen as a result of increasing the strength of the vortex.

In summary, the vortex airfoil interaction data using the C-141 airfoil section resulted in effectively no substantial change in the pressure distribution of the downstream airfoil for Mach numbers up to 0.75, while for supercritical flows the results indicated a somewhat stronger compression wave as a result of the vortex in such flows. The structure of the trailing vortex system at the trailing edge of the downstream airfoil, generated by this particular vortex generator was not studied. But considering the relatively long distance between the vortex and the airfoil such behavior is expected. A more detailed discussion of such behavior in perpendicular interaction problems will be given following the presentation of the interaction data for the NACA 0012 airfoil.

Vortex Airfoil Interaction Using NACA 0012 Airfoil

Similar to the C-141 interaction studies, the interaction results of the 0012 airfoil are presented in the form of pressure distribution changes from the vortex free cases. Although interaction studies for a range of Mach numbers (0.68-0.90) were conducted, the experimental data presented here in general consists of flows up to Mach numbers for which
the data are felt to be accurate (generally $M < 0.80$). The trailing vortices of various strengths are studied by operating the vortex generator at pitch angles of $0, -4, \text{ and } -8$ degrees, and the Reynolds number variations were similar to the C-141 data; in the range of 3.5-5.6 million based on the airfoil chord.

Typical pressure distributions for free stream Mach numbers of 0.71, 0.74, and 0.78 are illustrated in Figs. 6.7-6.9 for a vortex height of $h/C=0.3$ and vortex generator angles of -4 and -8 degrees. For the case of $M=0.71$, the pressure distribution does not seem to be substantially affected by the presence of the vortex. On the other hand, for test section Mach numbers in excess of 0.74, corresponding to critical and supercritical flows over the airfoil, a rather significant decrease in the magnitude of $C_p$ values for the leading 30 percent of the airfoil chord are evident. While beyond $X/C=0.3$, no substantial differences in comparison with the vortex free case are detected.

Similar results may be seen for a vortex height of 0.2, where a more pronounced decrease in $C_p$ measurements near the leading edge are observed. Figures 6.10-6.12 illustrate pressure distribution for Mach numbers of 0.71, 0.74, and 0.78 respectively. In comparison with the results of the previous case for $h/C=0.3$, the vortex effect seems to be shifted toward the airfoil leading edge, where a more pronounced effect of the vortex can be observed at $X/C=0.025$. 
The most significant change in the pressure distribution of the downstream airfoil was observed for the closest encounter for which the vortex height was 0.2 in. (h/C=0.1). These effects are shown in Figs. 6.13-6.15 for free stream Mach numbers of 0.70, 0.74, and 0.78, respectively. The vortex appears to only affect the leading 10 percent of the airfoil chord, downstream of which the pressure distribution resembles that of the vortex free flow. The erratic behavior in high supercritical flows caused by small surface irregularities of the airfoil which was discussed in Chapter IV was investigated for the present test configuration involving the vortex airfoil interaction tests with exactly similar results. Figure 6.16 illustrates typical data for a Mach number of 0.8 and vortex generator angles of 0, -4, and -8 degrees. For a vortex generator pitch angle of -8 degrees, the effect of the vortex near X/C=0.025 is so severe that the flow has been accelerated to nearly sonic conditions near the airfoil leading edge, while for vortex generator angles of 0 and -4 degrees the pressure distributions are similar to the vortex-free case.

Two of the important parameters determining the intensity of the vortex airfoil interaction problem are the geometric position of the trailing vortex with respect to the interacting airfoil (i.e., X_V and Z_V), and its strength. The earlier discussion concerning the spanwise deflection of the vortex in traversing the chord of the downstream airfoil, indicated that the spanwise deflection of the vortex core was
found to be directly proportional to the vortex height above the airfoil. Due to the spanwise deflection of the vortex the downstream airfoil experiences different loading along its chord. It is also obvious that under close encounters the strength of the vortex is not preserved in traveling from the leading to the trailing edge of the airfoil, and certain amount of energy which is responsible for alteration of the pressure distribution will be dissipated as a result of the interaction mechanism. The spanwise drift of the vortex core is crudely shown in Fig. 6.17 for three different vortex generator semi-spans. Since the exact trajectory of the vortex along the chord of the downstream airfoil is not known, a straight line connecting the vortex centers at the leading and trailing edges, as determined by the total pressure rake measurements, is drawn for simplicity. The substantial pressure distribution changes near the leading edge of the airfoil may be realized by considering the pressure gradient in the $z$ direction normal to the airfoil surface. This effect is caused by the pressure difference between the low pressure vortex core region and the higher pressure region near the airfoil leading edge (nearly stagnation pressure). For the closest encounter ($h/C=0.1$), both the pressure gradient effect and also the fact that the spanwise drift of the vortex which results in deflection of the core away from the position of pressure ports are partially responsible for such behavior (path C). For the vortex heights of 0.2 and 0.3, the pressure gradient is lower in comparison to the case of $h/C=0.1$, and
also the vortex drift is less than the previous case, causing the vortex to affect a longer portion of the airfoil chord (paths B and A). Also of vital importance in studying the interaction problem are the vortex breakdown and flow separation phenomena, which were experimentally observed by flow visualization techniques for similar interaction problems by several investigators (e.g. Ref. 40).

The vortex height effect may be clarified further by comparing the results of measurements for different vortex core heights under similar flow conditions. Such results are shown in Figs. 6.18-6.20, where pressure traces for a vortex generator angle of -8 degrees for free stream Mach numbers of 0.71, 0.74, and 0.78 are presented.

The data from interaction studies for the downstream airfoil angle of attack at 2 degrees confirm the earlier discussion for the airfoil angle of attack of zero, and the vortex effect is seen to be confined to the leading edge of the airfoil. Figures 6.21-6.22 are the results of interaction studies for Mach numbers of 0.74 and 0.76, respectively. The most pronounced effect may be seen in the pressure distribution of Fig. 6.22 where locally supersonic and separated flow region is seen in the leading 10 percent of the airfoil chord, as a result of the vortex airfoil interaction.

The combined effects of Mach number and vortex height on the aerodynamic characteristics of the downstream airfoil are
shown in Fig. 6.23, where the $C_p$ values are integrated along the airfoil chord to yield an average upper surface pressure coefficient, $C_p$. As the vortex height above the airfoil is reduced, a progressive decrease in the magnitude of the average pressure coefficient is evident. A $C_p$ decrease in excess of 30 percent for the closest encounter at a free stream Mach number of 0.71 is shown in Fig. 6.23, while this value is only slightly over 20 percent for a Mach number of 0.77 for the same vortex core height.

In the study of vortex airfoil interaction problem it is customary to indicate the vortex effect on the lift coefficient of the downstream airfoil. Since the NACA 0012 airfoil section used during this investigation was not equipped with the pressure ports on the lower surface, the average integrated pressure coefficient ($\overline{C_p}$) was used to simulate the effects of flow Mach number and vortex core height on the downstream airfoil.

In an attempt to obtain a single relationship between $\overline{C_p}$, $M$, and $h/C$, three correlation equations were considered as follows

$$\overline{C_p} = a_0(h/C)a_1(M) \frac{e^{a_2M}}{e^{a_3(h/C)}}$$  \hspace{1cm} (6.1)

$$\overline{C_p} = a_0 e^{a_1(h/C)} + a_2 e^{a_3M}$$  \hspace{1cm} (6.2)
\[ \bar{C}_p = a_0 \ln(a_1 \frac{h}{C}) + a_2 \ln(a_3 M) \]  \hspace{1cm} (6.3)

The method of least squares was applied to the above equations in order to determine the coefficients \( a_i \), such that the square of the differences between the experimental data and those of the above equations are minimized i.e.,

\[ \Delta \bar{C}_p = \left( \bar{C}_p^c - \bar{C}_p^e \right)^2 \] = minimum

where the subscripts \( c \) and \( e \) represent the calculated and experimental values respectively. This requires that:

\[ \frac{\partial \Delta \bar{C}_p}{\partial a_i} = 0 \] for \( i=1,2,\ldots \)

Simultaneous solution of the above equations results in calculation of the \( a_i \) coefficients. Among the above relations, equation 6.2 resulted in the best fit of the data (Table 6.1).

Equation 6.2 fits the experimental data with estimated errors of less than 10 percent with a somewhat better agreement for \( h/C \) of 0.1 and 0.2. Figure 6.24 indicates the comparison of the above data with the experimental data for a range of Mach numbers and vortex core heights.

Equation 6.2 results in curves which are very similar for all vortex core heights. In order to make the curves respond to various vortex core heights (h/C), an equation similar to
equation 6.2 was used in the following form

\[
\bar{C}_p = (a_0 e^{a_1 (h/C)} + a_2 e^{a_3 M}) (h/C)^4 \tag{6.4}
\]

Using an identical procedure to the one discussed earlier the coefficients \(a_1\) were determined. Figure 6.25 compares the results of equation 6.4 with the experimental data. Although this is an improvement over that of equation 6.2, the above relationship does not have the proper response for the Mach number variation. To remedy this deficiency an alternative form of equation 6.4 may be obtained by multiplying equation 6.4 by some polynomial function of \(M\) of the following form

\[
\bar{C}_p = (a_0 e^{a_1 (h/C)} + a_2 e^{a_3 M}) (h/C)^4 (b_0 + b_1 M + b_2 M^2) \tag{6.5}
\]

The coefficients for the above equation are listed in Table 6.1. The relationship given by equation 6.5 is shown in Fig. 6.26 to agree with the experimental data within 3 percent for \(h/C=0.3\) while the agreement is less than 2 percent for \(h/C\) of 0.1 and 0.2.

The difficulties in matching the pressure coefficient measurements of the NACA 0012 airfoil section at high free stream Mach numbers, which was discussed earlier in Chapter IV is further illustrated in the \(C_p\) plots of Figs. 6.23-6.26. The aforementioned plots indicate that the \(C_p\) distribution data deviates from those of Ref. 36 (as determined by integrating
the experimental data of Ref. 36) near Mach 0.78. This will limit the experimental data and empirical correlation to the present airfoil section only, and they will underestimate the lift force on the NACA 0012 airfoils at high Mach numbers.

An examination of the pressure coefficient data for vortex generator angles of 0 and -4 degrees indicate that in general no substantial change from the $C_p$ of the vortex free flow may be seen and the slight differences are well within the accuracy of the experiments.
Fig. 6.1 Simulation of perpendicular vortex airfoil interaction

Fig. 6.2 Vortex airfoil interaction using C-141 airfoil:

\[ \frac{h}{C} = 0.48, \ M = 0.73, \ Re = 5.1 \times 10^6 \]
Fig. 6.3 Vortex airfoil interaction using C-141 airfoil:

\[ h/C = 0.48, \ M = 0.75, \ Re = 5.2 \times 10^6 \]

Fig. 6.4 Vortex airfoil interaction using C-141 airfoil:

\[ h/C = 0.48, \ M = 0.76, \ Re = 5.3 \times 10^6 \]
Fig. 6.5 Vortex airfoil interaction using C-141 airfoil:

\[ \frac{h}{C} = 0.48, \ M = 0.83, \ \text{Re} = 3.9 \times 10^6 \]

Fig. 6.6 Vortex airfoil interaction using C-141 airfoil:

\[ \frac{h}{C} = 0.48, \ M = 0.85, \ \text{Re} = 5.5 \times 10^6 \]
Fig. 6.7 Vortex airfoil interaction using NACA 0012 airfoil:

\[ \frac{h}{c}=0.3, \ M=0.71, \ \text{Re}=5.1 \times 10^6 \]

Fig. 6.8 Vortex airfoil interaction using NACA 0012 airfoil:

\[ \frac{h}{c}=0.3, \ M=0.74, \ \text{Re}=5.2 \times 10^6 \]
Fig. 6.9 Vortex airfoil interaction using NACA 0012 airfoil:

\[ h/C = 0.3, \ M = 0.78, \ Re = 5.4 \times 10^6 \]

Fig. 6.10 Vortex airfoil interaction using NACA 0012 airfoil:

\[ h/C = 0.2, \ M = 0.71, \ Re = 5.1 \times 10^6 \]
Fig. 6.11 Vortex airfoil interaction using NACA 0012 airfoil:

\[ \frac{h}{C} = 0.2, \ M = 0.74, \ \text{Re} = 5.2 \times 10^6 \]

Fig. 6.12 Vortex airfoil interaction using NACA 0012 airfoil:

\[ \frac{h}{C} = 0.2, \ M = 0.78, \ \text{Re} = 5.5 \times 10^6 \]
Fig. 6.13 Vortex airfoil interaction using NACA 0012 airfoil:

\[ \frac{h}{C} = 0.1, \ M = 0.70, \ Re = 5.1 \times 10^6 \]

Fig. 6.14 Vortex airfoil interaction using NACA 0012 airfoil:

\[ \frac{h}{C} = 0.1, \ M = 0.74, \ Re = 5.2 \times 10^6 \]
Fig. 6.15 Vortex airfoil interaction using NACA 0012 airfoil:

\[ \text{h/C}=0.1, \text{M}=0.78, \text{Re}=5.4 \times 10^6 \]

Fig. 6.16 Vortex airfoil interaction using NACA 0012 airfoil:

\[ \text{h/C}=0.1, \text{M}=0.8, \text{Re}=5.6 \times 10^6 \]
Fig. 6.17 Illustration of spanwise drift of the vortex center

Fig. 6.18 Effect of vertical height of vortex core above airfoil surface: $M=0.71$, $Re=5.1 \times 10^6$
Fig. 6.19 Effect of vertical height of vortex core above airfoil surface: $M=0.74$, $Re=5.3 \times 10^6$

Fig. 6.20 Effect of vertical height of vortex core above airfoil surface: $M=0.78$, $Re=5.5 \times 10^6$
Fig. 6.21 Interaction for airfoil angle of attack of 2°:

\[ h/C=0.1, \, M=0.74, \, Re=5.3 \times 10^6 \]

Fig. 6.22 Interaction for airfoil angle of attack of 2°:

\[ h/C=0.1, \, M=0.76, \, Re=5.4 \times 10^6 \]
Fig. 6.23 Average pressure coefficient variation with Mach number for several vortex core heights

Fig. 6.24 Comparison of average pressure coefficient with equation 6.2 for different vortex heights
Fig. 6.25 Comparison of average pressure coefficient with equation 6.4 for different vortex heights

Fig. 6.26 Comparison of average pressure coefficient with equation 6.5 for different vortex heights
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CHAPTER VII

CONCLUSIONS AND RECOMMENDATIONS

Conclusions

One of the major drawbacks of the existing experimental data in transonic flows, especially in blade vortex interaction studies, is the complexity of the test configuration used. This limits their applicability for use in computational and theoretical schemes. Most of the available experimental and theoretical data in BVI investigations deal with the parallel interaction problem (Refs. 25 and 22). Since the perpendicular vortex airfoil interaction problem is also of practical importance, and due to limited availability of such experimental data in transonic flows, it is believed that the experimental results of the present study could be valuable for validating the results of computational fluid dynamics (CFD) codes.

The use of the Ludwieg tube wind tunnel facility for conducting large scale experimental studies proved to be promising for future investigations. It was found that the proper use of the secondary and primary flow rates through the system will result in a range of "true" transonic Mach and
Reynolds numbers seldom achieved by most available wind tunnels. It was also shown that by the proper use of the porous wall test section walls the wave reflection from the test section walls can be eliminated.

It was shown in the discussion of Chapter II, that it is possible to overcome the problem of relatively short steady state run time, by means of proper time delay settings and also by fine tuning of the operating pressures of the various valves. The quality of test section flow was shown to be fairly good by test section Mach number uniformity studies; where for a wide range of transonic flow conditions, excellent axial Mach number uniformity was measured in the test section. The repeatability of the UTA transonic wind tunnel facility was of great concern during the experimental studies, and it was verified by the centerline probe measurements and the airfoil calibration studies, for which the pressure distribution data of runs with similar operating conditions were shown to be repeatable to a high degree of accuracy.

The procedure developed during these studies for calibrating the test section Mach number against the Mach number in the plenum chamber proved to be a fairly reliable scheme. This may be verified by inspecting the NACA 0012 airfoil pressure distribution data, where for a given test section Mach number the agreement of the Ludwig-tube data with the published experimental data is good.
The airfoil calibration studies produced accurate results when compared to the experimental data taken from larger facilities with relatively less blockage effects. However, due to the small size of the test model, special care must be taken in design and fabrication of such models, since small surface irregularities are known to cause inaccurate measurements in transonic flows. The erratic pressure distribution behavior on the NACA 0012 airfoil section at high supercritical Mach numbers is thought to have been caused by the small wave like surface of the airfoil. Another critical problem caused by the model size restrictions is the long response time resulting from the use of small pressure tap sizes on the airfoil, which should also be seriously considered in the design of such models.

In the vortex structure studies, extremely useful qualitative results were obtained by conducting total pressure rake surveys of the vortex core. Such total pressure measurements indicate a well defined vortex core as well as the unsteady nature of the compressible vortex core. The rake pressure measurements proved to be instructive in determining the vortex geometric position relative to the downstream airfoil and when combined with the data obtained by five-hole cone pressure measurements, resulted in a fair approximation of the structure of trailing vortices. The spanwise displacement of the vortex after interacting with the airfoil, and possibility of the vortex break up into multiple vortices
were among the findings determined by the rake measurements.

The vortex airfoil interaction studies, indicated the significant change in the pressure distribution on the downstream airfoil near the leading 20-25 percent of the airfoil chord. The vortex strength and its geometric position were found to strongly influence the pressure distribution of the interacting airfoil.

Recommendations

Although, attempts were made to obtain the best possible experimental data with the available instrumentation, further improvements may be achieved by feasible modification of either operational procedures or by using a more accurate data acquisition system. The following recommendations are given:

1. Reduction of tunnel starting time may be achieved by using a diaphragm start instead of sliding sleeve valve. The use of such procedure has been used in the past (Ref. 30), but due to non-repeatable diaphragm rupture, the use of such scheme was abandoned in favor of the sliding sleeve valve.

2. Use of a 12 bit data system instead of the 8 bit system of the present investigations will significantly improve the accuracy of the experimental data.

3. If pressure transducers of smaller volume are used, the response time of the pressure measurements will be improved. Also by reducing the pressure differential between the initial
and final stages, and by minimizing the tubing length, longer steady state measurements are attainable. Positioning the small pressure transducers inside the plenum chamber will be a possible solution for such requirements.

4. In order to take full advantage of the five-hole cone probe pressure measurements, a complete calibration of such probe for a range of pitch, yaw, and roll angles is required. By combining the results of the present studies with the calibration data, a complete velocity field of the vortex core can be obtained.

5. The use of flow visualization techniques will result in more accurate measurements, and use of such techniques will be both helpful in understanding the physics of the interaction problem and also clarify some of the uncertainties of the present investigations. Such measurements will be particularly important in the vortex structure studies and in the vortex airfoil interaction experiments in close encounters where the possibility of vortex break-up was discussed in Chapter V.

6. Use of fast response pressure transducers on the surface of the airfoils will eliminate the problem of pressure system time lags, resulting in longer steady state run times.

7. Direct measurements of lift and drag of the vortex generator could be used to express the vortex strength in terms of its vorticity rather than its angle of attack which
was used in presenting the results of these studies.

The importance of the present vortex airfoil interaction studies has already been discussed earlier. The results of these investigations have proved to be sufficiently accurate for low transonic Mach numbers and Reynolds numbers of interest, that makes the results an ideal data base for computer program assessment.
APPENDIX A

COMPUTER PROGRAM FOR DATA REDUCTION
***************AIRFOIL.FOR***************

* THE AIRFOIL.FOR PROGRAM IS A DATA REDUCTION PROGRAM USED
* TO CALCULATE THE PRESSURE COEFFICIENTS OF THE AIRFOIL
* USING THE EXPERIMENTAL PRESSURE DATA.

* CPDVM.DAT --- WHICH IS A CALIBRATION DATA FILE WHERE THE
* D.V.M. READING (MV) IS STORED IN THIS FILE
* UNDER THE FORMAT F10.4.
* CSDVM.DAT --- WHICH IS A CALIBRATION DATA FILE WHERE THE
* HEX NUMBER READING FROM COMPUTER IS STORED
* IN THIS FILE UNDER THE FORMAT AS DESIGN.
* (THE FIRST LINE MUST BE BLANK)
* INPUT.DAT --- WHICH IS THE EXPERIMENT DATA CAPTURED BY
* THE COMPUTER.
* STAGN.DAT --- WHICH IS THE CALIBRATION DATA FILE OF
* STAGNATION PRESSURE THAT D.V.M. (MV) AND
* PRESSURE (PSIG) STORED IN THIS FILE UNDER
* THE FORMAT 2F10.2.

* DEFINITION OF VARIABLES

* VOLT(I) --- THE ITH VOLTAGE CALIBRATION DATA FOR
* STAGNATION PRESSURE (#10).
* SP(I) --- THE ITH PRESSURE CALIBRATION DATA FOR
* STAGNATION PRESSURE (#10).
* CP(I,J) --- THE JTH CALIBRATION DATA OF PRESSURE FOR
* THE ITH CHANNEL.
* CS(I,J) --- THE JTH CALIBRATION DATA OF SIGNAL FOR
* THE ITH CHANNEL.
* P(I) --- PRESSURE OF THE ITH CHANNEL.
* S(I) --- SIGNAL OF THE ITH CHANNEL.
* MACH(I) --- MACH NUMBER OF THE ITH CHANNEL.
* CPP(I) ------ PRESSURE COEFFICIENT
* T ------ TEMPERATURE

**************************
* MAIN PROGRAM *
**************************

IMPLICIT REAL*8(A-H,O-Z)
IMPLICIT BYTE(A)
REAL*8 MACH
DIMENSION CS(32,15),CP(32,15),SP(32),PS(32),PNEW(32),NUM(32),CPP(32)
DIMENSION A(80),B(16),INIT(16)
DIMENSION VOLT(110),SP(110)
DIMENSION MACH(20)

WRITE(5,560)
READ(5,830)PAIM
WRITE(5,700)
READ(5,710)TO
WRITE(5,912)
READ(5,913)CFACT
WRITE(5,914)
READ(5,915)TTEMP
CALL OPEN(1,'INPUT DAT',0)
CALL OPEN(2, 'FLOAT DAT', 0)

READ DATA FROM FILE INPUT.DAT

WRITE(5, 500)
WRITE(5, 510)
READ(1, 520) A
READ(1, 520, END=30) A
WRITE(5, 520) A

CONVERT DATA FROM HEX DECIMAL INTO FLOATING POINT

DO 20 I=1,16
J=(I-1)*3+9
CALL CODE(A, J, K)
IF(K .GT. 20) GO TO 900
J=J+1
CALL CODE(A, J, L)
IF(L .GT. 20) GO TO 900
INIT(I)=16*K+L
B(I)=INIT(I)
CONTINUE

WRITE FLOATING POINT DATA INTO FILE FLOAT.DAT

WRITE(2, 530) B
GO TO 10

READ CALIBRATION DATA FROM FILE CLIBR.DAT

CONTINUE

CALL OPEN(3, 'CPDV M DAT', 0)
CALL OPEN(7, 'STAGN DAT', 0)

READ D.V.M. READING FROM FILE CPDV M.DAT
AND STAGNATION PRESSURE CALIBRATION DATA
FROM FILE STAGN.DAT.

L=1
CONTINUE
READ(7, 790, END=50) VOLT(L), SP(L)
L=L+1
GO TO 40

L=L-1

ICP=0
CONTINUE
READ(3, 540, END=80) DVM
CALL STAGP(VOLT, SP, L, DVM, PRESU)
PRESU=PRESU+PATM
ICP=ICP+1
DO 70 I=1,32
CP(I, ICP)=PRESU
CONTINUE
GO TO 60
READ CALIBRATION DATA FROM FILE CSDVM.DAT

CONTINUE
WRITE(5, 820)
ENDFILE 3
CALL OPEN(4, 'CSDVM DAT', 0)
ID1=1
ID2=2
INDEX=ID1
IA=0
READ(4, 550) A
READ(4, 550, END=150) A
WRITE(5, 550) A
DO 100 I=1, 16
   J=(I-1)*3+9
   CALL CODE(A, J, K)
   IF(K .GT. 20) GO TO 910
   J=J+1
   CALL CODE(A, J, L)
   IF(L .GT. 20) GO TO 910
   INIT(I)=15*K+L
   B(I, J)=INIT(I)
100 CONTINUE
   J=(INDEX-1)*16+1
   K=J+15
IF(INDEX .EQ. ID2) GO TO 110
IA=IA+1
110 CONTINUE
   KA=1
   DO 120 KK=J, K
      CS(KK, IA)=B(KA)
   KA=KA+1
120 CONTINUE
GO TO (130, 140), INDEX
130 INDEX=ID2
GO TO 90
140 INDEX=ID1
GO TO 90
150 CONTINUE
IF(IA .NE. ICP) GO TO 890

CHECK LINEAR DEVIATION OF CALIBRATION

WRITE(5, 570)
DO 220 I=1, 132
   IF(I.EQ.13 OR I.GT.16) GO TO 220
   NUM(I)=0
   DO 170 J1=1, IA
      IF(CS(I, J1) .GE. 254.0 OR CS(I, J1) .LE. 1.0) GO TO 170
      NUM(I)=NUM(I)+1
      NI=NUM(I)
      CS(I, NI)=CS(I, J1)
      CP(I, NI)=CP(I, J1)
170 CONTINUE
   IREST=IA-NUM(I)
   IF(IREST .GT. 0) GO TO 180
   END
GO TO 200
CONTINUE
INUM=INUM(I)+1
LD 190 J2=INUM.IA
CS(I,J2)=0.
CP(I,J2)=3.
CONTINUE

C
CONTINUE
CMAX=0.
CSUM=0.
WRITE(S,20191)
IN=INUM(I)
DO 210 J=1,IN
CALL CURVE(CS,CP,IN,1,CS(I,J),CPNEW)
DD=(CP(I,J)-CPNEW)/CPNEW*100.0
DDP=DABS(CS)
CSUM=CSUM+DDP*DDP
RR=DABS(CMAX)
IF(RR.LT.DDP)CMAX=DD
WRITE(S,800)CP(I,J),CS(I,J),DD
CONTINUE

CMAX=(CSUM/IA)**0.5
ITT=INUM(I)
SPAN=1.0/(CS(I,ITT)-CS(I,1))*100.0
PMIN=SPAN*(CP(I,ITT)-CP(I,1))/100.0
WRITE(S,590)CMAX,PMAX,PMIN

C
******************************************************************************
C CALCULATE THE EXPERIMENT PRESSURE DATA
******************************************************************************
C
REWIND 2
CALL OPEN(6,'PRESS DAT',0)
WRITE(S,600)
INDEX=ID1
IC=0
WRITE(S,610)

C
TP=TP+0.0
CONTINUE
IF(INDEX.EQ.ID2)GO TO 240
INDEX=INDEX+1
IF(INDEX.EQ.ID2)GO TO 240

READ(2,620,END=300)(S(L),L=J,K)

C
ORDER THE ARREY P(M)

IF(INDEX.EQ.ID2)GO TO 241
MP=0

DO 260 M=J,K
IF(M.EQ.13.OR.M.GT.16)GO TO 260

INTERPOLATE EXPERIMENT DATAS

MP=MP+1
CALL CURVE(CS,CP,NUM(M),M,S(M),P(M))
PNEW(mp)=P(mp)
260 CONTINUE
ICHAL=15
C
C PRINT OUT THE RESULTS
C
IF(INDEX.EQ.ID1) GO TO 270
WRITE(6,630) (PNEW(I), I=1,ICHAL)
WRITE(5,640) TP , (PNEW(I), I=1,ICHAL)
270 CONTINUE
GO TO (280,290), INDEX
280 INDEX=ID2
GO TO 230
290 INDEX=ID1
GO TO 230
C
C ************************************************************
C PLOT THE PRESSURE DISTRIBUTION
C ************************************************************
C
300 CONTINUE
C WRITE(5,650)
C READ(5,660) INDEX
INDEX=2
GO TO (310,360), INDEX
310 CONTINUE
REWIND 6
WRITE(5,670)
READ(5,680)IP
PMAX=0.
PMIN=1000.
320 CONTINUE
READ(6,690,END=330) (PNEW(I), I=1,ICHAL)
IF(PMAX.LT. PNEW(IP)) PMAX=PNEW(IP)
IF(PMIN.GT. PNEW(IP)) PMIN=PNEW(IP)
GO TO 320
C
330 CONTINUE
REWIND 6
I=0
RANGE=PMAX-PMIN
DT=1.0
340 CONTINUE
READ(6,720,END=350) (PNEW(L), L=1,ICHAL)
I=I+1
TIME=T0+(I-1)*DT
CALL PLOT(TIME,PNEW,PMIN,RANGE,IP)
GO TO 340
350 CONTINUE
WRITE(5,730)
WRITE(5,740)
READ(5,750)IDP
GO TO (310,360),IDP
C
C **********************************************
C CALCULATE MACH NUMBER
C **********************************************
C
360 CONTINUE
C

REWIND b
C

RGAS=1.40
EXP=(RGAS-1.0)/RGAS
JCAL=JCAL-1
WRITE(5,760)
TM=0
370 CONTINUE
TM=TM+1.0
READ(a,770, END=990) (PNEW(I), I=1,ICHAL)
C

CHECK STAGNATION PRESSURE
C

DO 360 I=1,ICHAL
IF(PNEW(I) .GT. PNEW(S)) GO TO 370
360 CONTINUE
DO 390 I=1,ICHAL
PR=PNEW(S)/PNEW(I)
MACH(I)=2/(RGAS-1.)*PR**EXP-1.0)**0.5
390 CONTINUE
IF(MACH(I).GT.5.) MACH(I)=0.
WRITE(5,780)TM,(MACH(I), I=1,ICHAL)
TSECTM=MACH(I)*CFACT
EXPP=RGAS/(RGAS-1.)
PINF=PNEW(S)/(1.+(RGAS-1.)/2.*TSECTM**2)**EXPP
C

CALCULATE PRESSURE COEFFICIENT
C

DO 391 I=1,ICHAL
CPP(I)=2./RGAS/TSECTM**2*(PNEW(I)/PINF-1.)
391 CONTINUE
C

CALCULATE VISCOSITY AND REYNOLDS NUMBER
C

WRITE(5,780)TM,(CPP(I), I=1,ICHAL)
IF(TM.LT.9.0) GO TO 370
ITEMP1=ITEMP+459.67
ITEMP2=ITEMP/1.6
ITEMP3=ITEMP/0.9
TSTAT=ITEMP3/1.+((RGAS-1.)/2.*TSECTM**2)
SSOUND=(RGAS**2.569*TSTAT)**0.5
VISC=1.4508-6.*(TSTAT**1.5)/(TSTAT+110.4)
VEL=TSECTM*SSOUND
P04=PINF*101325.0/14.696
RHO=RGAS*P04/SSOUND**2
REYNLD=RHO*VEL**0.0508/VISC
WRITE(5,91b) REYNLD
GO TO 370
C

500 FORMAT(/,2X,'**************',/,1,2X,'** INPUT DATA **',/2,2X,'**************',///)
510 FORMAT(15X,'CAUGHT DATA IN FILE INPUT.DAT',///)
520 FORMAT(80A1)
530 FORMAT(16F4.0)
540 FORMAT(F10.4)
550 FORMAT(80A1)
560 FORMAT(1X,*** CALIBRATION DATA DOES NOT MATCH ***',///)
570 FORMAT(/,3X,'LINEAR DEVIATION CHECK FOR EACH CALIBRATION DATA',/1,3X,'*2 ------ AIRFOIL',/3X,'*3 ------ AIRFOIL',/3X.)
2. /3X, ' #--------AIRFOIL',/3X,'*5--------AIRFOIL',/3X,
3*8. =---- AIRFOIL',/3X, '*7 ...... AIRFOIL',/3X,
4*5 .. ------ STATIC',/3X, 'Ov------TOTAL',/3X, 'O10------AIRFOIL',
5/3X, 'O11 ---- AIRFOIL/3X, 'O13 ---- AIRFOIL',/3X,
6*8. =AIRFOIL',/3X, 'O16 ---- AIRFOIL',/3X,
7*9. ---- AIRFOIL',/3X, 'O18 ---- AIRFOIL',/3X,
8*10. =AIRFOIL',/3X, 'O20 ---- AIRFOIL',/3X,
9*31 ---- PLENUM CAVITY',/)

570 FORMAT(1X, '**** INPUT THE ATMOSPHERE PRESSURE (PSIA) ***')
579 FORMAT(5X, 'RMS ERROR = ', F10.2, 1X, ', %', 1X,
3      ' MIN. UNIT P = ', F10.3, 1X, 'PSIA', /)
600 FORMAT(1X, '# OUTPUT DATA #', 1X,
1      'EXPERIMENT PRESSURE DATA', 1X, 'TIME (MS)',
2      'STATIC', 'STAGNATION', 'AIRFOIL', 'PLENUM CAVITY', 1X,
5    'S11/17', 'S2', '6/12/16', /)
620 FORMAT(16F4.0)
630 FORMAT(26F8.2)
640 FORMAT(26F8.2, 1X, '3F8.2')
650 FORMAT(1X, 'DO YOU WANT PLOT PRESSURE DISTRIBUTION ?', 1X,
1      'IF YES, THE TYPE 1. IF NO, THEN TYPE 2')
660 FORMAT(11)
670 FORMAT(1X, 'WHICH CHANNEL DO YOU WANT TO PLOT ?', 1X,
1      'ENTER BY NUMBER 1 TO 18', 1X,
2      ' PLENUM CAVITY ------ 18', 1X,
3      ' STATIC ---------------- 8', 1X,
4      ' STAGNATION ------------ 9', 1X,
5      ' AIRFOIL --------- 1-7,10-17', 1X,
680 FORMAT(11)
690 FORMAT(26F8.2, 1X, '3F8.2')
700 FORMAT(1X, 'WHAT IS THE TIME DELAY BETWEEN SSV-OPEN AND', 1X,
1      'STARTING DAT (MS) ?', 1X,
710 FORMAT(10.2)
720 FORMAT(26F8.2, 1X, '3F8.2')
730 FORMAT(1X, 'VERTICAL AXIS ------ TIME (MS) ', 1X,
1      'HORIZONTAL AXIS ------ PRESSURE (PSIA) ', 1X,
2      'DO YOU NEED PLOT SOME OTHER CHANNEL ?', 1X,
3      'IF YES, THEN TYPE 1. IF NO, THEN TYPE 2')
750 FORMAT(11)
760 FORMAT(1X, 'EXPERIMENT MACH NUMBER', 1X,
1      'TIME (MS)', 'STATIC', 'STAGNATION', 'AIRFOIL', 'PLENUM CAVITY', 1X,
2      'S11/17', 'S2', '6/12/16', 1X,
770 FORMAT(26F8.2, 1X, '3F8.2')
780 FORMAT(26F8.2, 1X, '3F8.2')
790 FORMAT(1X, 'CHANNEL', 'PRESSURE (PSIA)', 'SIGNAL(MV)', 'ERROR(%)', 1X,
800 FORMAT(1X, 'CALIBRATION DATA FROM FILE CSDVM.DAT', 1X,
830 FORMAT(1X, 'CALIBRATION DATA FROM FILE CSDVM.DAT', 1X,
860 FORMAT(10.4)
870 FORMAT(12)
880 FORMAT(' ', 'INPUT MACH NO CORRECTION FACTOR')
890 FORMAT(' ', 'INPUT TOTAL TEMPERATURE IN F ----')
910 FORMAT(F5.1)
916 FORMAT(10X, '*RE = ', E11.3)
930 WRITE(5, 560)
STOP
STOP OFF
END

********************
* SUBROUTINE *
********************

THIS SUBROUTINE IS WRITTEN FOR CONVERTING HEX DECIMAL NUMBER INTO FLOATING POINT NUMBER.

********************
SUBROUTINE CODE(A,J,I)
IMPLICIT BYTE(A)
DIMENSION A(80)

I=A(J)-48
IF(I .GT. 15) I=I-7
IF(I .GT. 15) I=100
IF(I .LT. 0) I=100
RETURN
END

******************** CURVE ********************
THIS SUBROUTINE IS FOR LINEAR SQUARE CURVE FITTING TO GET THE UNKNOWN P
DEFINE :
X = DATA POINT (INPUT)
Y = DATA POINT (INPUT)
N = NUMBER OF DATA POINTS
M = CHANNEL INDEX
S = SIGNAL DATA (INPUT)
P = PRESSURE DATA (OUTPUT)

********************
SUBROUTINE CURVE(X,Y,N,M,S,P)
IMPLICIT REAL*8(A-H,O-Z)
DIMENSION X(32,15),Y(32,15)

SUMX=0.
SUMXSQ=0.
SUMY=0.
SUMXY=0.

DO 10 I=1,N
SUMX=SUMX+X(M,I)
SUMXSQ=SUMXSQ+X(M,I)*X(M,I)
SUMY=SUMY+Y(M,I)
SUMXY=SUMXY+X(M,I)*Y(M,I)
10 CONTINUE

C1=SUMY*SUMXSQ-SUMXY*SUMX
C2=N*SUMXY-SUMX*SUMY
C3=N*SUMSQ-SUMX*SUMX
A0=C1/C3
A1=C2/C3
P=A0+A1*S
RETURN
END

***************************************************************************
**PLUT**************************************************************************
**
**THIS PROGRAM IS WRITTEN FOR PLOTTING TIME - STAGNATION
**PRESSURE RELATION.
**
***************************************************************************

SUBROUTINE PLUT(TIME,PNEW,PXMIN, RANGE, IP)
IMPLICIT REAL*8(A-H,O-Z)
DIMENSION FMAT(30)
DIMENSION PNEW(12)

M=(PNEW(IP)-PMIN)*50./RANGE+0.5
ENCOD(FMAT,1000000000)
FORMAT(16I1,2L6.2,1X,I12,11HX,'*'),F6.2)
WRITE(FMAT,TIME,PNEW(IP))
RETURN
END

***************************************************************************
**STAGP**************************************************************************
**
**SUBROUTINE STAGP IS WRITTEN FOR LINEAR LEAST SQUARE CURVE FITTING.
**
***************************************************************************

DEFINE :
X = DATA POINT (KNOWN)
Y = DATA POINT (KNOWN)
P = INTERPOLATED POINT (KNOWN)
S = INTERPOLATED POINT (UNKNOWN)
N = NUMBER OF DATA POINTS

***************************************************************************

SUBROUTINE STAGP(X,Y,N,S,P)
IMPLICIT REAL*8(A-H,O-Z)
DIMENSION X(110),Y(110)

SUMX=0.
SUMSQ=0.
SUMY=0.
SUMAY=0.

DO 10 I=1,N
SUMA=SUMX+X(I)
SUMSQ=SUMSQ+X(I)*X(I)
SUMY=SUMY+Y(I)
SUMAY=SUMAY+X(I)*Y(I)
10 CONTINUE
C1=SUMY*SUMSQ-SUMXY*SUMX
C2=N*SUMXY-SUMX*SUMY
C3=N*SUMSQ-SUMX*SUMX
AU=C1/C3
A1=C2/C3
P=AG+A1*S
RETURN
END
APPENDIX B

ERROR ANALYSIS
Error Analysis

This section contains the calculations of the errors produced by the transducers, amplifiers and analog to digital converters. These analyses are followed by an uncertainty study, by introducing the system total error into the equations used to determine the Mach numbers and $C_p$ calculations for the flow over the airfoil.

The Kulite ITQS-500F-500SG pressure transducers used during the experimental investigation, according to the manufacturers specifications, have a combined error of 0.5 percent of the full scale (linearity and hysteresis). The variable gain amplifiers were used to amplify the output of the pressure transducers to match the requirements of a specific run condition. For example, for Mach numbers of interest (0.6-1.2) and for airfoil chord Reynolds numbers of 3-6 million, it can be shown that the airfoil pressure readings are in the range of 30-80 psi. To obtain the most accurate measurements possible using the Kulite pressure transducers, the output from the variable gain amplifiers was adjusted prior to the experiment so that the entire range of
the amplifier output was used for the pressure range of the interest for a specific run, rather than the recommended maximum attainable pressure. If this procedure is used properly, the overall system error, as will be shown in the following sections, will be substantially less than the values suggested by the transducer specifications. On the other hand, without using the variable gain amplifiers, relative errors in excess of 2.5 psi will result, which can easily be shown to be totally unacceptable for the Mach numbers of interest in the present study. In fact, results of several runs using Kulite 100 psi transducers, (maximum error of 0.5 psi), indicated results identical to the data obtained for the 500 psi transducers used in the course of the present investigation under similar operating conditions.

The combined error for typical pressure transducer can be obtained using

\[ E_T = \frac{(0.005)(F.S)}{P} \]

where \( E_T \) is the combined transducer error (linearity and hysteresis), F.S represents the full scale range of the output (typically about 70-90 psi) and \( P \) is the pressure.

The Kulite KH-102 amplifiers used to amplify the output of each individual pressure transducer, according to the manufacturers specifications, have a total error of 0.1
percent of the full scale, thus

\[ E_A = \frac{(0.001)\text{(F.S)}}{p} \]

The total error introduced by the A-D converters, was also a significant factor in the overall error of the system. For the 8 Bit Xerox 820 microcomputer, which was used in the present study, the overall error can be determined from the following equation

\[ E_{AD} = \frac{\text{F.S}}{2^\text{(n-1)}} \frac{1}{p} \]

with \( n \) being 8 for an 8 Bit converter system.

Adding the above equations, the overall system accuracy may be calculated. The stagnation pressure measurements were however subject to a relatively less error since the full scale was adjusted to approximately 70 psi as opposed to about 100 psi used for the static pressure measurements. The results of such calculations based on a typical operating condition are shown in Fig. A.1, where the system overall error as a function of pressure is presented.

**Uncertainty Analysis**

The governing equation for determining the experimental Mach number based on an isentropic flow assumption is
\[ M = \left\{ \frac{2}{\gamma-1} \left[ \frac{P_t}{P} (\gamma-1)/\gamma - 1 \right] \right\}^{1/2} \]

and partial derivatives of \( M \) with respect to \( P \) and \( P_t \) are given by:

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

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

While the pressure coefficients for the airfoil are obtained by using the following equation

\[
C_p = \frac{2}{\gamma M^2_{\infty}} (P/P_{\infty} - 1)
\]

or

\[
C_p = \frac{2}{\gamma M^2_{\infty}} \left\{ \frac{1+(\gamma-1)/2 M^2_{\infty}}{\gamma/(\gamma-1)} \right\} - 1
\]

and the derivatives of \( C_p \) with respect to \( M \), and \( M_{\infty} \) are given by the following equations, respectively
\[
\frac{\partial C_p}{\partial M_\infty} = \frac{-4}{\gamma M_\infty^3} \left\{ \frac{1+(\gamma-1)/2 M_\infty^2}{1+(\gamma-1)/2 M^2} \right\} \left[ \frac{1}{\gamma/(\gamma-1)} \right]
\]

\[
+ \frac{2}{M_\infty} \left[ \frac{1+(\gamma-1)/2 M_\infty^2}{1+(\gamma-1)/2 M^2} \right]^{1/(\gamma-1)}
\]

\[
\frac{\partial C_p}{\partial M} = \frac{-2 M}{M_\infty^2} \left[ \frac{1+(\gamma-1)/2 M_\infty^2}{1+(\gamma-1)/2 M^2} \right]^{2(\gamma-1)/(\gamma-1)}
\]

The uncertainties may then be calculated from the following equations (Ref. 41):

\[
W_M = \left\{ \left( \frac{\partial M}{\partial P} W_P \right)^2 + \left( \frac{\partial M}{\partial P_t} W_{P_t} \right)^2 \right\}^{1/2}
\]

\[
W_{Cp} = \left\{ \left( \frac{\partial C_p}{\partial M_\infty} W_{M_\infty} \right)^2 + \left( \frac{\partial C_p}{\partial M} W_M \right)^2 \right\}^{1/2}
\]

Where

- \( W_M \) uncertainty in Mach number
- \( W_P \) uncertainty in pressure
- \( W_{P_t} \) uncertainty in total pressure
- \( W_{Cp} \) uncertainty in pressure coefficient
- \( W_{M_\infty} \) uncertainty in free stream Mach number

The results of these calculations are shown in Fig. A.2 for both Mach number and pressure coefficient calculations.
using the experimental pressures for a range of model pressures. Figure A.2 is obtained by assuming a free stream Mach number of 0.75 and a total pressure of 100 psi which is representative of runs for which Re=5.5 million. Typical pressures on the airfoil surface vary from 100 psi at the leading edge stagnation point to about 40 psi corresponding to a local Mach number of slightly greater than 1.2. In the aforementioned pressure range Fig. A.2 indicates a Mach number uncertainty of about 0.01 while the pressure coefficient uncertainties vary from 0.06 at high local Mach numbers to about 0.03 near the airfoil leading edge stagnation point. Although the uncertainties for both Mach number and pressure coefficient calculations described above seem to be slightly high, the experimental data described in Chapter IV indicated a good agreement with the data of Ref. 36.
Fig. A.1 Combined error for experimental pressure measurements

Fig. A.2 Uncertainty in experimental Mach number and pressure coefficient
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