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**An Investigation on the Effects of the High Maximum-
Thickness-to-Chord Ratio on The Performance of Nozzle Guide
Vanes in a Transonic Planar Cascade.**

by

Rama Radmard
B. Eng (Mechanical Engineering)

A Thesis submitted to
the Faculty of Graduate Studies and
Research in partial fulfilment of the
requirements for the degree of

Master of Engineering
(Aerospace Engineering)

**OTTAWA-CARLETON INSTITUTE FOR MECHANICAL
AND AEROSPACE ENGINEERING**

**DEPARTMENT OF MECHANICAL AND AEROSPACE
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CARLETON UNIVERSITY

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Dedication

I dedicate this thesis to my loving parents, whom without their support and belief in me, I could not have finished this work.

"Thank you"

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Abstract

The performance of turbine airfoils is usually predicted by empirical correlations. However, these correlations are inadequate to predict the performance of airfoils with maximum-thickness-to-chord ratio of higher than 25%.

To create a data base from which the performance of turbine airfoils with the maximum-thickness-to-chord ratio of higher than 25% could be predicted, a collaborative work between Pratt & Whitney Canada, National Research Council and Carleton University was initiated.

A planar cascade consisting of four airfoils was constructed to allow the investigation of the effect of the maximum-thickness-to-chord ratio on the airfoil performance. Three airfoil sets with maximum-thickness-to-chord ratio of 15.2% (baseline), 26.6% (27% airfoil) and 48.2% (48% airfoil) were used. Measurements included surface Mach number distributions for the baseline airfoil, total pressure loss coefficients and deviation angles for isentropic exit Mach numbers of 0.7 (design), 0.9 and 1.1. Also the effect of varying the inlet boundary layer thickness and free-stream turbulence level was investigated. The results showed that the 27% airfoil produced lower losses as predicted by the Kacker and Okapuu correlation (1982). The introduction of turbulence produced a significant redistribution of the losses in the exit plane. The secondary loss decreased as the leading edge diameter was increased. Except

for the baseline blade where high under-turning in exit flow angle was observed, the 27% and 48% airfoils showed a decrease in over-turning with increasing exit Mach number, as predicted by Ainley and Mathieson (1951).

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Nomenclature

A_p	inlet area per passage, $H * S$
c	airfoil chord length
c_x	airfoil axial chord length
C_p	static pressure coefficient, $(P - P_{o1}) / (P_{o1} - P_{s2,ave})$
C_{po}	total pressure loss coefficient, $(P_{o1} - P_{o2}) / (P_{o1} - P_{s2,ave})$
H	airfoil height
i	incidence angle, $\alpha_1 - \beta_1$
$i_{(des)}$	design incidence angle, $\alpha_{(des)} - \beta_1$
$i_{(eff)}$	effective incidence angle, $\alpha_{(eff)} - \beta_1$
k	ratio of specific heats
m	mass flow rate
M	Mach number
P	pressure
Q	dynamic pressure, $P_o - P_s$
R_o	Reynolds number

S	pitch length
t	airfoil thickness
t_e	trailing edge diameter
T	temperature
Tu	turbulence intensity
U	velocity
V	velocity
x	axial direction
y	pitchwise direction
Y	total pressure loss coefficient, $(P_{o1} - P_{o2}) / (P_{o2} - P_{s2ave})$
z	spanwise direction
α	flow angle
β	metal angle
β_e	stagger angle, angle between the chord and the axial direction
γ	ratio of specific heats, c_p/c_v
δ	boundary layer thickness
δ	deviation angle, $\beta_2 - \alpha_2$
δ_1	displacement thickness
θ	momentum deficit thickness
ζ	kinetic energy coefficient, $1 - (V_2^2 / V_{2is}^2)$
ϕ^2	kinetic energy coefficient, $1 - (V_2^2 / V_{2is}^2)$
ρ	density

Subscripts

a	exit flow condition
atm	atmospheric
ave	average
ex	exit plane condition
in	inlet condition
is	isentropic
max	maximum
ms	midspan
opt	optimal
p	profile
s	secondary
s	static, when used with pressure
wp	wedge probe
◦	total condition
1	inlet condition
2	airfoil exit condition

Superscripts

= passage averaged , $\frac{1}{A_p} \int_0^S \int_0^H C_{p_0} dx dz$

— pitch averaged, $\frac{1}{S} \int_0^S C_{p_0} dx$

Chapter 1

Introduction

The design of turbomachines and in particular turbine airfoils is governed by the need to satisfy certain aerodynamic and stress/dynamic requirements. The requirement for higher cycle efficiency has driven the turbine inlet temperature (TIT) to the point that it has exceeded the allowable limits imposed by the material for acceptable level of strength and durability. Traditionally, this problem has been dealt with by providing internal and/or film cooling. But these methods are only suitable for large scale engines where manufacturing internal passages and film cooling holes are possible using the present state of technology. However, for smaller engines, such as those manufactured by Pratt & Whitney Canada, the above mentioned options become less suitable due to the increased manufacturing difficulties.

A possible solution to cope with this problem is to use thicker airfoils,

where the increased thickness and leading edge diameter will facilitate the manufacturing of internal passages and the drilling of shower head cooling holes. Therefore, as a first step in designing turbine airfoil with higher maximum-thickness-to-chord ratios, it is essential to be able to predict the performance.

Traditionally, there have been several methods to predict the performance of turbine airfoils, as summarized by Sieverding (1985). For example, one of the well known methods is due to Ainley and Mathieson (1951). Their correlation, which is expressed as follows:

$$Y_p = \left\{ Y_{p(\beta_1=0)} + \left(\frac{\beta_1}{\alpha_2} \right)^2 [Y_{p(\beta_1=\alpha_2)} - Y_{p(\beta_1=0)}] \right\} \left(\frac{t_{max}/c}{0.2} \right)^{\frac{\beta_1}{\alpha_2}}$$

(where $Y_{p(\beta_1=0)}$ (Figure 1.1) and $Y_{p(\beta_1=\alpha_2)}$ (Figure 1.2) are the profile loss coefficient for nozzle and impulse airfoil rows respectively, having the same pitch-to-chord ratio (S/c) as the actual airfoil), showed that the profile loss is a function of inlet metal angle, exit flow angle, pitch-to-chord ratio and maximum-thickness-to-chord ratio.

However, this method is only applicable for a limited range of maximum-thickness-to-chord ratio. In fact, the method is deemed reliable only for $15\% \leq t_{max}/c \leq 25\%$.

Figure 1.1: Profile loss coefficients for nozzle airfoil at zero incidence, reproduced from Ainley and Mathieson (1951).

Figure 1.2: Profile loss coefficients for impulse airfoil at aero incidence, reproduced from Ainley and Mathieson (1951)

In 1982, Kacker and Okapuu proposed a modification to the improved Ainley and Mathieson method (Dunham and Came (1970)). They introduced factors to compensate for the compressibility effects and recent improvements in turbine airfoil design. Their correlation, which is summarized as follows:

$$Y_p = 0.914 \left(\frac{2}{3} Y_{p,AMDC} K_p + Y_{SHOCK} \right)$$

(where $Y_{p,AMDC}$ is the profile loss due to Ainley, Mathieson, Dunham and Came, K_p is the correction for the effects of exit Mach number and channel acceleration, and Y_{SHOCK} is due to the local flow acceleration adjacent to the highly curved leading edges), produced better agreement with experimental results, but, it was limited to the same range of maximum-thickness-to-chord ratio as the Ainley and Mathieson method. However, as it will be shown later, this method did predict the performance of thicker airfoils with reasonable accuracy.

There are also other methods proposed by Balje and Binsley (1968), Craig and Cox (1970), Denton (1973) and Traupel (1977) which follow different approaches. However, these methods are based on correlations derived from the data produced by conventional turbine airfoils, which have t_{max}/c ratios comparable to those used by Ainley and Mathieson. Therefore, these methods may also be deemed unreliable for t_{max}/c ratios of higher than 25%.

This lack of adequate design data for airfoils with t_{max}/c of higher than 25% results in a need for further research in this area.

The thesis describes a collaborative experimental study, between Pratt & Whitney Canada (P&WC), National Research Council (NRC) and Carleton University, in a planar cascade test section in which turbine airfoils with t_{max}/c ratios of 15, 27 and 48% were tested. The test measurements include total pressure loss, deviation angle, and airfoil surface Mach number distribution; the latter measurement was performed only for the baseline airfoil ($t_{max}/c = 0.152$). The measurements were taken at exit Mach numbers of 0.7 (design), 0.9 and 1.1. The Reynolds number was based on inlet flow conditions and the axial chord and ranged from 2.8×10^5 to 3.2×10^5 .

The 27% airfoil had the lowest total pressure loss coefficient among the profiles tested. The losses for the 27% and 48% airfoils were in agreement with the Kacker & Okapuu (1982) correlation. Finally, conclusions and recommendations for further work conclude this report.

Chapter 2

Literature review

2.1 Airfoil and loss terminology

The losses in a turbine airfoil row can be divided into different components. The individual components are assumed to be independent of each other and only a function of airfoil geometry and inlet/outlet gas conditions. The loss components are categorized as follows:

1. **Profile loss;** loss due to the skin friction on the suction and pressure sides of the airfoil which is exposed to purely two-dimensional flow conditions.
2. **Trailing edge loss;** loss due to base pressure at the trailing edge, caused by the finite trailing edge thickness. The profile and trailing edge losses are usually grouped together since they conveniently

provide the so called midspan loss, which is easily measured experimentally.

3. **Secondary loss;** loss due to the interaction of the endwall boundary layer and the airfoil leading edge.
4. **Tip clearance loss;** loss due to the leakage of fluid between the blade tip and the endwall.
5. **Annulus loss;** loss due to the skin friction on the casing at the hub and tip of the blade row.

Except for the annulus loss, all other loss components are strong functions of airfoil geometry. Among more important factors are inlet gas angle, inlet and outlet metal angles, throat opening, trailing edge diameter and maximum-thickness-to-chord ratio.

The present study is mainly concerned with the effect of the maximum-thickness-to-chord ratio on the losses of the first and third type with lesser emphasis on the secondary loss. The following section presents the basic airfoil terminology used throughout this thesis followed by a description of previous works which have direct or indirect relevance to this study.

The convention used for the study of losses in turbine airfoils is summarized in Figure 2.1. The losses are commonly expressed in term of total pressure loss coefficient Y , which is defined as difference between the inlet and exit total pressures normalized by exit dynamic pressure, or in term of kinetic

energy coefficient ζ^* , which is a measure of kinetic energy deficit between the actual and isentropic flow. These two coefficients are defined as follows:

$$\gamma = \frac{P_{02} - P_{01}}{P_{02} - P_{s_{ave}}}$$

$$\zeta = 1 - \frac{V_2^2}{V_{2is}^2}$$

the two may be related using the following relation:

$$\gamma = \frac{\left[1 - \frac{\gamma-1}{2} M_2^2 \left(\frac{1}{1-\zeta} - 1\right)\right]^{\frac{\gamma}{\gamma-1}} - 1}{1 - \left(1 + \frac{\gamma-1}{2} M_2^2\right)^{\frac{\gamma}{\gamma-1}}}$$

* kinetic loss coefficient is also symbolized by ϕ^* .

Figure 2.1: Turbine blading terminology, reproduced from Moustapha (1988)

2.2 Survey results

On the effect of high maximum-thickness-to-chord ratio on the losses, an extensive literature search has highlighted the need for further research. Specifically, there has been little work done on investigating the effects of maximum-thickness-to-chord ratio on the losses for airfoils with t_{max}/c of higher than 25% in the transonic flow regime. There were only three previous works found that had indirectly touched on this topic.

In 1979, Graham and Kost presented work on shock boundary layer interaction in a high turning transonic turbine cascade. They studied two different airfoils with equal turning. The airfoils designated profile RA and profile RB had maximum-thickness-to-chord ratios of about 37% and 53% respectively.

Their cascade wind tunnel was of the suction type with 9 to 15 airfoils depending on the inlet angle (the test section could take up to 20 airfoils). The Reynolds number based on the chord length and exit flow conditions was 8.5×10^5 and the turbulence level of the inlet flow was of the order of 1 percent. The airfoil profiles and cascade geometry are shown in Figure 2.2 and Figure 2.3 respectively. The cascade geometry is presented because it has been found that the test setup can effect the results (see Kiock et al. (1985)).

They showed that one of the important factors governing the performance is the curvature of the airfoil suction side, since it controls the appearance of the separation bubble. The separation bubble and its location can affect the subsonic loss coefficient.

Their results are shown in terms of pressure loss coefficient, Y_p , and exit flow angles α_2 in Figure 2.4 and Figure 2.5 respectively. The pressure loss coefficient shows an increase in losses from the 37% airfoil to the 53% airfoil. The magnitude of the losses fall within the same range as those measured in this project.

DESIGN GAS CONDITIONS:

Inlet Angle = 60°
 Exit Angle = 68°
 Exit Mach Number = 1.15

Figure 2.2: Cascade profile RA and RB, reproduced from Graham & Kost (1979)

Figure 2.3: Sketch of the test section showing the cascade geometry, reproduced from Graham & Kost (1979).

Figure 2.4: Variation of profile loss coefficient, Y_p , with isentropic exit Mach number, reproduced from Graham & Kost (1979).

Figure 2.5: Variation of exit angle, β_2 , with exit Mach number, reproduced from Graham & Kost (1979).

In 1986, Baines et al. (1986) reported on the aerodynamic development of a highly loaded nozzle guide vane. Among the airfoils they studied, two airfoils designated T8 and T12 had a maximum-thickness-to-chord ratio of 26.8% and 21% respectively. The exit flow angle and the total pressure loss coefficients were investigated over a range of exit Mach number from 0.3 to 1.4. The airfoil geometry is presented in Figure 2.6. The main cascade parameters reported are as follows:

profile	S/c	β_1	α_2	R_{α_2}	Tu
T8	0.842	44.85	65.0	5×10^5	4%
T12	0.946	49.95	65.0	5×10^5	4%

The pitch and span were kept constant between the two airfoils at 67.2 mm and 300 mm respectively, therefore, the aspect ratio (span / chord) varied from 3.75 for T8 to 4.24 for T12. Both cascades contained seven airfoils and in both cases, two central airfoils were equipped with approximately 50 static pressure taps.

The result presented in Figure 2.10 showed that the T8 airfoil provided better performance than the T12 airfoil despite the fact that it was thicker. Furthermore the T12 has consistently higher losses. The increasing losses at lower Mach number is consistent with a boundary layer growth caused by the diffusion in the airfoil suction side, but it is interesting that this diffusion trend was not predicted by the analytical scheme used. The trend experienced

between the T8 and T12 airfoil is consistent with the trend observed in the 27% and baseline airfoil. Their result has shown a need for instrumenting all airfoil sets with static pressure taps (due to unreliability of analytical prediction schemes to provide reasonable surface Mach number distribution at high t_{max}/c) and providing endwall windows for obtaining Schlieren photographs, which provide invaluable insight in shock boundary layer interaction, which in transonic ranges is of utmost importance.

Recently, Sieverding (1990) reported a detailed study on a transonic turbine cascade with high turning and maximum-thickness-to-chord ratio.

The airfoil used had a t_{max}/c of $\approx 27\%$ and turning of 80° . The total pressure loss coefficient and the exit flow angles were investigated over a range of exit Mach numbers from 0.7 to 1.4.

The airfoil and cascade geometry are presented in Figure 2.8. The cascade geometry was of a suction type with 9 airfoils (8 passages). The main cascade characteristics are summarized below:

- pitch-to-chord ratio $S/c = 0.78$
- stagger angle $\gamma = 58.8^\circ$
- gauging angle $\arccos(o/S) = 80.5^\circ$
- aspect ratio $h/c = 1.25$
- chord length $c = 40.07 \text{ mm}$
- inlet turbulence $Tu = 1\%$

Figure 2.6: Comparison of airfoil profiles, scaled to the same pitch, reproduced from Baines et al.(1986).

Figure 2.7: Total pressure loss coefficient and exit flow angles as a function of isotropic exit Mach number, reproduced from Baines et al. (1986).

where o and h are the throat opening and airfoil height or span respectively.

The results in terms of variation of total pressure loss coefficient and exit flow angles with isentropic exit Mach numbers are presented in Figure 2.12.

The result obtained by Sieverding (1990), corresponds extremely well with results obtained in this project for the 27% airfoil with 0.4% free stream turbulence.

2.3 Conclusion

The literature survey has highlighted the lack of direct study in the effect of the maximum-thickness-to-chord ratio on the performance of turbine blades. Furthermore, the literature survey has shown that it seems possible to design turbine blades having maximum-thickness-to-chord ratio of higher than the traditional ($t_{max}/c > 25\%$) with a performance comparable to blades having maximum-thickness-to-chord ratio of less than 25%.

Figure 2.8: The airfoil and cascade geometry, reproduced from Sieverding (1990)

Figure 2.9: Variation of total pressure loss coefficient and exit flow angle with isentropic exit Mach number, reproduced from Sieverding (1990).

Chapter 3

Experimental equipment

3.1 Introduction

The Pratt & Whitney Canada Planar Cascade was designed to allow the measurement of the relative performance of three different airfoil profiles in identical flow conditions.

The cascade, (Figure 3.1), was operated in suction mode with air being drawn from the laboratory. This provided a low turbulence level ($Tu \approx 0.4\%$), and also allowed the use of the NRC National Exhauster Facility, a 2680 hp (2 MW) unit capable of exhausting 10 lb/s of air at 2 inches of mercury absolute.

The data acquisition and control system was computer operated using menu driven software which allowed the automation of experimental data

acquisition. The automation covered the traversing and nulling of the probe, control of multiport pressure transducers for measurement of static pressures, acquisition of data from total and static pressure transducers, data acquisition from thermocouples, preliminary data reduction to provide operating point conditions and generation of standardised documented data files. The software was designed to be capable of performing several different data acquisition scenarios. It must be noted that the original software was written and previously used by NRC staff, (Carscallen, 1990). However, it required minor modification in order to allow airfoil surface static pressure and downstream midspan total pressure distributions to be obtained.

3.2 Design of planar cascade

3.2.1 General design

Figure 3.2 presents the schematics of the planar cascade test section. The cascade constructed for this experimental study was of the suction type and consisted of four airfoils and four passages. To achieve four passages with only four airfoils, bleed plates were shaped as the airfoil camber line, which was believed to simulate the streamline at mid passage (mean-streamline), and placed adjacent to the first and fourth airfoils. The centre nozzle passage was the passage under study. The transonic turbine stator airfoils were scaled to 4.2 times engine size. The general design parameters are summarized in Table 3.1. The profiles were based on midspan profile of actual turbine blades.

Adjustable boundary layer bleeds were installed on both sidewalls adjacent to the first and fourth airfoil to remove side wall boundary layers and developing corner vortices. An adjustable porous tailboard was positioned downstream, continuing at the nominal angle of the bleed plate adjacent to the first airfoil (see Figure 3.2). The porous tailboard was used to provide further means for adjusting the flow and achieving flow periodicity. Also, the tailboard by the nature of its construction eliminated the shock wave reflection at transonic exit Mach numbers (Malek , 1986).

3.2.2 Working section

The working section was designed such that the air would be drawn inward through the inlet section and into the working section. The air was turned through 67° by the airfoils and exited into the horizontal transition piece which connected the cascade's rectangular exit to the circular intake of the exhaust inlet system. The transition piece was covered by an insulating material to reduce the noise created by the shock waves reflecting inside the transition piece.

The cascade endwalls were made of 0.625 inches thick aluminum sheets to limit the inward deflection of the endwalls to less than 0.0001 inches under the maximum of one atmosphere of pressure difference. There was a ± 0.03 inches of internal surface waviness, attributed to the manufacturing process of aluminum and the thermal stresses caused by the welding process used

Inlet Temperature	530 °R	(293 °K)
Inlet Total Pressure	14.7 psia	(101.3 KPa)
Inlet Mach Number	0.175 (choked)**	
Exit Mach Number	0.7 to 1.1	
Mass Flow Rate	= 8.5 lb/s (choked)	
Pressure Ratio***	1.38 to 2.14	
Scale Factor	4.189 : 1	
Span	4.440 in	(112.8 mm)
Pitch	4.455 in	(113.2 mm)
Nozzle Throat Opening	1.474 in	(37.4 mm)
Nozzle Turning Angle	67 deg	
Axial Chord Length	3.895 in	(98.9 mm)

Airfoil Maximum-thickness-to-chord ratio

Designation	Actual t_{max}/c
The Baseline Airfoil	15.2%
The 27% Airfoil	26.6%
The 48% Airfoil	48.2%

Table 3.1: PWC/NRC planar cascade general design summary

** Choked refers to the condition where the flow Mach number has reached 1.0 at the blade throat.

*** Pressure ratio is based on inlet total and exit static pressures (total to static).

1) Left hand endwall

2) From right to left; passages 1,2 (centre) and 3

Figure 3.1: Photograph of PWC/NRC Planar Cascade Test Section.

- 1) Intake bellmouth
- 2) Upstream traverse slot
- 3) Downstream traverse slot
- 4) Tailboard
- 5) Upstream bleed valve and plate
- 6) Downstream bleed valve and plate
- 7) Cascade intake
- 8) Cascade exit
- 9) Exhauster intake
- 10) Downstream sidewall
- 11) Upstream sidewall

Figure 3.2: Schematics of PWC/NRC Planar Cascade.

to connect the flanges to the endwalls.

The cascade, as mentioned before, consisted of four airfoils and four passages. In order to achieve four passages with only four airfoils, sheet metal plates curved to the mean-streamline shape were placed at half pitch before and after the first and the fourth airfoil, respectively. The bleed valve assemblies were located between these plates and the sidewalls (see Figure 3.3). The boundary layer bleed intake opening allowed a maximum of a 0.5 inch thick boundary layer to be bled off.

An adjustable, porous surface, tailboard, Figure 3.1, was located just downstream of the mean-streamline-plate and continued at the nominal streamwise direction for 19 inches. This provided another mechanism to complement the boundary layer bleeds in achieving flow periodicity in the cascade. The porous surface was 0.188 inches thick, with 23% porosity. The holes were 0.188 inches in diameter and 0.375 inches apart (centre to centre), and the hole aspect ratio was 1.0.

The airfoils for the planar cascade were machined from stainless steel, using a numerically controlled electronic discharge machine. The coordinates and the machining of the airfoils were provided by Pratt & Whitney Canada. The three airfoil profiles referred to as the baseline, 27% and 48% airfoil had t_{max}/c of 15.2%, 26.6% and 48.2% respectively. All airfoils had equal exit flow angle, throat opening and axial chord (Table 3.1). The airfoil profiles (same

scale) are summarized in Figure 3.4.

3.2.3 Probe traversing locations

Two traversing slots, 14.4 inches in length and 0.375 inches wide were incorporated upstream and downstream of the working section on the left hand (see Figure 3.1) endwall. The upstream slot was located at 51% of axial chord from the airfoil leading edge, and the downstream slot was at 26% of axial chord downstream from the airfoil trailing edge. Both traversing slots are shown in Figure 3.5.

3.2.4 Static pressure tap locations

The two baseline airfoils bounding the centre passage were instrumented with static pressure taps located at midspan. There were 12 pressure taps on the pressure side and 15 on the suction side. The locations of pressure taps are shown in Figure 3.6.

A total of 67 pressure taps were located at 26% of axial chord downstream of the trailing edge in the traversing plane. The spacing was such as to allow at least 13 pressure taps per passage (0.343 inches centre-to-centre) for passages 1 and 3, and 26 pressure taps (0.172 inches centre-to-centre) for the centre passage, where the actual traverse was performed (see Figure 3.7).

Figure 3.3: Photograph of the bleed valve assemblies.

Figure 3.4: Sketch of the baseline, 27% and 48% airfoils.

Figure 3.5: Planar cascade upstream and downstream traversing slots.

Suction Side	Pressure Side
X location	X location
0.078	0.078
0.195	0.195
0.390	0.390
0.779	0.779
1.169	1.169
1.558	1.558
1.948	1.948
2.143	
2.338	2.338
2.532	
2.727	2.727
3.117	3.117
3.506	3.506
3.701	3.701
3.924	---

● Pressure tap locations

Figure 3.6: Surface pressure tap locations for the baseline airfoil

Figure 3.7: Upstream and downstream static pressure tap locations.

There were three pressure taps located upstream over passages number 1, 2 and 3 to provide the inlet static pressure distributions.

A pitot tube was placed upstream, outside the boundary layer, to provide the inlet dynamic head. Also the pitot tube was used as the reference point for all except barometric pressure transducer. All static taps were 0.04 inches in diameter.

3.3 Measurement probes

Three types of probes were used in this project. A total head tube constructed of 0.0625 inch stainless steel tubing was mounted on the sidewall in the mainstream flow to measure the inlet total pressure. This pressure combined with the upstream static pressure was used to evaluate the inlet dynamic pressure (Q_1). A custom designed total head probe was used for all inlet velocity profile measurements. Finally, a 0.1875 inch diameter United Sensor wedge probe, model WT187, was used to measure the downstream total pressure, flow angle and total temperature, as shown in Figure 3.8. The wedge angle was 30 degrees.

Calibration of the wedge probe was carried out by NRC staff in the NRC National Aeronautical Establishment's 5 × 5 inches high speed pilot tunnel. Complete details, results and discussion of the calibration reproduced from Carscallen (1990) are included in Appendix A.

Figure 3.8: Schematic of the wedge probe.

3.4 Pressure Measuring Equipment

3.4.1 Introduction

The schematic of pressure plumbing and pressure transducer arrangements for the planar cascade is given in Figure 3.9.

All pressures except for the barometric pressure, were referenced to the total pressure measured via the pitot tube located on the downstream sidewall (see Figure 3.2), upstream of the airfoil leading edge. Pressures were measured with either a multiport Scanivalve system or stand-alone pressure transducers. All pressure transducers could be checked for zero offset by pneumatically short circuiting the transducers. This was carried out automatically by the data acquisition system at the beginning of each test run. All pressure transducers were calibrated using a Texas Instrument Inc. Precision Pressure Test Set, model 156-02-0166, which itself had been calibrated by the Institute for National Measurement Standards (INMS). It must be noted that the transducers were calibrated every six months by NRC staff and the calibration constants in the data acquisition software were updated accordingly.

Table 3.2 summarizes the type and range of transducers used in this project.

Function	Make	Serial No.	Range (Imp)	Range (SI)
Scanivalve Transducers				
No. 1	Statham	61160	± 10.0 psig	± 68.9 KPa
No. 2	Statham	61161	± 10.0 psig	± 68.9 KPa
No. 3	Statham	59269	± 10.0 psig	± 68.9 KPa
No. 4	Statham	61134	± 10.0 psig	± 68.9 KPa
Stand-Alone Transducers				
Pitot tube	Statham	392	± 0.2 psig	± 1.37 KPa
Operating point	Statham	14205	± 0.15 psig	± 1.03 KPa
Wedge p. total	Statham	59270	± 10.0 psig	± 68.9 KPa
Wedge p. static	Kulite	4074-2-12	± 10.0 psig	± 68.9 KPa
Wedge diff. stat	Kulite	4074-2-10	± 10.0 psig	± 68.9 KPa
Barometric	Viatran	11427168	15.25 ± 2.25	105.2 ± 15.6
			psia	KPa abs.

Table 3.2: Planar cascade pressure transducer specifications.

PLUMBING DIAGRAM

Figure 3.9: Planar cascade pressure plumbing diagram.

3.4.2 Scanivalve system

The Scanivalve system, used to select and measure the endwall and airfoil surface static pressure taps, consisted of type 'S' model 48SGM pressure scanner incorporating 4 scanners with integral pressure transducers. Each scanner was capable of measuring 48 separate inputs. The unit had an integral solenoid controller to step or home the scanner and a position encoder. Also, a Scanivalve Solenoid Controller, model CTRL2/S2-S6 was used to manually control the Scanivalves and also to interface the Scanivalve system to the data acquisition and control computer. The position of the pressure scanner was indicated on a Scanivalve Odd-Even Decoder, model CTR2/OETM-BCD.

3.4.3 Probe pressure transducers

The measurement of pressure by the wedge probe , total head probe and the operating point probe was carried out by stand-alone pressure transducers. The reason was that the varying pressure ranges made it impractical to use a single transducer with a Scanivalve system.

3.5 Probe Traverse Gear

Probe movement in the spanwise, pitchwise and yaw direction was carried out by an L.C. Smith traverse system. The traverse system was comprised of the following components; a model **BBM 7.5 180** Actuator which allowed up to 7.5 inches of spanwise travel and 180 degrees of yaw movement and a model **L3MB** Linear Actuator allowing up to 20 inches of pitchwise movement. The **BBM 7.5 180** was mounted on the **L3MB**. The fin's assembly was mounted over the slot being surveyed. A photograph of the L.C. Smith traverse system is given in Figure 3.10.

The L.C. Smith actuators were controlled by an L.C. Smith Digital Indicator, model **D1-3R** and a model **D1-3R-3SB** Switch Box Control. The digital indicator controlled the motor drive selection (i.e. pitchwise, spanwise and yaw), direction selection (left/right, in/out, CW/CCW), speed selection and indicated the location of the traverse gear.

The digital indicator, through a modification by NRC staff, allowed external computer control of the above mentioned functions. Computer interfacing was carried out through low voltage and current mechanical relays.

Figure 3.10: Photograph of the L.C. Smith traverse gear assembly.

3.6 Data Acquisition and Control Systems

A Hewlett Packard HP3054C Data Acquisition and Control System was used for acquisition of data, control of Scanivalves and the L.C. Smith traverse gear system.

This system was comprised of an HP1000 A600+ minicomputer with 1.5 megabytes of memory, an HP7908 16 megabyte hard disc drive, a 64 megabyte streaming tape cartridge drive, an HP 9121 dual 3.5 inches floppy disc drive with 0.25 megabyte capacity each, an HP 2631B printer, an HP 7475A plotter, an HP3456A digital voltameter and an HP3497 data acquisition and control unit. The HP1000 A600+ minicomputer operated under RTE-A

VC + version 24.40. All computer codes were written in Fortran 77.

The data acquisition and control software was designed by Carscallen (1990) to be a user friendly, menu driven package with acquisition and storage of experimental data fully automated. The automation covered the traversing and nulling of the wedge probe, control of the multiport pressure transducers, acquisition of data from total and static pressure taps and pressure transducers, acquisition of data from thermocouples and preliminary data reduction to provide operating point conditions. The automation also covered the generation of standardised and documented output data files. The software was designed to allow several different data acquisition scenarios. These included: the downstream flow periodicity check (based on total pressure loss), two-dimensional downstream total pressure and total temperature traverse, downstream static pressure distribution, two-dimensional upstream total pressure distribution and airfoil surface static pressure distributions.

A block diagram of the data acquisition system is given in Figure 3.11. Figure 3.12 and Figure 3.13 show the software flow diagrams for the data acquisition and control software and the wedge probe nulling software.

Figure 3.11: Data acquisition system block diagram, reproduced from Carscallen (1990).

Figure 3.12: Data acquisition and control software flow diagram, reproduced from Carscallen (1990).

Figure 3.13: Wedge probe nulling software flow diagram, reproduced from Carscallen (1990).

Chapter 4

Commissioning of Planar Cascade

4.1 Introduction

This chapter is concerned with the shakedown tests carried out to determine the aerodynamic performance of the Pratt & Whitney Canada Planar Cascade. These included investigating the flow quality on the bellmouth surface, blade and endwalls surfaces and mean-streamline plates, also addressed is the cascade flow two-dimensionality. To accomplish this, direct flow field measurements were taken. Flow visualization of the limiting streamlines were used where direct flow field measurements were not possible.

4.2 Intake Bellmouth

The bellmouth was designed and constructed by Pratt & Whitney Canada

from fibreglass with internal dimensions of 18.82 inches long, 4.44 inches wide and 6.00 inches high. The bellmouth was designed to provide a clean attached flow with minimum boundary layer thickness. Flow visualization was used to investigate the flow quality over the bellmouth surface. The flow visualization techniques used in this project were the oil-dot method for the bellmouth surface and the oil-film method for the endwall and airfoil surfaces (see Merzkirch, 1987). Surface flow visualization on the bellmouth surface showed a separation on the upstream half of the bellmouth followed by reattachment, as shown in Figure 4.1. This implied an increase in the boundary layer thickness in this region. To quantify this increase and to survey the inlet flow uniformity, a spanwise traverse was carried out at five pitchwise stations, centred on the midpitch of the centre passage and with a distance of half a pitch apart. The result of the inlet flow survey is shown in Figure 4.2. The survey did not show a noticeable increase in the boundary layer thickness due to the separation and confirmed the thin boundary layer assumption.

4.3 Airfoil surface flow visualization

The airfoil surface flow visualization was used to investigate the state of the boundary layer flow on the airfoil surfaces. All airfoil surface flow visualizations were performed at the exit Mach number of 0.7. In all cases investigated, a separation bubble was apparent on the suction side for all three airfoil profiles. Surprisingly, as shown in Figure 4.3, there was also a separation bubble on the pressure side of the 48% airfoil.

a Separation bubble

Figure 4.1: Photograph of the bellmouth flow visualization

Figure 4.2: Spanwise traverse of the inlet velocity profile at five pitchwise stations.

a Separation bubble on the pressure side

Figure 4.3: Flow visualization on the pressure side of the 48% airfoil at exit Mach number of 0.7.

As presented in Figure 4.4, the airfoil surface flow visualization on the suction side of the baseline airfoil highlighted the presence of a separation bubble. It is believed that this separation was triggered by surface discontinuities caused by the manufacturing process. The presence of these discontinuities became evident after the flow surveys were performed for all airfoils. As a result all airfoils were reworked to remove the surface discontinuities and the experiment was performed over again. However, due to budget constraints the test cases with $M_\infty = 0.9$ and 1.1 were omitted. A summary of all configurations tested in this project will be presented in Chapter 5.

4.4 Mean-Streamline-Plate Effectiveness

The mean-streamline-plates were originally designed with the philosophy that they would simulate the flow at mid passage. The endwall flow visualization showed the generation of secondary flows by the mean-streamline-plate leading knife edge. Furthermore, the migration pattern of the pressure leg of the horseshoe vortex, see Langston (1990), from the pressure side to the suction side was different than that observed in the centre passage. However, flow visualization provides a clear picture of the "limiting streamlines" at the endwall and does not necessarily present the flow along the span. Therefore, in order to fully investigate the effectiveness of the mean-streamline-plates in simulating the mid passage flow, direct flow field measurements are required. A direct flow field measurement of the flow over

the mean-streamline-plates was not obtained due to inadequate length of the traverse slot on the cascade endwall.

The use of a knife edged leading edge does raise the possibility of flow separation at any incidence other than zero which could affect the performance of the mean-streamline-plates and consequently the bleed valve assembly as well. Therefore, the use of a knife edged leading edge should be avoided unless the insensitivity of the incidence angle to the bleed valve setting is certain.

4.5 Flow two-dimensionality

4.5.1 Introduction

The ability to separate each loss component from the passage total pressure loss coefficient is greatly dependent on the ability to measure or deduce the purely two-dimensional loss. Therefore, it is essential to be able to produce a two-dimensional flow, even if this region is limited to the partial length of the full span. Conceptually, two-dimensional flow could be achieved with infinite airfoil span. However, physically this is not a viable solution. Therefore, a criterion has to be established to specify the tolerable level of secondary flow effects. The most satisfactory test for two-dimensionality is an examination of the momentum balance for the airfoil boundary layers along both surfaces of the airfoil. However, this would require detailed boundary

b Location of surface discontinuity

Figure 4.4: Flow visualization on the suction side of the baseline airfoil (# 2) at exit Mach number of 0.7.

layer profiles which are rarely available. An alternative method was suggested by Rodger et al. (1992). They suggested that if the average velocity density ratio (AVDR) is close to unity and the spanwise distribution of the pitchwise averaged total pressure loss coefficient shows a region of constant loss around midspan, supported by suction surface flow visualization, then it could be concluded that the midspan flow is fairly two-dimensional. They also showed that a mild flow convergence ($AVDR > 1$), which is normally present in cascade measurements, does not lead to large error in total pressure loss coefficient. In this project, due to the uncertainty in yaw angle measurements it was not possible to obtain values for average velocity density ratio with enough accuracy to be of any use. However, flow visualizations of the suction surface and the spanwise distribution of the pitchwise averaged total pressure loss coefficients were obtained for all three airfoils at exit Mach number of 0.7. The results for the baseline airfoil are presented in Figure 4.5 and Figure 4.6. The results show a minimal streamtube convergence on the airfoil suction side and furthermore, the existence of a region of constant pressure loss about the midspan.

4.5.2 Conclusion

Although, as mentioned earlier, the examination of momentum balance is the most satisfactory test for flow two-dimensionality, the flow visualizations combined with the spanwise distribution of the total pressure loss coefficients have provided supporting evidence of two-dimensional flow at midspan.

$$t_{\max}/c = 0.15$$

Figure 4.5: Spanwise total pressure loss distribution for the baseline airfoil.

Figure 4.6: Flow visualization on the suction side of the baseline airfoil at exit Mach number of 0.7.

4.6 Effect of error in deviation angle on mass-averaged quantities

As shown by Carscallen (1990) (see Appendix B), the expected uncertainty in measuring exit flow angle was about ± 0.6 degrees at the pressure ratio of 2.3. This suggested that a discrepancy between the exit and inlet mass flow rates was to be expected. Since it was believed that error in exit mass flow rate directly affects the mass-averaging, the existence of a discrepancy between inlet and exit mass flow rates cast a shadow on the validity of mass-averaged quantities. In order to correct this problem a yaw offset was introduced. This yaw offset was a constant value which was either added or subtracted from the yaw angles until the inlet and exit mass flow rates were equal. It was observed that depending on the type of airfoil used and the exit Mach number, this yaw offset ranged from ± 0.5 to ± 10 degrees. This range of yaw offset proved to be quite unacceptable since it strongly affected the deviation distribution. Therefore, an analytical study was carried out to investigate the effect of errors in the yaw angle on the mass-averaged total pressure loss coefficient. The result showed that the total pressure loss coefficient was extremely insensitive to error in the deviation angle (i.e., less than $\pm 2\%$ at ± 10 Degrees of yaw offset). Furthermore, the degree of sensitivity was a function of the exit Mach number and the free-stream turbulence level. The result for the baseline airfoil is summarized in Figure 4.7. As a result of this study it was decided that no yaw offset would be used in processing the data, therefore, all deviation angles presented will be those measured directly at the exit plane.

Figure 4.7: Effect of the yaw offset on the total passage pressure loss coefficient (baseline airfoil).

Chapter 5

Experimental Procedure

5.1 Measurement of Pressure Ratio

This chapter describes the experimental procedure utilized in the measurement of gas stream parameters.

The pressure ratio, defined as inlet total pressure over average exit static pressure ($PR = P_{o1} / P_{s,2ave}$), was based on an area averaging of 26 out of 67 static pressure taps located downstream of the centre nozzle passage and the inlet total pressure. The isentropic exit Mach number is based on the pressure ratio obtained above. It is termed "isentropic", because the inlet total

pressure rather than the exit total pressure has been used in calculating the Mach number. Since the inlet total pressure does not account for any losses in the nozzle passage it assumes that the flow has gone through an isentropic process. It is necessary to use the inlet total pressure because at the start of a given experimental run the exit total pressure is unknown, therefore, the isentropic exit Mach number will provide a nominal exit Mach number which is repeatable for all airfoil sets used in the project.

Since the presence of the wedge probe affected the local downstream static pressure distribution and thus influenced the pressure ratio, to obtain repeatable results, as suggested by Carscallen (1990), the pressure ratio was set with the wedge probe in the fully retracted position.

Setting the pressure ratio was an iterative procedure with the calculation of the pressure ratio carried out automatically by the data acquisition system. The inlet pressure to the Brown Boveri exhaustor was adjusted until the desired pressure ratio was achieved. The pressure was then held constant for the duration of the test. The drift in the pressure ratio was less than 2% in the normal 3 hours duration of a test.

5.2 Measurement of downstream static pressure

In order to calculate any flow parameter from the data taken from the wedge probe (i.e., total pressure, total temperature and local Mach number) it

is necessary to know the local static pressure. Using a wedge probe it should be possible to measure the actual local static pressure directly from the wedge probe static pressure taps, using the appropriate calibration factors. However, Carscallen (1990), showed that the wedge probe used in this project provided erroneous results as the Mach number of the approaching gas stream exceeded 0.4. An alternative approach was suggested (Carscallen, 1990) to obtain the downstream static pressure distribution. Due to the low inlet boundary layer thickness it was assumed that the flow through the nozzles is nearly two-dimensional, which led to low curvature of the streamlines in the spanwise direction. Therefore, it was assumed that the static pressure gradient in this direction was negligible. The assumption of flow two-dimensionality was later confirmed with the aid of surface flow visualizations on the airfoil suction and pressure side (e.g. see Chapter 4). Therefore it was decided to use the static pressures measured on the endwall in the pitchwise direction and assume no change in the static pressure along the span.

The data from the 67 static pressure taps located on the endwall in the traversing plane were taken at the start of a traverse with the wedge probe at the fully retracted position.

5.3 Achieving periodicity

A four airfoil cascade is not periodic (Langston, 1990). The details of the flow entering and exiting each passage can vary depending on the proximity

of the passage to the cascade sidewalls. The best one can achieve in such a cascade is to obtain centre passage periodicity. This means to adjust the flow such that the centre passage would behave as if it were in a cascade with periodic flow from passage to passage. This could be achieved by adjusting the two bleed valves and the tailboard. However, these adjustments are not independent, since setting one (e.g. the tailboard) can affect the other (e.g. the upstream bleed flow).

Nominal periodicity was obtained by adjusting the tailboard and the bleed valves until the total pressure loss coefficients at the wakes on the two airfoils bounding the centre passage were equal (i.e. within 5%). Langston (1990) suggests that the centre passage periodicity is reached when the static pressure distribution on the airfoils bounding the centre passage closely approximates the static pressures obtained from an inviscid, irrotational flow calculation. However, this method was unsuitable for our purposes due to the lack of surface static pressure taps on the 27% and 48% airfoils. The method was used as a further check of periodicity for the baseline airfoil at exit Mach number of 0.7, where measured and analytical surface static pressure distributions were available.

5.4 Downstream Flow Survey

The wedge probe was operated under computer control which controlled

the location and the nulling**** of the probe. A total of 9 spanwise and 28 pitchwise positions were surveyed. The grid covered 55% of the span from the right hand endwall and 140% of pitch, centred about the passage #2. The grid spacing was biased toward the high gradient regions (i.e., the trailing edges and the endwall).

The approach to the end wall was limited by contact between the wedge probe tip and the endwall. For this condition the total pressure and total temperature ports were 0.164 inches and 0.078 inches from the endwall respectively.

Experiments were conducted at three nominal exit Mach numbers. The Mach numbers were as follows: 0.7 (design), 0.9 and 1.1.

The following data sets were taken for each pressure ratio:

- For the baseline airfoil, suction and pressure surface static pressure distribution at midspan on the two airfoils bounding the centre passage.
- Downstream static pressure distribution on the instrumented endwall (i.e., right endwall).
- Downstream distribution of total pressure, total temperature and exit flow angle.

**** nulling refers to the process of rotating the probe until the total pressure port is pointing directly into the incoming flow, hence providing an angle corresponding to the flow angle.

5.5 Reduction of Wedge Probe Data

Data taken with the wedge probe (yaw angle, β , total pressure, P_{owp} and total temperature, T_{owp}) were processed to obtain the following local parameters:

- Exit Mach number, M_a .
- Total pressure, P_o .
- Total temperature, T_o .
- Air density, ρ .
- Velocity, u .
- Mass flow per unit area of exit plane, m_j .

The data reduction software is included in Appendix D. The reduction scheme for the wedge probe data was as follow :

1. P_s was obtained as described in Section 5.2.
2. Local M_a calculated using P_s and P_{owp} .

$$M_a = \left[\left(\left(\frac{P_{owp}}{P_s} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right) \frac{2}{\gamma-1} \right]^{\frac{1}{2}} \quad (5.1)$$

3. If the $M_a > 1.0$ then both M_a and P_{owp} were corrected for the presence of the wedge probe normal shockwave. This was

accomplished using normal shock relations (equation 5.2) through an iterative procedure.

$$\frac{P_{o_{wp}}}{P_{o_{actual}}} = \frac{\left[\frac{\frac{\gamma+1}{2} M_a^2}{1 + \frac{\gamma-1}{2} M_a^2} \right]^{\frac{\gamma}{\gamma-1}}}{\left[\frac{2\gamma}{\gamma+1} M_a^2 - \frac{\gamma-1}{\gamma+1} \right]^{\frac{1}{\gamma-1}}} \quad (5.2)$$

4. Wedge probe temperature recovery factor , α , was calculated based on M_a and empirical correlations derived through probe calibration (see Appendix A).

$$\alpha = 0.7378 + 0.1461 M_a^2 - 0.03935 M_a^3 \quad (5.3)$$

5. Local T_o calculated based on α and $T_{o_{wp}}$.

$$\frac{T_{o_{wp}}}{T_o} = \frac{\left(\alpha \frac{\gamma-1}{2} M_a^2 + 1 \right)}{\left(\frac{\gamma-1}{2} M_a^2 + 1 \right)} \quad (5.4)$$

6. Local flow parameters calculated (i.e. T_s , ρ , u , m_2)

$$T_s = \frac{T_0}{1 + \frac{\gamma-1}{2} M^2}$$

$$\rho = \frac{P_s}{RT_s}$$

$$u = M_s \sqrt{\gamma RT_s}$$

$$m_2 = \rho \cdot u$$

5.6 Test Configurations

In light of the results of the flow visualization discussed in Chapter 4, it was decided to refinish all blades and repeat the experiment. At this point Pratt & Whitney Canada expressed an interest in investigating the effects of increasing the free-stream turbulence intensity and inlet boundary layer thickness. To comply with these changes and also stay within the allowable budget the original scope of the project was changed to include these additional configurations. Presented below is a summary of the configurations which were tested.

5.6.1 Original configuration

The original configuration refers to tests carried out before the detection of surface discontinuities. In this configuration all three airfoil profiles were tested at the nominal exit Mach number of 0.7, 0.9 and 1.1, except for the 48% blade, where the test at $M_{ex} = 1.1$ was not performed.

5.6.2 Refinished airfoils

This configuration is identical to the previous configuration, however, the airfoils used were refinished to remove the surface discontinuities. The airfoils were only tested at the exit Mach number of 0.7. This configuration was used as a baseline for comparison with other configurations.

5.6.3 Increased inlet boundary layer thickness

In order to further simulate the engine condition within the cascade environment it was decided to increase the inlet boundary layer thickness to 10% of span compared to the original 3% used in the previous test configurations. To accomplish this, an extension was constructed to be placed between the existing bellmouth and the cascade inlet (see Figure 5.1). The necessary length to achieve the desired increase in boundary layer thickness was based on classical one-dimensional turbulent boundary layer thickness

analysis as outlined in Schlichting (1968). The baseline, 27% and 48% airfoils were tested at the exit Mach number of 0.7. The results were compared to those obtained in Section 5.6.2 to study the effects of increased inlet boundary layer thickness on the total pressure loss coefficients.

5.6.4 Increased free-stream turbulence intensity

To obtain a free-stream turbulence intensity which more realistically represented the flow under engine conditions, the free-stream turbulence intensity was increased from 0.4% to 5% by means of placing a turbulence generating grid at the junction of the bellmouth and the inlet extension. The grid was designed based on the method outlined by Blair et al. (1981). All airfoil profiles were tested at exit Mach number of 0.7 and 1.0.

The value of turbulence levels quoted in the thesis are measured values. The free stream turbulence level was measured using a single wire hot wire anemometer. The measurements were performed at the upstream traverse plane with the probe perpendicular to the flow and parallel to the pitchwise direction.

a Location of turbulence generation grid

b Extension duct

Figure 5.1: Photograph of the PWC/NRC planar Cascade with the extension duct.

Chapter 6

Results and Discussion

6.1 Introduction

This chapter presents the results of the measurements obtained in the P&WC/NRC cascade. The results include passage and midspan total pressure loss coefficients, Table 6.1, and the deviation angles for all three airfoil profiles. The effects of the maximum-thickness-to-chord ratio, the inlet boundary layer thickness, the free-stream turbulence and the exit Mach number on the total pressure loss coefficient are discussed. Also discussed is the effect of the exit Mach number on the deviation angle.

The pitchwise averaged total pressure loss coefficients and exit flow angles for all profiles tested are presented in Appendix C. Also presented in Appendix C is the surface Mach number distribution for the baseline airfoil.

6.2 The effect of surface discontinuities

6.2.1 The original test

Figure 6.1 presents the passage total pressure loss coefficient with an uncertainty of ± 0.002 for all three airfoils at all exit Mach numbers tested. Also included are the predicted results for exit Mach number of 0.7 based on the modified Kacker and Okapuu correlation. The modifications were introduced based on the results obtained in this project, the details of the modifications will be described later. In all cases the trend in passage total pressure loss coefficient has remained the same as the maximum-thickness-to-chord ratio was increased. In fact the trend was predicted by the Kacker and Okapuu (1982) correlation, however, the Kacker and Okapuu correlation has underestimated the losses for all three airfoil sets at exit Mach number of 0.7.

Although discrepancies between the experimental and predicted results were expected for the 27% and the 48% airfoil, an accurate estimation of the losses based on Kacker and Okapuu correlation was expected for the baseline airfoil. However, the prediction underestimated the experimental values. As mentioned earlier in Chapter 4, these discrepancies were attributed to the presence of surface discontinuities caused by improper manufacturing processes.

The surface discontinuities have triggered the formation of a separation

bubble on the suction surface of the airfoil. The formation of the separation bubble expedited the boundary layer transition from laminar to turbulent (Schlichting, 1968), hence creating a longer length of turbulent boundary layer along the suction surface. The increased turbulent boundary layer length along the suction surface increased the surface friction loss generation thus increasing the momentum deficit thickness, θ . According to Traupel (1977), the profile loss coefficient is related to momentum deficit thickness via the following equation

$$Y_p = \left[\left(1 + \frac{\gamma M_2^2}{2} \right) \left(2 \cdot \bar{\theta} + \left(\frac{\bar{t}\theta}{1 - \bar{t}\theta} \right)^2 \cdot \sin^2 \alpha_2 + C_{p_b} \cdot \bar{t}\theta \right) \right] \quad (6.1)$$

where

$$\bar{\theta} = \frac{(\theta_{ss} + \theta_{ps})}{(S \cdot \sin \alpha_2)}$$

$$\bar{t}\theta = \frac{t\theta}{S \cdot \sin \alpha_2}$$

$$C_{p_b} = \frac{(P_2 - P_b)}{(\rho/2 \cdot V_2^2)}$$

Equation 6.1 shows that an increase in momentum deficit thickness would lead to higher profile loss which in turn leads to higher total pressure loss.

2

PM-1 3 1/2"x4" PHOTOGRAPHIC MICROCOPY TARGET
NBS 1010a ANSI/ISO #2 EQUIVALENT

6.2.2 Refinished airfoils

To confirm the effect of surface discontinuities on the momentum deficit thickness, the airfoils were refinished to remove all measurable surface discontinuities. Figure 6.2 presents the passage total pressure loss coefficient for the three airfoil profiles at exit Mach number of 0.7 after the removal of surface discontinuities. Also included in Figure 6.2 are the predicted results based on the modified Kacker and Okapuu (1982) correlation.

The passage total pressure losses decreased after removal of surface discontinuities and correspond better with the predicted values, which supports the above suggested theory. But the trend in passage total pressure loss coefficient as a function of maximum-thickness-to-chord ratio has remained unchanged.

As a consequence of the above result, the original tests (with the defective airfoils) were deemed unreliable and would not be used in the discussion.

Figure 6.1: Mass-averaged passage total pressure loss coefficient as function of maximum-thickness-to-chord ratio, defective airfoils.

Figure 6.2: Mass-averaged passage total pressure loss coefficient as function of maximum-thickness-to-chord ratio, refinished airfoils, $M_{\infty} = 0.7$

6.2.3 The trend in passage total pressure loss coefficients

Before proceeding with the explanation of the trends observed in total pressure loss coefficients, it must be explained that these airfoils were designed to operate in the same engine. As a result of this requirement several factors had to remain constant. These are: the pitch, the throat opening, the blade turning angle and the axial chord. In order to accomplish this, the true chord had to decrease as the maximum-thickness-to-chord ratio was increased. This change in the true chord affected the pitch-to-chord ratio, S/c .

The pitch-to-chord ratio is an important factor in determining the performance of turbine blades. At high pitch-to-chord the performance of turbine blades will deteriorate due to flow separations from the airfoil suction surface, and at low values of pitch-to-chord ratio the performance will drop due to increased surface friction loss. Therefore, for a given airfoil there exists an optimal pitch-to-chord ratio where the losses due to separation and surface friction balance each other in such a way that the profile loss is at minimum. According to Ainley and Mathieson (1951), see Figure 1.1, this optimal pitch-to-chord ratio is about 0.76 for a exit flow angle of 67 degrees.

Also shown by Ainley and Mathieson (1951), the effect of increasing the maximum-thickness-to-chord ratio on the profile loss becomes less profound as the flow acceleration through the passage is increased. This implies that the effect is more pronounced in impulse blades (i.e. $\beta_1 = \alpha_2$), where flow

acceleration is nearly zero, and negligible in nozzle blades (i.e. $\beta_1 \approx 0$), where flow acceleration is at its maximum. The inlet metal angle, β_1 , for P&WC airfoils are 5.44, 3.0 and 18.5 degrees for the baseline, the 27% and the 48% airfoil, respectively. This implies that the effect of the maximum-thickness-to-chord ratio is less profound for the 27% airfoil followed by the baseline and 48% airfoils.

The pitch-to-chord ratio of P&WC airfoils are 0.67, 0.78 and 0.93 for the baseline, 27% and 48% airfoils, respectively. It is evident that the 27% airfoil is in fact operating at optimal pitch-to-chord ratio. This optimal pitch-to-chord ratio coupled with the low inlet metal angle is the reason for the 27% airfoil producing the lowest profile loss coefficient, which has led to the lowest passage total pressure loss coefficient.

As for the baseline airfoil, the pitch-to-chord ratio is lower than optimal. This implies that the baseline blade is suffering from high surface friction loss which in turn has led to a higher passage total pressure loss coefficient.

The 48% airfoil is operating at higher pitch-to-chord ratio (i.e. $S/c = 0.93$), which implies that this airfoil could be experiencing flow separation from the suction surface. Also having an inlet metal angle of 18.5 degrees implies that the effect of maximum-thickness-to-chord ratio can no longer be assumed negligible. Therefore, the combined effect of pitch-to-chord ratio and inlet metal angle is believed to be the reason for the 48% airfoil experiencing higher

loss than the 27% airfoil.

In conclusion, the trend in total pressure loss coefficient corresponded well with that predicted by the Kacker and Okapuu (1982) correlation and, although the project was initiated to examine the effect of maximum-thickness-to-chord ratio on the total pressure loss, the pitch-to-chord ratio was the governing factor in determining the total pressure loss rather than the maximum-thickness-to-chord ratio.

6.3 The effect of inlet boundary layer thickness

As mentioned in Chapter 5, in order to more realistically simulate the flow under engine conditions, it was decided to increase the inlet boundary layer thickness from 3% to 10% of span. The result of an inlet flow traverse indicated that the inlet boundary layer thickness was increased to 4.4% of span rather than 10%. It is believed that this discrepancy was caused by the presence of a step between the extension and the cascade inlet. The step has triggered the separation of the boundary layer followed by the growth of a new boundary layer, either at the step location or further downstream, depending on the direction of the step (i.e., step up or step down). Although the absence of intended boundary layer thickness hindered the original intention, it nevertheless provided some interesting results which will be discussed below.

Figure 6.3, 6.4 and 6.5 show the effect of increased inlet boundary layer

thickness on the midspan and secondary losses. In all cases studied there was no significant increase in total pressure loss, however, a slight drop in the midspan loss was observed. The decrease in midspan loss is believed to be caused by the increase in the inlet boundary layer thickness which affected the convergence through the passage. Convergence refers to the case where the axial velocity density ratio across the blade row is higher than unity. The result of Rodger et al. (1992) shows that an increase in the convergence can influence the midspan loss by reducing the trailing edge loss. However, this is only a speculation since our airfoils were not instrumented with trailing edge pressure taps to allow direct measurement of the base pressure and hence the trailing edge loss.

Furthermore, in all cases the secondary loss increased as the inlet boundary layer thickness was increased. Additionally, the impact of the increased boundary layer thickness on the secondary loss became less profound as the leading edge diameter was increased (the increase in leading edge diameter is a direct result of increasing the maximum-thickness-to-chord ratio).

Although previous works (see Sieverding (1985)) have shown a dependency between the secondary loss and the inlet boundary layer thickness, it is generally believed that a constant factor is adequate to account for the effects of inlet boundary layer thickness. Furthermore, there has been no mention on the effects of leading edge diameter on the secondary loss. Figure 6.6 presents the secondary loss, Y_s , as function of the leading edge diameter

to inlet boundary layer thickness ratio, LED / δ_1 . It is evident that there exists a systematic reduction in secondary loss with increasing (LED / δ_1) . Based on this result a factor has been introduced to account for the effects of inlet boundary layer thickness and leading edge diameter on the secondary loss. This factor is summarized below:

$$\frac{Y_{sMKO}}{Y_{sKO}} = \left[0.86 - 0.0235 \left(\frac{LED}{\delta_1} \right) \right]$$

where Y_{sMKO} and Y_{sKO} are the secondary losses based on the modified and original Kacker and Okapuu (1982) correlation, respectively. The predicted results used in this project includes this modification. Figure 6.7 presents the experimental and predicted secondary losses based on the original and the modified Kacker and Okapuu correlation. It is clearly seen that the modified correlation provides a better prediction of the losses. This modification is one of two modifications introduced to the original Kacker and Okapuu (1982) correlation. The second modification which addresses the effect of the free-stream turbulence on profile loss will be discussed in the next section.

Figure 6.3: Midspan and secondary losses for the baseline airfoil at inlet boundary layer thickness of 3.0% and 4.4% of span, at exit Mach number of 0.7.

Figure 6.4: Midspan and secondary losses for the 27% airfoil at inlet boundary layer thickness of 3.0% and 4.4% of span, at exit Mach number of 0.7.

Figure 6.5: Midspan and secondary losses for the 48% airfoil at inlet boundary layer thickness of 3.0% and 4.4% of span, at exit Mach number of 0.7.

Figure 6.6: Variation of the secondary loss with increasing $\frac{LED}{\delta_1}$.

Figure 6.7: Experimental and predicted secondary losses based on the original and modified Kacker and Okapuu (1982) correlation.

6.4 Effect of the free-stream turbulence intensity

Figure 6.8 presents the total pressure loss coefficients at the free-stream turbulence intensity of 5%. The free-stream turbulence intensity is defined as follows:

$$Tu = \frac{\sqrt{u'^2}}{U_\infty} \times 100$$

Figure 6.8 shows an increase in total pressure losses with increasing free-stream turbulence intensity. The trend in total pressure loss coefficient with increasing maximum-thickness-to-chord ratio has remained unchanged.

It is generally believed that increasing the free-stream turbulence intensity would decrease the transition Reynolds number, Re_t (see Schlichting (1968)). The transition Reynolds number can either be based on the boundary layer thickness at the point of transition or the length along the surface from the total point to the point of transition. The experimental transition Reynolds number refers to the Reynolds number and therefore the point at which the transition is complete. As the transition Reynolds number decreases with increasing free-stream turbulence intensity the point of transition moves upstream, closer to the stagnation point, therefore, the length of turbulent boundary layer along the airfoil surface increases. This increased length of turbulent boundary layer would in turn increase the surface friction loss which, increases the profile and total pressure losses. The original Kacker and Okapuu

(1982) correlation proved an adequate estimate of the total pressure loss at the free-stream turbulence intensity of 0.4%, however, it underestimated the total pressure losses for free-stream turbulence intensity of 5%. This is because the Kacker and Okapuu (1982) correlation does not account for the effects of the free-stream turbulence intensity.

A literature review highlighted that the only published result on the effect of the free-stream turbulence intensity on the profile loss is that due to Dejeu and Trojanoskij (1973) (Sieverding, 1985). Based on their result, Figure 6.9, a correction factor is produced to account for the effect of the free-stream turbulence intensity on profile loss. The correlation is presented below:

$$\frac{Y_p}{Y_{p_{Tu=0}}} = (2.5 \times 10^{-4} Tu^4 - 3.6 \times 10^{-3} Tu^3 + 2 \times 10^{-2} Tu^2 + 5.1 \times 10^{-3} Tu + 0.99)$$

The predicted values based on the modified Kacker and Okapuu correlation are presented in Figure 6.8. An interesting effect observed with the free-stream turbulence intensity increase was the more uniform distribution of losses across the exit plane, Figure 6.10. Although the mechanism by which turbulence has affected the loss distribution in the exit plane is not fully understood, the following explanation is suggested. The transition Reynolds

number is strongly affected by the surface quality. In fact, the effect of surface roughness is in the same direction as that of the increasing free-stream turbulence intensity (Schlichting, 1968). Therefore, the surface roughness distribution becomes a major factor in determining the point of transition. Since the airfoils were refinished by hand, it is possible that the surface roughness distribution was not uniform over the entire airfoil surface, therefore the point of transition along the span was affected. Accordingly, this influenced the boundary layer development and created non-uniformities in loss distribution in the exit plane. With the introduction of turbulence, the transition was forced to occur closer to the leading edge and furthermore, the point of transition became less sensitive to the surface quality. Since the transition was forced to occur at approximately the same location along the span, it led to a more spanwise-uniform boundary layer development and hence, more spanwise-uniform loss distribution across the exit plane.

In conclusion, increasing the free-stream turbulence intensity has increased the total pressure loss coefficients by forcing transition from laminar to turbulent flow to occur closer to the airfoil leading edge. Furthermore, the current semi-empirical correlation schemes do not seem to account for the effect of varying the free-stream turbulence intensity level. Therefore, correction factors, such as that suggested above, should be developed to more realistically predict the effect of the free-stream turbulence intensity level on the profile loss and hence the total pressure loss coefficient.

Figure 6.8: Total pressure loss coefficient for P&WC airfoils at free-stream turbulence intensity of 5% .

Figure 6.9: The effects of free-stream turbulence intensity level on the profile loss coefficient, based on Dejc and Trojanoskij (1973), see Sieverding (1985).

a)

b)

Figure 6.10: Total pressure loss coefficient contour plots for the baseline blade at $M_\infty = 0.7$; a) $Tu = 0.4\%$, b) $Tu = 5\%$.

6.5 The effect of exit Mach number on midspan loss

Figure 6.11 presents the midspan total pressure loss coefficient, with experimental uncertainty of ± 0.006 , for the three airfoil profiles at exit Mach number of 0.7 and 1.0. The midspan total pressure losses have increased with increasing exit Mach number. Ainley and Mathieson (1951) have shown that the manner by which the profile loss, and hence the midspan loss, changes with increasing exit Mach number greatly depends on the geometry of the airfoil. In particular, they have identified the curvature of the rear suction surface as the governing factor in determining the trend. For an airfoil with a curved suction surface, such as the P&WC airfoils, in a subsonic flow, the surface friction loss increases as the local free-stream velocity and therefore velocity gradient increases (Sieverding, 1985). As the losses increase with increasing exit Mach number, it seems reasonable to expect that the amount of the increase in the losses would be a function of maximum-thickness-to-chord ratio. However, as it is apparent from Figure 6.11, the increment in the losses with increasing exit Mach number are not a function of $t_{\max}/c$.

The explanation for the trend in midspan losses with increasing exit Mach number lies with pitch-to-chord ratio.

For the 27% blade operating at optimal pitch-to-chord ratio one would expect an increase in the midspan loss with increasing exit Mach number. The increase in midspan loss is due to the increase in the free-stream velocity and

possibly a reduction in the boundary layer thickness. In both cases, the velocity gradient through the boundary layer is increased which leads to higher surface friction loss and therefore profile loss.

The baseline airfoil is operating at lower than optimal pitch-to-chord ratio. This implies that it is suffering from additional surface friction loss. Therefore, the effect of increasing the free-stream velocity (i.e., increasing the exit Mach number) and hence the velocity gradient through the boundary layer has a stronger impact on the baseline airfoil.

As for the 48% airfoil, higher than optimal pitch-to-chord ratio implies that the boundary layer on the airfoil suction surface is unstable and prone to separation. This instability coupled with the local shock wave, present at sonic condition, could create flow separations from the airfoil suction surface which would lead to higher profile and midspan losses, as observed in Figure 6.11.

In conclusion, as with the trend in total pressure loss with increasing $t_{\max}/c$, it appears that although the losses seem to increase with increasing exit Mach number, the trend is governed by pitch-to-chord ratio rather than $t_{\max}/c$.

Figure 6.11: Midspan loss as function of maximum-thickness-to-chord-ratio for $M_a = 0.7$ and 1.0.

6.6 Effect of maximum-thickness-to-chord ratio on deviation angle

Figure 6.12 presents the deviation angle, $\delta = \beta_2 - \alpha_2$, as a function of maximum-thickness-to-chord ratio with the free-stream turbulence level of 5% and exit Mach number of 0.7 and 1.0. The experimental uncertainty was ± 0.6 degrees.

Based on Ainley and Mathieson (1951), the trend in the deviation angle with increasing exit Mach number is in the direction of decreasing over-turning (over-turning refers to the case where $\delta < 0$). Except for the baseline airfoil which has exhibited high under-turning ($\delta > 0$), the 27% and the 48% airfoils have behaved as expected.

High under-turning usually indicates a flow separation from the airfoil suction surface. In such a case, an increase in the total pressure loss coefficient can be observed. However, since the experimental and predicted total pressure loss agree, it suggests that the suction surface flow separation was not present.

Based on the above discussion, the reason for the high under turning of the baseline airfoil is unknown. However, the following explanation is suggested.

As described in Section 3.2.2, the tailboard was used to achieve periodicity. The baseline airfoil proved to be less sensitive to the changes in the tailboard angle. In fact, for the baseline airfoil the periodicity was achieved purely by adjusting the bleed valves. The tailboard was set at a nominal angle corresponding to the airfoil trailing edge metal angle.

The flow after leaving the trailing edge is guided by the tailboard, therefore, if the tailboard is accidentally set at an angle less than the airfoil metal angle, it could affect the overall flow turning. In such a case under-turning may be experienced.

The above explanation can not be substantiated since the cascade was disassembled after the conclusion of the experiments on the baseline airfoil in order to prepare the cascade for the 27% and the 48% airfoil.

In conclusion, the baseline airfoil has exhibited an unusually high under-turning which is attributed to improper tailboard setting. As for the 27% and the 48% airfoils, a decrease in over-turning with increasing exit Mach number was experienced, as predicted by Ainley and Mathieson (1951).

Figure 6.12: Deviation angle as function of t_{max}/c for exit Mach number of 0.7 and 1.0.

Airfoil	Test	Exit Mach	Mass-Averaged		Area-Averaged	
			Y_{mid}	Y_T	Y_{mid}	Y_T
			± 0.006	± 0.002	± 0.003	± 0.001
Baseline	Original	0.68	0.096	0.089	0.106	0.147
		0.86	0.110	0.098	0.124	0.156
		1.1	0.110	0.112	0.116	0.160
	Config. A	0.68	0.057	0.072	0.058	0.121
	Config. B	0.68	0.046	0.070	0.047	0.117
	Config. C	0.68	0.079	0.098	0.083	0.152
		1.0	0.120	0.130	0.124	0.181
27%	Original	0.69	0.059	0.071	0.064	0.129
		0.9	0.082	0.081	0.088	0.139
		1.0	0.144	0.105	0.152	0.154
	Config. A	0.68	0.062	0.063	0.052	0.096
	Config. B	0.67	0.048	0.062	0.052	0.111
	Config. C	0.68	0.071	0.081	0.078	0.134
		0.95	0.083	0.087	0.087	0.138
48%	Original	0.67	0.073	0.075	0.080	0.139
		0.87	0.079	0.081	0.085	0.139
	Config. A	0.69	0.070	0.071	0.079	0.122
	Config. B	0.67	0.066	0.074	0.075	0.129
	Config. C	0.67	0.066	0.081	0.072	0.133
		1.0	0.099	0.112	0.109	0.160

Legend:

Original: $\delta_1/H = 3.0\%$, $Tu = 0.4\%$, defective airfoils

Config. A: $\delta_1/H = 3.0\%$, $Tu = 0.4\%$, refinished airfoils

Config. B: $\delta_1/H = 4.4\%$, $Tu = 0.4\%$, refinished airfoils

Config. C: $\delta_1/H = 4.4\%$, $Tu = 5\%$, refinished airfoils

Table 6.1: Summary of the total and midspan pressure loss coefficients.

Chapter 7

Conclusions and Recommendations for Further Work

7.1 Conclusions

The conclusions obtained in this project are as follows:

- It seems possible to design turbine airfoils operating in transonic flow regions with maximum-thickness-to-chord ratios of higher than conventional limits (above 25%) with reasonable total pressure loss coefficients.
- The 27% airfoil seems to produce lower loss compared to the baseline and 48% airfoils due to its optimal pitch-to-chord ratio.
- The trend in the deviation with increasing exit Mach number was in the direction of decreasing over-turning as predicted by Ainley and

Mathieson (1951).

- There seems to be a systematic reduction in secondary loss with increasing leading edge diameter to inlet boundary layer thickness ratio, LED/δ_1 .
- It appears that the influence of the free-stream turbulence intensity level on profile loss can be predicted satisfactorily, using a simple correction factor.
- A moderate amount of inlet turbulence level appears to have a stabilizing effect on the uniformity of the loss distribution at the exit plane.
- Attention to the airfoil surface quality is of utmost importance in attaining repeatable, reliable results.
- With the level of instrumentation and adjustments provided in this type of cascade, the question of periodicity can not be answered in definite terms.
- The uncertainty in exit mass flow rate due to the uncertainty in measuring the exit flow angles appears to have only negligible influence in obtaining the mass-averaged pressure loss coefficients.

- In order to study the effects of the maximum-thickness-to-chord ratio on total pressure loss coefficient, the pitch-to-chord ratio must be kept constant.

7.2 Recommendation for Further Work

- The introduction of turbulence showed a significant redistribution of losses in the exit plane. It would be very useful to investigate and document the effect of inlet turbulence on the performance of turbine airfoils. This could aid in either establishing a standard for the free-stream turbulence intensity, under which all future cascade work will be performed, or, finding a more comprehensive correlation which could relate the results obtained in the cascade with different free-stream turbulence intensity levels.
- In order to produce repeatable, reliable results, it is important to establish the critical Reynolds number for a given inlet turbulence level and be certain that the cascade is operated above that limit.
- It would be useful to investigate the effects of the leading edge diameter on the secondary loss. It appears that increasing the leading edge diameter at the blade root could be used as a method to reduce the secondary loss.

- It would be helpful to perform more flow visualization and obtain Schlieren photographs at different exit Mach numbers. This would allow better understanding of the physics involved and could aid in explaining the trends in losses and deviations experienced in this project.
- It would be necessary to redesign the planar cascade and its accompanying data acquisition system to allow faster, more accurate results to be obtained and to allow the above-suggested tasks to be performed.

7.3 Suggestions for redesigning the planar cascade

7.3.1 Introduction

The design of turbine cascades in general may be divided into three distinct categories: subsonic, transonic and supersonic cascades. One of the most important conditions, which must be met to ensure accurate, reliable results is attaining periodicity. The means by which periodicity is achieved in a planar cascade of turbine airfoils differs depending on exit flow conditions. In a subsonic cascade reasonable levels of periodicity may be achieved if adequate number of airfoils (conventionally more than seven and preferably odd numbered (Gostelow, 1984)) and bleed control mechanisms are provided. In supersonic cascades, it is quite important to ensure that the blading is

dimensionally stable and that the throat width of each passage is equal. Furthermore, attention to the discharge boundary condition of the supersonic jet is of utmost importance. In supersonic flow, there will be expansion and shock waves emanating from the trailing edge region of the airfoils, and the interaction of these waves with the downstream flow and passages can seriously affect the periodicity. This problem could be dealt with by placing a porous tailboard downstream of the flow where it will eliminate any expansion or shock wave reflecting from the shear layer, thereby eliminating the problem. In a transonic cascade both subsonic and supersonic conditions are observed and therefore, the designed cascade must include necessary provisions to meet the periodicity condition both at subsonic and supersonic speeds. This necessity makes the design of transonic cascades more difficult than either subsonic or supersonic cascades.

7.3.2 Redesigning the Planar Cascade

A preliminary study on possible designs showed that a cascade which allows for all the suggestions in Section 7.2 would be quite complex and expensive. One of the problems is to vary the Reynolds number. Traditionally in suction type cascades, this is achieved by throttling the inlet flow. Once the flow is throttled, provisions must be made to ensure uniform inlet flow entering the working section. This could be achieved by providing a settling chamber before the working section. Designing a settling chamber for a suction type cascade is far more challenging than that of the blower type, due

to the direction of the net force acting on the chamber. Such a design would require thick walls which implies a massive, heavy structure. Therefore, it was suggested to use the existing cascade to carry out this study. Once the critical range is established, then it is easy to construct a cascade to allow for other requirements.

One of the fundamental problems with the present planar cascade is the fact that the flow entering the cascade vertically must be turned by 90 degrees in order to enter the exhauster intake. About 67 degrees of this turning occurs within the nozzle, the rest must take place between the trailing edge and the exhauster intake. The mechanism by which this turning takes place is not fully understood, however, it is certain that this turning does occur between these two points and its effect on the exit flow angle in the plane of measurement must not be ignored. In order to alleviate this problem, two solutions are suggested. One would be to allow the jet exiting the nozzle to discharge in to a settling chamber, the other would be to make sure that the overall cascade turning is equal to the turning of the airfoils. The first solution is preferable since it allows for airfoils with different turning to be tested in the same cascade.

Finally, larger boundary layer bleeds and a more accurate mechanism for tracking the tailboard position must be incorporated. In addition, to facilitate the disassembly and re-assembly of the cascade for flow visualization purposes, it is suggested that clamps, rather than screws, be used to secure the endwalls.

7.3.3 Redesigning the Instrumentation

In order to ensure that the inlet flow is uniform, it is necessary to be able to measure the incidence of the incoming flow. One possible way to achieve this is by identifying the location of the stagnation point on the leading edge of the airfoil. To accomplish this, it is necessary to provide closely spaced static pressure taps in the leading edge region of the airfoils. Furthermore, static pressure taps on the midspan of both the suction and pressure surfaces should be provided. The static pressure distribution, along with the inviscid, irrotational analytical solution could be used to check if the condition of periodicity is met (Langston, 1990). Furthermore, in order to deduce the trailing edge loss, pressure taps should be provided at the airfoil trailing edge.

Since the exhauster at the NRC's National Compressor/Exhauster Facility costs about \$500 per hour to operate, in the long run, it may prove time and cost effective to invest in a fast traverse gear and data acquisition system to reduce the overall running time (which is about 3 hours for 2 D downstream total pressure and yaw measurement).

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Appendix A

Wedge Probe and Thermocouple Calibration^{††}

A.1 Introduction

Calibrations of both United Sensor wedge probes and copper-constantan thermocouples are described in this appendix. All data from temperature calibrations were in SI units. Correlation equations based on these data are given only in SI units.

United Sensor two-dimensional wedge probes were used to measure gas stream total temperature, total pressure and gas stream flow direction in a plane perpendicular to the probe axis. An illustration of the United Sensor Wedge Probe is shown in Figure A.1. The probes were supplied from the manufacturer with calibration sheets. Unfortunately, the extent of the calibrations was limited to Mach numbers less than 0.30. In order to calibrate the wedge probes over both the expected Mach number range ($0.4 \leq Ma \leq 1.2$)

††

Since the wedge probe and thermocouples used in this project are same as those used by Carscallen(1990), the following Appendix is a direct reproduction of Carscallen (1990), Appendix A, pp 129-148

Figure A.1: Illustration of United Sensor Wedge Probe - WT187

and varying probe yaw angles, a comprehensive probe calibration program was undertaken. This program covered calibration of the copper-constantan thermocouple within the probe tip and calibration of the wedge probe's ability to correctly measure total temperature and pressure as a function of both yaw angle and Mach number. Probe Reynold's number and flow Mach number similarity was maintained between that expected in the planar cascade and the transonic calibration tunnel.

Results of the wedge probe calibrations allowed determination of wedge probe total temperature recovery factors as a function of Mach number and thus allowed the correct interpretation of temperature data taken with the wedge probes. Interpretation of total temperature measurements from the HLTTR had been found by Williamson [8] to be difficult due to his inability to obtain precise wedge probe total temperature recovery factors as a function of Mach number.

Calibration of the type "T" copper-constantan thermocouples was carried out by the NRC Division of Physics Laboratory for Basic Standards. Calibration of the United Sensor wedge probes was carried out in the 5.0 by 5.0 inch (127.0 by 127.0 mm) transonic test section of the pilot blow down tunnel at the High Speed Aerodynamics Laboratory of the National Aeronautical Establishment, NRC.

Three wedge probes were calibrated, one of two original wedge probes from the HLTTR and two newly purchased wedge probes manufactured to the same specifications as the originals. The model number of the United Sensor Wedge Probes and respective serial numbers are as follows:

Model Number	WT-187-24-F-18-CD-C/C-36
Probe Serial Numbers	
B 1966-1	
B 2169-1	B 2169-2

Table A.1: United Sensor Wedge Probes

A.2 Experimental Procedures

A.2.1 Thermocouple Calibration

Calibration of the type "T" copper-constantan thermocouples was carried out by the staff of the NRC Division of Physics Laboratory for Basic Standards. The thermocouples were calibrated in a stirred liquid bath from -58 to 58°F ; 1.8°F steps (-50 to 50°C ; 1.0°C steps). A table of thermocouple EMF (microvolts) vs temperature ($^{\circ}\text{C}$) was generated for each thermocouple.

Second or 3rd-order polynomials were fit to the calibration data by the method of least squares. The data employed in the polynomial curve fits were those data covering the anticipated operating temperature ranges.

A.2.2 Wedge Probe Calibration

Calibration of the United Sensor wedge probes was carried out in the 5.0 by 5.0 inch (127.0 by 127.0 mm) transonic test section of the pilot blow down tunnel at the High

Speed Aerodynamics Laboratory of the National Aeronautical Establishment (NAE), NRC by NAE staff. Standard NAE calibration procedures were followed.

No tests were carried out to determine the effect of probe immersion depth or pitch angle on measurement of either total pressure or temperature. The probes were cantilevered out from one side of the tunnel such that the probe tip was located at the tunnel centre line. Prior to calibration of the wedge probes any twist in the probe shafts was determined by measuring the angle between the half moon probe head and the wedge shaped sensing end. Knowledge of the twist allowed the probes to be installed in an L.C. Smith angular actuator which itself was mounted on the pilot tunnel such that a zero yaw angle specified the probe was pointing exactly upstream.

Calibrations were initially carried out at four Mach numbers and seven probe yaw angles. Initial Mach numbers were as follows:

$$M_a = 0.4, 0.9, 1.0 \text{ and } 1.2;$$

whilst the yaw angle range was as follows:

$$\text{Yaw angle} = \pm 30.0^\circ, \pm 20.0^\circ, \pm 10.0^\circ \text{ and } 0.0^\circ.$$

The Mach number range covered allowed an assessment of the calibration results for compressibility and supersonic flow effects. This assessment led to additional data being taken at the following transonic Mach numbers:

$$M_a = 0.95, 0.98, 1.01, 1.02, 1.03, 1.04, 1.08, 1.18.$$

A.3 Theory

Derivation of the wedge probe recovery factor is now presented.

The wedge probe recovery factor α is defined as follows:

$$\alpha = \frac{T_{op} - T}{T_o - T} \quad (A.1)$$

where

T_{op} is total temperature as measured by the wedge probe.

T_o is gas stream total temperature.

T is gas stream static temperature.

With the following relations

$$T_o - T = \frac{V^2}{2C_p}$$

$$R = C_p - C_v$$

$$k = \frac{C_p}{C_v}$$

equation A.1 may be rewritten as follows:

$$\alpha = \frac{\frac{T_{op}}{T} - 1}{\frac{V^2}{2C_p T}} = \frac{\frac{T_{op}}{T} - 1}{\left(\frac{k-1}{2}\right) M_a^2} \quad (A.2)$$

where M_a^2 is true Mach number at the wedge probe location.

Equation A.2 is the relationship between α and T_{op} , T and M_a for a wedge probe. From a measurements perspective the static temperature, T is extremely difficult to measure.

To make the equation more tractable the ratio of $\frac{T_{op}}{T}$ is transformed as follows:

$$\frac{T_{op}}{T} * \frac{T}{T_o} = \frac{T_{op}}{T_o}$$

The ratio $\frac{T_{op}}{T_o}$ is a measurable quantity and thus Equation A.2 is reworked to reflect this new total temperature ratio. Now:

$$\frac{T_o}{T} - 1 = \frac{V^2}{2C_p T} = \frac{V^2}{2C_p T} * \frac{kR}{kR}$$

$$\frac{T_o}{T} = \left(\frac{k-1}{2} \right) M_a^2 + 1$$

and thus

$$T = \frac{T_o}{\left(\frac{k-1}{2} \right) M_a^2 + 1} \quad (\text{A.3})$$

Substituting Equation A.3 into Equation A.2 yields the following:

$$\alpha = \frac{\frac{T_{op}}{T_o} \left[\left(\frac{k-1}{2} \right) M_a^2 + 1 \right] - 1}{\left(\frac{k-1}{2} \right) M_a^2} \quad (\text{A.4})$$

Equation A.4 displays the dependence of the total temperature recovery factor, α on both $\frac{T_{op}}{T_o}$ and M_a .

A.4 Results and Discussion

A.4.1 Thermocouple Calibration

Table A.2 lists the 2nd or 3rd-order polynomials which correlate thermocouple EMF (μV) with temperature ($^{\circ}\text{C}$). Uncertainty of calibration was $\pm 0.1^{\circ}\text{C}$ over a range of -50 to 50°C ; 1.0°C steps. The residuals between the observed and predicted data from the curve fits were less than 0.01°C .

A.4.2 Wedge Probe Calibration

Figures A.2, A.3 and A.4 are plots of α versus M_a for the wedge probes calibrated. Wedge probes B-2169-1 and B 2169-2 were used in the present experimental studies, while wedge probe B 1966-1 was used in the HLTTR studies. Time constraints in the transonic pilot tunnel resulted in only wedge probe B 2169-1 being calibrated over the

Name	Range °C	Polynomial
B 2169-2	-10 to 30 °C	$\text{EMF} = 38.59722 \cdot T + 4.106183 \text{E-} 2 \cdot T^2$ $T = 2.590438 \text{E-} 2 \cdot \text{EMF} - 6.728934 \text{E-} 7 \cdot \text{EMF}^2$
IPTS	-10 to 40 °C	$\text{EMF} = 38.69695 \cdot T + 3.787755 \text{E-} 2 \cdot T^2$ $+ 4.464254 \text{E-} 5 \cdot T^3$ $T = 2.584177 \text{E-} 2 \cdot \text{EMF} - 6.538843 \text{E-} 7 \cdot \text{EMF}^2$ $+ 1.382394 \text{E-} 11 \cdot \text{EMF}^3$
T_i	10 to 40 °C	$\text{EMF} = 38.81165 \cdot T + 4.284843 \text{E-} 2 \cdot T^2$ $T = 2.572415 \text{E-} 2 \cdot \text{EMF} - 6.495882 \text{E-} 7 \cdot \text{EMF}^2$
DT_o	10 to 40 °C	$\text{EMF} = 38.81378 \cdot T + 4.143040 \text{E-} 2 \cdot T^2$ $T = 2.572531 \text{E-} 2 \cdot \text{EMF} - 6.304412 \text{E-} 7 \cdot \text{EMF}^2$

Table A.2: Thermocouple EMF/Temperature Correlations

entire Mach number range with emphasis being place on the transonic region $0.90 \leq M_a \leq 1.15$ and limited interest being placed on the range of $0.40 \leq M_a \leq 0.85$.

Figure A.2: α versus M_a for Wedge Probe B 2169-1

Variation of α with M_a for wedge probe B 2169-1 is readily seen in Figure A.2. The lay of the data from wedge probe B 2169-2 over the transonic region as seen in Figure A.3 is the same as that for probe B 2169-1, although the magnitude of the data for probe B 2169-2 is a few percentage points lower. The data from wedge probe B 1966-1 is shown in Figure A.4. Though the data are sparse, the dependence of α on M_a is obvious and the magnitude of the dependent variable α in Figure A.4 for a given M_a is similar to that

Figure A.3: α versus M_a for Wedge Probe B 2169-2

Figure A.4: α versus M_a for Wedge Probe B 1966-1

presented in Figures A.2 and A.3. The magnitude and trends displayed in the data of Figures A.2, A.3 and A.4 are very similar as would be expected with data from wedge probes of the same design and manufacturer. Noteworthy is the fact that α decreases in magnitude with decreasing values of M_a . The present calibration studies are insufficient to make further comment on this phenomena.

Unfortunately, during the preliminary testing of the planar cascade the thermocouple in wedge probe B 2169-1 was irreparably damaged. Furthermore, insufficient calibration data had been taken for wedge probe B 2169-2 (the backup wedge probe) to allow the required correlation between α and M_a to be calculated. The following describes the approach taken to overcome this problem and generate the required correlation.

As previously mentioned, wedge probe calibration data were wedge probe specific but similar in trend and magnitude due the commonality of the physics describing the flow about and through the wedge probes. Thus, a decision was made to amalgamate all wedge probe data in order to represent the variation of α with M_a for use with the one remaining wedge probe, B 2169- 2.

This amalgamated data set had one conspicuous failing; the lack of data over Mach number ranges $0.40 \leq M_a \leq 0.85$ and $1.15 \leq M_a \leq 1.20$. To obtain data for these two missing ranges and thus obtain a better description of the relationship between α and M_a over the entire range of $0.40 \leq M_a \leq 1.20$, unpublished calibration data for wedge probes B 1966-1 and B 1966-2 from Williamson [8] were employed.

This enhanced data set was utilised in the generation of a 3rd-order polynomial which described the relationship between α and M_a over the entire range of Mach numbers

covered during calibration. This data set, correlation curve and it's equation are displayed in Figure A 5.

Figure A.5: α versus M_a for Amalgamated Wedge Probe Data Set

The accuracy with which the equation displayed in Figure A.5

$$\alpha = 0.7378 + 0.1461 * M_a^2 - 0.03935 * M_a^3 \quad (A.5)$$

represents the actual α versus M_a for wedge probe B 2169-1 may only be estimated. It is assumed that the shape of the curve is correct but that the Y-intercept may be in error. Following is an estimation of the possible error in this approach.

Assume $M_a = 1.0$ and $T_{op} = 530.0^\circ R$ If the value of α from Figure A.3 is assumed

Data	Recovery Factor (α)	T_o ($^{\circ}R$)
Equation A.5	0.8438	544.2
Figure A.3	0.8100	547.3

correct then the error associated with the use of Equation A.5 to calculate T_o is as follows:

$$\text{percent error} = \left(1 - \frac{544.2}{547.3}\right) * 100.0 = 0.56$$

Figure A.6 is a plot of the error in the measurement of total pressure by the wedge probe B 2169-2 as a function of Mach number. The error ranges from 0.01 % to 0.6 % with the average being 0.25 %. Figure A.7 is a plot of the error in the measurement of total pressure by the wedge probe B 2169-1 as a function of Mach number. The data plotted on the latter figure are more extensive and reveal the excellent total pressure measuring capabilities of these wedge probes. The apparent increase in error at $M_a \geq 1.0$ is attributable to loss in total pressure across the wedge probe bow shock wave.

Figure A.8 is a plot of the error in the measurement of total pressure by the wedge probe B 2169-2 as a function of yaw angle. Only positive yaw angles are displayed as the data from wedge probe B 2169-1 were found to be symmetric about a Yaw angle = 0.0° . The effect of decreasing Mach number is to lessen the effect of yaw angle on the error in measuring the total pressure. The single data point at (30,0.075) is for $M_a = 0.4$, all other data was taken for transonic Mach numbers. Errors in the measurement of total pressure are less than 1.0 % for Yaw angles $\leq 8.0^{\circ}$.

Figure A.9 is a plot of the error in the measurement of total temperature by the wedge probe B 2169-2 as a function of yaw angle. Only positive yaw angles are displayed as the

data from wedge probe B2169-1 were found to be symmetric about a Yaw angle = 0.0° . Errors in the measurement of total temperature, relative to the error at a Yaw angle = 0.0° are less than 0.5 % for Yaw angles $\leq 30.0^\circ$. The errors in measuring total temperature for Yaw angle = 0.0° are due to Mach number and recovery factor effects and have been discussed previously.

Figure A.10 is a plot of the error in the measurement of static pressure by the wedge probe B 2169-2 as a function of Mach number. Data not shown on this figure from wedge probe B 2169-1 indicate that the static pressure at a Mach number of 0.4 as measured by the wedge probe is correct. This is in agreement with the claims of the manufacture United Sensor Ltd. and may be assumed true for this wedge probe B 2169-2. The figure indicates that the errors in measuring the local static pressure become very significant at high subsonic Mach numbers, reaching error levels of 40 %. As a result of this no static pressure measurements were taken with the wedge probes during the course of the present experimental program.

Figure A.6: $(P_o - P_{owp}) / P_o$ vs M_p (Yaw = 0.0°) for Wedge Probe B 2169-2

Figure A.7: $(P_o - P_{oup}) / P_o$ vs M_p (Yaw = 0.0°) for Wedge Probe B 2169-1

Figure A.8: $(P_o - P_{owp}) / P_o$ vs Yaw Angle for Wedge Probe B 2169 2

Figure A.9: $(T_o - T_{oup}) / T_o$ vs Yaw Angle for Wedge Probe B 2169-2

Figure A.10: $(P_{swp} - P_{st}) / P_{st}$ vs M_p (Yaw = 0.0°) for Wedge Probe B 2169 2

Appendix B

Uncertainty Analysis^{*}**

B.1 Introduction

Single sample uncertainty analysis was employed to determine uncertainty in the experimental results. This uncertainty analysis was based on the work of Moffat (1982), (1985), (1988), Kline (1985) and Kline and McClintock (1953). Single sample, refers to single sample experiments in which each data point was measured once or at most a few times. The term 'uncertainty' refers to the overall possible error that the experimental results might have.

In the present experimental study the signals from the thermocouples or

The following chapter in a direct reproduction of Carscallen (1990) (pp 76-93), except for the last section where the uncertainty for total pressure loss coefficient is presented.

pressure transducers were integrated over ten power line cycles and then averaged to filter out 60 Hz noise. One hundred of these filtered signals were then averaged to yield the required and subsequently processed single sample measurement. Noting that statistical conclusions are independent of units employed, only Imperial units are used in this chapter.

Moffat (1982) concluded that there were three baseline levels of supplementary experimental replication which were useful to consider when carrying out uncertainty analysis. They were as follows:

- Repeated trials with the process held absolutely constant.
The zeroth-order replication.
- Repeated trials with the process running, but no fixed errors.
The first-order replication.
- Repeated trials with the process running and all fixed errors 'on'.
The Nth-order replication.

The detailed studies, specifically the 0th, 1st and Nth-order replication outlined by Moffat were applied to those measurements which were required to investigate the potential for redistribution of total temperature through the planar cascade. The following sections describe the methods employed and results of the uncertainty analysis.

B.2 Traverse Gear Positional Uncertainties

Table B.1 lists uncertainties for the L.C. Smith traverse gear which was used to perform the upstream and downstream spanwise and pitchwise flow surveys. The last column indicated the 95% confidence interval, $\pm 2S$ for these measurements. S , is the standard deviation of the sample. Calibration constants for the three L.C. Smith components making up the traverse gear are deemed to be 'hard' numbers, ie. the accuracy of the measurement of the calibration constants was an order of magnitude greater than the resolution of the L.C. Smith position indicators. Units for the calibration constants are either counts per inch (cpi) or counts per degree (cpd). Note, counts are designated as 'c'.

L.C. Smith actuators employed twenty five turn 1000Ω wire wound potentiometers to feed back actuator position. Linearity of the potentiometers was $\pm 0.1\%$ ie. $\pm \Omega$. Nominally, 95% of the potentiometer range covered the entire range of actuator movement. Suitable electronics converted resistance into an indicated position which was then displayed as a number between 1 and 999, this number was called a 'count' or output as a voltage between 0.00 and 9.99 volts. Thus, linearity was interpreted as ± 1 count.

Actuator	Cal.	Pot	Positional	Gear	$\pm 2S$
Movement	Const.	Linearity	Readout	Play	
Pitch	123.51 cpi	± 1 c	± 0.5 c	0.019 in	± 0.021 in
Span	130.10 cpi	± 1 c	± 0.5 c	0.015 in	± 0.017 in
Yaw	5.62 cpd	± 1 c	± 0.5 c	0.28 deg	± 0.350 deg

Table B.1: L.C. Smith Traverse Gear Positional Uncertainties

Spanwise and pitchwise positional uncertainty may be interpreted as uncertainty in indicated or recorded position of the wedge or total head probe during a complete spanwise / pitchwise flow survey. Yaw positional uncertainty refers to uncertainty in the L.C. Smith Yaw actuator and does not fully describe uncertainty in 'measured' yaw angle.

B.3 Pressure Transducer Uncertainties

Table B.2 lists the 95% confidence interval ($\pm 2S$) in the slopes (input pressure / output voltage) of the pressure transducer calibration curves. Units of the $\pm 2S$ column are given as percentage of slope and were subsequently converted to units of pressure by multiplying them by; the average pressures as determined from the 1st-order uncertainty analysis which was carried out using data taken at the design pressure ratio, $PR = 2.3$.

B.4 P_{owp} , ΔP_{swp} , Yaw, P_{atm} , T_{in} , ΔT_o and Q_{in} Measurement Uncertainties

B.4.1 Zeroth-Order Uncertainties

Zeroth-order uncertainty estimates the overall uncertainty of both the fixed and variable errors arising from the instrumentation system alone. This uncertainty analysis was carried out as two separate ancillary experimental procedures. For both procedures there was no air flow through the planar cascade, the inlet to the bellmouth was capped and the isolation valve between the planar cascade and the Brown Boveri exhaustor was closed. These precautions were taken to limit the air movement within the planar cascade and maintain the temperature within the planar cascade at a nominally constant value.

Function	Make	Serial No.	Range (Imp)	Range (SI)
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Scanivalve Transducers

No. 1	Statham	61160	0.386 psi/mV	±0.26%
No. 2	Statham	61161	0.380 psi/mV	±0.26%
No. 3	Statham	59269	0.409 psi/mV	±0.49%
No. 4	Statham	61134	0.404 psi/mV	±0.99%

Stand-Alone Transducers

Pitot tube	Statham	392	0.128 H ₂ O/mV	±0.00%
Operating point	Statham	14205	0.238 H ₂ O/mV	±0.42%
Wedge p. total	Statham	59270	0.232 psi/mV	±0.43%
Wedge p. static	Kulite	4074-2-12	0.259 psi/mV	±0.39%
Wedge diff. stat	Kulite	4074-2-10	0.260 psi/mV	±0.77%
Barometric	Viatran	11427168	1.000 psi/V	±0.40%

Table B.2: Planar cascade pressure transducer specifications.

Firstly, the wedge probe (B2169-2) and its companion thermocouple for the measurement of ΔT_o were immersed in the Guildline constant temperature bath. The Guildline bath was set at 15 °C. The data acquisition system was operated as per a full downstream spanwise and pitchwise traverse. Thirty one data points were used in the statistical analysis for ΔT_o . Results were as follows:

Average ΔT_o

$$\overline{\Delta T_o} = 0.055 \text{ } ^\circ\text{F}$$

Standard deviation of the sample, S

$$S = 0.010 \text{ } ^\circ\text{F}$$

95% confidence interval ($\pm 2S$)

$$\pm 2S = \pm 0.020 \text{ } ^\circ\text{F}$$

The average ΔT_o was expected to have been 0.0 and thus its non-zero value (0.055°F) represented a positive fixed zero-offset error which was subsequently incorporated in the data reduction scheme and subtracted from all ΔT_o data values. This fixed zero-offset error in the averaged value of ΔT_o was attributed to limitations in the thermocouple calibrations.

Secondly, the wedge probe and its associated thermocouple were reinstalled in the planar cascade and the data acquisition system was operated as per a full downstream spanwise and pitchwise traverse. Thirty one data points were used in the zeroth-order statistical analysis for P_{owp} , ΔP_{swp} , Yaw,

P_{atm} , T_m , ΔT_o , Q_{in} and P_{sDR10} . P_{sDR10} was a representative downstream static pressure reading taken at midpitch by the ganged Scanivalve system from pressure tap DR10. Results are listed in Table B.3. The last row indicated the 95% confidence intervals, $\pm 2S$.

Note, all raw pressure measurements used in this uncertainty analysis were made relative to SPIR, a static pressure tap on the right hand endwall at the inlet to the test section. An exception to the aforementioned was the differential measurement of ΔP_{swp} , the static pressure differential between the two wedge probe static pressure taps which was used to null the wedge probe. Inlet 'Q' (Q_{in}) referenced SPIR to the barometric total pressure, P_{atm} .

	P_{swp}	ΔP_{swp}	Yaw	P_{atm}	T_m	ΔT_o	Q_{in}	P_{sDR10}
	psi	psi	deg	psi	'R	F	psi	psi
Average	0.000	0.000	195.2	14.810	529.730	0.268	0.000	0.000
S	0.000	0.000	0.0	0.000	0.013	0.031	0.000	0.000
$\pm 2S$	± 0.000	± 0.000	± 0.0	± 0.000	± 0.026	± 0.062	± 0.000	± 0.000

Table B.3: Zeroth-Order Uncertainty Results

These results indicate that there was no significant zero offset or variability in the data under no flow conditions. The average yaw angle of 195.2 degrees indicates the wedge probe 'home' position. The negative value of ΔT_o indicates a slight thermal stratification in the quiescent planar cascade. Note the increase in standard deviation for ΔT_o from 0.01 to 0.03 as the wedge

probe and its associated thermocouple were moved from the Guildline constant temperature bath to the planar cascade.

B.4.2 First-Order Uncertainty

First-order uncertainty analysis estimates the uncertainty in the results due to process unsteadiness. The results of thirty one (31) repeated trials with the planar cascade operating at the design pressure ratio ($PR = 2.3$) and using the data acquisition system to record the results of a full downstream spanwise and pitchwise traverse were employed in this ancillary experiment. The wedge probe (B2169-2) and its companion thermocouple for the measurement of ΔT_o were installed as per a full downstream spanwise and pitchwise traverse. The data acquisition system was operated with the L.C. Smith traverse gear repeatedly returning to the same representative location. Thirty one data points were obtained and used in the 1st-order statistical analysis of P_{owp} , ΔP_{swp} , Yaw, P_{atm} , T_i , ΔT_o , Q_{in} and P_{sDR10} . Results are listed in Table B.4. The last row indicates the 95% confidence interval, $\pm 2S$.

	P_{owp}	ΔP_{swp}	Yaw	P_{atm}	T_i	ΔT_o	Q_{in}	P_{sDR10}
	psi	psi	deg	psi	$^{\circ}R$	$^{\circ}F$	psi	psi
Average	0.342	0.016	80.58	14.810	523.98	-13.450	0.157	8.205
S	0.022	0.124	0.36	0.001	0.670	0.936	0.000	0.005
$\pm 2S$	± 0.0044	± 0.248	± 0.07	± 0.002	± 1.340	± 1.872	± 0.000	± 0.009

Table B.4: First-Order Uncertainty Results

Inlet 'Q' (Q_{in}) was continuously measured and recorded in order to normalize all data with respect to this parameter and thus remove any influence of inlet flow variability on the data. As may be seen, the standard deviation of the sample of Q_{in} is 0.000. This indicates insignificant variability in inlet 'Q' and normalization of measured data was thus not carried out with respect to Q_{in} , although Q_{in} was still measured and recorded.

S, calculated above, is the standard deviation of the sample about the mean of the data set of 'n' values. As this uncertainty analysis was concerned with the uncertainty of any result reported on, S had to be correlated to the standard deviation of the population, σ . This correlation was carried out using the chi-squared statistic, which related the standard deviation of a sample S of 'n' values to the standard deviation of its parent population, σ . The chi-squared statistic yields a range for σ , the random uncertainty component in the measurement. The extent of the range was dependent on the number of readings in the data set. The range gave the most likely ($\sigma = S$), the largest or the smallest estimate of σ from S. The range of σ given that there were 31 readings in the data set was as follows:

$$0.80S \leq \sigma \leq 1.34S.$$

The most likely estimate of σ was chosen, ie. $\sigma = S$.

B.5 Nth-Order Uncertainty

B.5.1 Introduction

Moffat (1982) states that Nth-order uncertainty of a result is a measure of its overall uncertainty, accounting for all sources of fixed and variable errors in the experiment as it was run. The Nth-order uncertainty was calculated as the root-sum-square (RSS) combination of the first-order uncertainty $\delta X_{i,1}$, where

$$\delta X_{i,1} = 2S_i$$

with the RSS combination of fixed errors from every source.

$$\delta X_{i,N} = \left[(\delta X_{i,1})^2 + (\text{RSS} \delta X_{i,\text{fixed}})^2 \right]^{1/2} \quad (\text{B.1})$$

where $\delta X_{i,1}$ is the first-order uncertainty in X_i , and $\text{RSS} \delta X_{i,\text{fixed}}$ is the root-sum square of all fixed error contributions from every level of error source.

In the following subsections Equation B.1 will be reduced for the sake of brevity to the following equation:

$$\delta X_{i,N} = \text{RSS}(\delta X_{i,1}, \delta X_{i,0}, \delta X_{i,\text{fixed}}, \text{etc.})$$

B.5.2 Nth-Order m_1 Uncertainty

Uncertainty in the centre nozzle inlet mass flow, m_1 was dependent on

uncertainties in T_{in} , P_{atm} , Q_{in} and inlet area A_{in} . The following assumptions were made for this uncertainty analysis:

$$P_{in} = P_{atm} = 14.70 \text{ psi}$$

$$T_{in} = 530.0 \text{ } ^\circ\text{R}$$

$$PR = 2.30$$

$$Q_{in} = 0.151 \text{ psi}$$

To determine uncertainty in m_1 at the design pressure ratio the method of sequential perturbation was carried out on the equation relating the above four variables to m_1 . This equation is as follows:

$$m_1 = A_{in} \left(\frac{P_{atm}}{RT_{in}} 2gQ_{in} \right)^{\frac{1}{2}}$$

Inlet area, A_{in} was determined as follows:

A power law velocity distribution was assumed for the velocity distribution within the endwall boundary layer. ie.

$$\frac{u}{U_\infty} = \left(\frac{y}{\delta} \right)^{\frac{1}{n}}$$

and

$$\frac{\delta_1}{\delta} = \frac{1}{1+n}$$

Knowing

$$\delta = 0.5 \text{ in}$$

$$\text{airfoil span} = 4.440 \text{ in}$$

$$\text{airfoil pitch} = 5.817 \text{ in}$$

$$\text{and assuming } n = 6 \pm 1$$

then

$$A_{in} = \left(\text{Span} - \frac{2\delta}{n+1} \right) \text{Pitch}$$
$$A_{in} = \left(4.440 - \frac{2 \cdot 0.5}{n+1} \right) 5.817$$

Nth-order uncertainties in the variables T_{in} , P_{atm} , Q_{in} and A_{in} were evaluated and are as follows:

$$\delta T_{in, N} = \text{RSS} (\delta T_{in, 1}, \delta T_{in, \text{calibration}})$$

$$\delta T_{in, N} = [(1.340)^2 + (0.018)^2]^{0.5}$$

$\delta T_{in, N} = 1.340^\circ \text{R}$ and thus, the reported values of T_{in} are accurate to within $\pm 1.340^\circ \text{R}$ ($\pm 0.744^\circ \text{C}$) of the actual value 95% of the time.

Similarly

$$\delta P_{atm, N} = \text{RSS} (\delta P_{atm, 1}, \delta P_{atm, \text{calibration}})$$

$$\delta P_{atm, N} = [(0.001)^2 + (0.059)^2]^{0.5}$$

$\delta P_{atm, N} = 0.059 \text{ psi}$ and thus, the reported values of P_{atm} are accurate to within $\pm 0.059 \text{ psi}$ of the actual value 95% of the time.

Similarly

$$\delta Q_{in, N} = \text{RSS} (\delta Q_{in, a}, \delta Q_{in, \text{calibration}})$$

$$\delta Q_{in, N} = [(0.000)^2 + (0.000)^2]^{0.5}$$

$\delta Q_{in, N} = 0.000$ psi and thus, the reported values of Q_{in} are accurate to within ± 0.000 psi of the actual value 95% of the time.

Finally, the Nth-order uncertainty in A_{in} as a function of the uncertainty in 'n' is as follows:

$\delta A_{in, N} = \pm 0.104 \text{ in}^2$ and thus, the reported values of A_{in} are accurate to within $\pm 0.104 \text{ in}^2$ of the actual value 95% of the time.

Following are the uncertainties, as were determined by the method of sequential perturbation in m_1 for the design pressure ratio resultant from Nth order uncertainties in T_{in} , P_{atm} , Q_{in} and A_{in} .

$$\frac{\partial m_1}{\partial T_{in, N}} \delta T_{in, N} = 0.002$$

$$\frac{\partial m_1}{\partial P_{atm, N}} \delta P_{atm, N} = 0.004$$

$$\frac{\partial m_1}{\partial Q_{in, N}} \delta Q_{in, N} = 0.000$$

$$\frac{\partial m_1}{\partial A_{in, N}} \delta A_{in, N} = 0.001$$

Thus, the Nth-order uncertainty in m_{in} was as follows:

$$\delta m_{in, N} = \left[\left(\frac{\partial m_i}{\partial T_{in, N}} \delta T_{in, N} \right)^2 + \left(\frac{\partial m_i}{\partial P_{atm, N}} \delta P_{atm, N} \right)^2 + \left(\frac{\partial m_i}{\partial Q_{in, N}} \delta Q_{in, N} \right)^2 + \left(\frac{\partial m_i}{\partial A_{in, N}} \delta A_{in, N} \right)^2 \right]^{1/2}$$

$$\delta m_{in, N} = [(0.002)^2 + (0.004)^2 + (0.000)^2 + (0.001)^2]^{0.5}$$

$$\delta m_{in, N} = 0.005 \text{ lb/s}$$

Thus, there was a 95% confidence that the reported centre nozzle inlet mass flow, m_{in} for the design pressure ratio ($PR = 2.3$) and the assumptions stated previously was as follows:

$$m_{in} = 1.777 \pm 0.005 \text{ lb/s}$$

B.5.3 Nth-order Yaw Angle Uncertainty

The uncertainty in the indicated value of the Yaw angle was determined at the design pressure ratio, $PR = 2.30$. The uncertainty was determined as follows. Firstly, the uncertainty arising from the differential pressure transducer which sensed ΔP_{swp} was determined and secondly, the uncertainty associated with the yaw actuator of the L.C. Smith traverse gear was determined. Total uncertainty in the indicated value of the yaw angle was then computed as the RSS of the uncertainties associated with the pressure transducer and L.C. Smith traverse gear.

Calculation of the uncertainty in the Yaw angle arising from the differential pressure transducer was calculated as follows:

$$\delta\Delta P_{\text{swp}} = \text{RSS} [\delta\Delta P_{\text{swp},1}, \delta\Delta P_{\text{swp},0}, \delta\Delta P_{\text{swp, fixed}}]$$

$$\delta\Delta P_{\text{swp}} = [0.248^2 + 0.000^2 + (0.016 * 0.0077)^2]^{0.5}$$

$$\delta\Delta P_{\text{swp}} = 0.248 \text{ psi}$$

From the wedge probe calibration as discussed in Appendix A the differential wedge probe static pressure for a 10 deg Yaw angle was 5.0 psi. Thus, a factor of 0.5 psi/degree allows conversion to degrees of the uncertainty in Yaw as measured in psi.

$$\delta\text{Yaw}_{\text{pswp}} = 0.248 / 0.5 \text{ degrees}$$

$$\delta\text{Yaw}_{\text{pswp}} = 0.496 \text{ degrees}$$

Calculation of the uncertainty in the Yaw angle arising from the L.C. Smith (LCS) traverse gear was calculated as follows:

$$\begin{aligned} \delta\text{Yaw}_{\text{LCS}} &= \text{RSS}[\delta\text{Yaw}_{\text{LCS},1}, \delta\text{Yaw}_{\text{LCS},0}, \delta\text{Yaw}_{\text{LCS, fixed}}] \\ &= 0.357 \text{ degrees} \end{aligned}$$

The Nth-order uncertainty in Yaw angle was then

$$\delta\text{Yaw}_{\text{LCS}} = [(0.496)^2 + (0.357)^2]^{0.5}$$

$$\delta\text{Yaw}_{\text{LCS}} = 0.611 \text{ degrees}$$

Thus, there was 95% confidence that the indicated Yaw angle for an operating PR = 2.3 was accurate to within ± 0.611 degrees.

B.5.5 Nth-Order ΔT_o Uncertainty

Uncertainties in ΔT_o associated with L.C. Smith positional errors were determined at the design pressure ratio, $PR = 2.3$. Pitchwise and spanwise ΔT_o gradients ($^{\circ}F/in$) were measured across and along the airfoil wake respectively. These temperature gradients were multiplied by the positional uncertainties as shown in Table B.1 to yield fixed uncertainties in ΔT_o associated with pitchwise and spanwise movement of the L.C. Smith traverse gear. In the case of Yaw, the positional uncertainty in Yaw (see Table B.1) was combined via RSS with the 1st-order uncertainty in Yaw (see Table B.4) and then multiplied by the pitchwise T_o gradient to yield the uncertainty in ΔT_o associated with uncertainties in Yaw. Table B.5 lists these fixed uncertainties.

Calibration of the wedge probe and its associated thermocouple are discussed in Appendix A. Accuracy of the calibration (95% confidence level) was $\pm 0.018^{\circ}F$.

Thus, the Nth-order uncertainty, $\delta\Delta T_{o,N}$ associated with the measurement of ΔT_o

	ΔT_o Gradient	Positional Uncertainty	Fixed ΔT_o Uncertainty
Pitch	20.0 F/in	± 0.021 in	± 0.420 F
Span	4.0 F/in	± 0.017 in	± 0.068 F
Yaw	20.0 F/in	± 0.362 deg	± 0.012 F

Table B.5: Fixed ΔT_o uncertainties associated with L.C. Smith positional error.

was as follows:

$$\delta \Delta T_{o,N} = \pm 1.92 \text{ F}$$

Thus, the reported values of ΔT_o are accurate to within ± 1.92 °F (± 1.07 °C) of the actual values 95% of the time.

B.5.5 Nth-Order M_s Uncertainty

Nth-order uncertainty in the calculated M_s was dependent on Nth order uncertainty in both P_{owp} and the local downstream static pressure, P_s , which for the purpose of this uncertainty analysis was represented by the uncertainty in P_{sDR10} (i.e., $\delta P_s = \delta P_{sDR10}$).

As previously mentioned all pressures were referenced to SPIR, a static pressure port located on the right hand endwall at midchannel of the centre nozzle passage and two axial chord lengths upstream of the airfoil row. Thus,

uncertainty in all pressure readings was dependent on uncertainties in Q_{in} , which related SPIR to the barometric total pressure, P_{atm} . P_{atm} was set equal to 14.7 psia for the purpose of this uncertainty analysis. Thus

$$\delta P_{owp,n} = 0.073 \text{ psi}$$

Thus, there was a 95% confidence that the reported P_{owp} pressure for an operating PR = 2.3 was as follows:

$$P_{owp} = 14.201 \pm 0.073 \text{ psia}$$

Similarly

$$\delta P_{s,n} = 0.063 \text{ psi}$$

Thus, there was a 95% confidence that the reported P_s pressure for an operating PR = 2.3 was as follows:

$$P_s = 6.495 \pm 0.063 \text{ psi}$$

Uncertainty in the isentropic M_s for the design pressure ratio was determined by the method of sequential perturbation which was carried out on the equation relating P_{owp} and P_s to the M_s . The uncertainties in M_s for the design pressure ratio (PR = 2.3) resultant from Nth-order uncertainties in P_{owp} and P_s are as follows:

$$\frac{\partial M_s}{\partial P_{owp,N}} \delta P_{owp,N} = 0.004$$

$$\frac{\partial M_s}{\partial P_{s,N}} \delta P_{s,N} = 0.008$$

Thus the Nth-order uncertainty in M_a was as follows:

$$\delta M_{a,N} = 0.009$$

Thus, there was 95% confidence that the reported isentropic Mach number for the operating PR = 2.3 was as follows:

$$M_a = 1.159 \pm 0.009$$

Similarly for total pressure loss coefficient, since

$$C_{p_o} = \frac{P_{o1} - P_{o2}}{P_{o1} - P_{s_{ave}}}$$

$$\delta C_{p_o,N} = \pm 0.018$$

However, this is the uncertainty for one data point. The uncertainty for the passage and midspan pressure loss may be obtained using the following relation:

$$\delta C_{p_o} = \left(\frac{\delta C_{p_o}^2}{n} \right)^{\frac{1}{2}}$$

Which for $n = 252$ (a 28 by 9 grid (half passage)) is equal to ± 0.001 .

The uncertainty for pressure losses are summarized below:

$$\delta \bar{\bar{C}}_{p_{\text{mass-ave}}} = \pm 0.002$$

$$\delta \bar{\bar{C}}_{p_{\text{area-ave}}} = \pm 0.001$$

$$\delta \bar{C}_{p_{\text{mass-ave}}} = \pm 0.006$$

$$\delta \bar{C}_{p_{\text{area-ave}}} = \pm 0.003$$

B.6 Discussion

The above presentation of the various uncertainties associated with the measurements is mathematically correct and the uncertainty analysis does instill confidence in the results which will be presented and discussed in the following chapter. This research study has been carried out knowing the measurement of compressible flow parameters is fraught with difficulties and uncertainties. Within the limits of available NRC facilities and time constraints detailed calibrations and uncertainty analysis have been carried out on all transducers, wedge probes and the planar cascade. These studies have not been definitive, items such as effects of pitch or immersion depth on wedge probe measurements could not be investigated. Assumptions have been made regarding the flow through the supposedly normal shock waves in front of the wedge probes; isentropic relationship have been used to analyze these flows. Without a doubt, more questions regarding uncertainties have arisen in attempting to answer specific questions on uncertainty, and the limitations with present measurement techniques have been revealed.

Appendix C

- Part a) Airfoil surface Mach number distribution for the baseline airfoil.**
- Part b) Pitchwise averaged total pressure loss coefficients.**
- Part c) Pitchwise averaged deviation angles.**

**Part a) Airfoil surface Mach number distribution for the
baseline airfoil.**

*Thin boundary layer
 **Thick boundary layer

Part b) Pitchwise averaged total pressure loss coefficients.

Pitchwise Area Averaged Pressure Loss Coefficient

$$t_{\max}/c = 0.15$$

Pitchwise Area Averaged Pressure Loss Coefficient

$$t_{\max}/c = 0.266$$

Pitchwise Area Averaged Pressure Loss Coefficient

$$t_{\max}/c = 0.482$$

Pitchwise Area Averaged Pressure Loss Coefficient

$$t_{\max}/c = 0.266$$

Pitchwise Area Averaged Pressure Loss Coefficient

$$t_{\max}/c = 0.15$$

Pitchwise Area Averaged Pressure Loss Coefficient

$$t_{\max}/c = 0.482$$

Part c) Pitchwise averaged deviation angles.

Pitchwise Area Averaged Exit Deviation Angle, δ

Pitchwise Area Averaged Exit Deviation Angle, δ

Pitchwise Area Averaged Exit Deviation Angle, δ

Pitchwise Area Averaged Exit Deviation Angle, δ

3 OF/DE 3

PM-1 3 1/2" x 4" PHOTOGRAPHIC MICROCOPY TARGET
NBS 1010a ANSI/ISO #2 EQUIVALENT

Pitchwise Area Averaged Exit Deviation Angle, δ

Pitchwise Area Averaged Exit Deviation Angle, δ

Appendix D

Data Reduction Program


```

PROGRAM THEWORKS
C .....
C THIS PROGRAM CONVERTS DATA FROM PLANAR CASCADE FILES DWP
C AND DSS INTO COMPUTABLE MATRICES. IT REDUCES THE MATRICES
C AND EVALUATES THE VOLUME UNDER THE CURVE PRODUCED BY THE
C RESULTS.
C .....
C RAMA RADNARD MAY 1991. NRC PWC
C MANFRED KLEIN JUNE 1990. WEC NRC
C PATRIC OUELLETTE MAY 1988. WEC NRC
C .....
COMMON /DATA/ R,T,YAW,CPO,MA,MDOOT,DTO,STA,DWP,DSS,WALL,PS
* .APS,MAVPR,SUMMASS,SUMCPO,SUMDTO,SUMMA,SUMYAW
* .FIRST,LAST,TOTAL,TOTALPS,CONST,RHO2,VEL2,PM,
* QM
C
REAL DWPI(15,30,10),DSSI(71,3),CPOIO(15,0,30),MAIO(15,0,30),
* MDOOTIO(15,0,30),DTIOIO(15,0,30),RIO(15,0,30),TIO(15,0,30),
* YAWIO(15,0,30),STAIIO(15,0,30),WALL PS(871,PS(87,2))
* .APS,MAVPR,SUMMASS,SUMCPO,SUMDTO,SUMMA,SUMYAW,
* VEL2IO(15,0,30),RHO2IO(15,0,30)
C
INTEGER NEWR,NEWT,FIRSTIO(20),LASTIO(20),OR
C
CHARACTER*20 WPHNAME,DSSHNAME,WPNAME,DSSNAME
CHARACTER FLEROOT*8,CR*1
C
READ THE INPUT FILE NAMES
WRITEI*,*)'PLEASE INPUT THE WEDGE PROBE FILE NAME'
READI*,2)WPNAME
C
DATEFLAG IS USED TO CORRECT THE WP PO VALUES FOR MEASUREMENTS
PERFORMED BEFORE DAY 76 OF 1989
C
WRITEI*,*)'PLEASE INPUT THE DOWNSTREAM STATICS FILE NAME'
READI*,2)DSSNAME
2 FORMAT(A20)
C
WRITEI*,*)'PLEASE INPUT THE OUTPUT FILE NAME (NO EXTENSION)'
READI*,3)FLEROOT
3 FORMAT(I8)
C
CALL THE SUBROUTINE TO READ IN THE DWP DATA FILE
C
WRITEI*,*)'CONVERTING DWP FILE'
CALL DWPCONVERT(WPNAME,NEWR,NEWT,WPHPNAME)
C
CALL THE SUBROUTINE TO READ IN THE DSS DATA FILE
C

```

```

C
WRITE(*,*)'CONVERTING DSS FILE'
CALL DSSCONVERT(DSSNAME,DSSHNAME)
C
CALL THE DATA REDUCTION SUBROUTINE
C
OR = 0
776 WRITE(*,*)'REDUCING DATA'
CALL DATAREDUCTION(NEWR,NEWT,FLEROOT,OR)
C
CALL THE MASS-AVERAGING SUBROUTINE
WRITE(*,*)'MASS-AVERAGING'
CALL MAVERAGE(NEWR,NEWT,FLEROOT)
READI*,737) CR
737 FORMAT(I1)
IF (CR.EQ.'R') THEN
OR = OR + 1
GOTO 776
ELSE
GOTO 777
END IF
C
CALL THE PITCHWISE MASS-AVERAGING SUBROUTINE
C 777 WRITE(*,*)'PITCHWISE MASS-AVERAGING'
CALL PWMMA(NEWR,NEWT,FLEROOT)
C
CALL THE MATRIX SUBROUTINE
C
WRITE(*,*)'MATRIX PRODUCTION'
CALL MATRIX(NEWR,NEWT,FLEROOT)
C
CALL THE AREA AVERAGING SUBROUTINE
777 WRITE(*,*)'AREA AVERAGING'
CALL AWMMA(NEWR,FLEROOT,WPHPNAME)
C
WRITE(*,*)'PITCHWISE MASS-AVERAGING'
CALL PWMMA(NEWR,NEWT,FLEROOT)
C
WRITE(*,*)'MATRIX REDUCTION'
CALL MATRIX(NEWR,NEWT,FLEROOT)
C
WRITE(*,*)'THATS ALL FOLKS'
END
C
***** THIS SUBROUTINE CONVERTS THE SELECTED
C ***** DWP FILE *****
C
SUBROUTINE DWPCONVERT(WPNAME,NEWR,NEWT,WPHPNAME)
C

```



```

C INITIALIZE THE DSS MATRIX
DO 170 I = 1, 71
DO 180 J = 1, 3
DSS(I, J) = 0
180 CONTINUE
170 CONTINUE
C
C OPEN FILE DSS AND READ INFORMATION
C
OPEN(4, FILE = DSSNAME, STATUS = 'OLD')
READ(4, 100) DSSHPNAME
100 FORMAT(A20)
WRITE(*, *) 'CONVERTING FILE : ' DSSHPNAME
READ(4, 110) P2(1), P2(2), P2(3), P2(4), P2(5), P2(6), P2(7), P2(8)
110 FORMAT(//I1, F6.3, T9, F6.3, T18, F6.3, T25, F6.3, T33, F6.3,
* T44, F7.0, T55, F8.4, T66, F6.1)
C
C FILLING DSS MATRIX WITH DATA FROM DSS FILE
K = 1
DO 375 I = 1, 3
DO 400 J = 1, 3
DSS(I, J) = P2(K)
K = K + 1
400 CONTINUE
375 CONTINUE
C
READ(4, 120)
120 FORMAT(I)
C
IND = 4
DO 500 IND = 4, 71
READ(4, 130) END = 1000, DAT2(1), DAT2(2), DAT2(3)
130 FORMAT(I2, F9.6, T18, F8.6, T32, F8.6)
DO 500 J = 1, 3
DSS(IND, J) = DAT2(J)
500 CONTINUE
1000 DSS(3, 3) = IND - 4
C
C END SUBROUTINE
WRITE(*, *) 'DSS FILE CONVERSION TERMINATED'
CLOSE(4)
END
C
C ***** DATA-REDUCTION SUBROUTINE *****
C
SUBROUTINE DATAREDUCTION(R, NT, INFILE, QR)
COMMON /DATA/ R, T, YAW, CPO, MA, MDO, DTO, STA, DWP, DSS, WALL, PS
APS, MAVPR, SUMMASS, SUMPCO, SUMDTO, SUMMA, SUMYAW,
* FIRST, LAST, PTOTAL, TTOTAL, PS, CONST, RHO2, VEL2, QW
* QM
C
REAL DWP(15, 30, 10), DSS(71, 3), CPO(10, 15, 3, 30), MA(10, 15, 0, 30),
* MDO(10, 15, 0, 30), DTO(10, 15, 0, 30), R(10, 15, 0, 30), T(10, 15, 0, 30),
* YAW(10, 15, 0, 30), STA(10, 15, 0, 30), PS(67, 2), WP, PO, WALL, PS(67)
* V1, V2, V3, V4, V5, V6, V7, V8, K, CP, MV, PSX, POX, NMX, WP, TO, ALPHA
* TO, DELTA, T, TSX, RHOX, V, X, MD, DHO, CX, CONVERT, MDO, A,
* DHO, A, SLOPE, PSA, WP, CONST, PST, RHO2(10, 15, 0, 30)
* APS, MAVPR, SUMMASS, SUMPCO, SUMDTO, SUMMA, SUMYAW,
* VEL2(10, 15, 0, 30)
C
INTEGER CR, CT, I, J, NR, NT, FINAL, FIRST(10, 20), LAST(10, 20), QR
C
CHARACTER INFILE*8, OUTFILE*16
C
WRITE(*, *) 'ENTER THE YAW OFFSET'
READ(*, *) CONST
C
CP = .24
K = 1, 4
V1 = (K - 1)/K
V2 = 2/(K - 1)
V3 = (K + 1)/2
V4 = (K - 1)/2
V5 = K/(K - 1)
V6 = 2*K/(K + 1)
V7 = (K - 1)/(K + 1)
V8 = 1/(K - 1)
C
C LOAD THE STATIC TAP LOCATIONS
C
OPEN(4, FILE = 'TAPPOS2.DAT', STATUS = 'OLD')
DO 102 I = 1, 67
READ(4, 98) PS(I, 1)
102 CONTINUE
CLOSE(4)
98 FORMAT(I10, 5)
C
C INITIALIZE THE OUTPUT MATRICES
C
DO 190 I = 0, 15
DO 192 J = 0, 30
R(I, J) = 0
T(I, J) = 0
YAW(I, J) = 0
CPO(I, J) = 0
MA(I, J) = 0
MDO(I, J) = 0

```

```

      DTO(I,J)=0
      STAI(J)=0
192      CONTINUE
190      CONTINUE
C
C      FINAL = INT(DSS/3.31)
C
C      CALCULATE THE STATIC PRESSURE (PSIA)
      DO 230 I = 1, FINAL
          PSI(I,2) = DSS(I,1)/DSS(I,3.1)
230      CONTINUE
C
      APS = 0.0
      DO 300 I = 25.51
          APS = APS + (PSI(I,2) + PSI(I,1,2))/2 * (PSI(I,1,1) + PSI(I,1,1))
300      CONTINUE
C
      APS = APS/4.455
C
C      PROCESS WEDGE PROBE DATA
      DO 240 CR = 1, NR
          DO 250 CT = 2, NT + 1
              C --- MEASURED TOTAL PRESSURE: WP_PO = PATM + WP_PO (PSIA)
              C
              WP_PO = DWPICR(1,1) + DWPICR(CT,4)
              WRITE(*,1) DWPICR(1,1)
              C
              C      WALL STATIC PRESSURE MEASUREMENTS ARE USED TO COMPUTE THE
              C      WEDGE PROBE MACH NUMBERS
              C
              I = 1
              C
              C      INTERPOLATE THE WALL STATIC PRESSURES
              C
              IF (PSI(I,1) LE DWPICR(CT,2)) AND
                  (PSI(I,1) GE DWPICR(CT,2)) THEN
                  SLOPE = (S(I,2)-PSI(I,1,2))/(PSI(I,1)-PSI(I,1,1))
                  WP = PS(CT,1) + SLOPE*(DWPICR(CT,2)-PSI(I,1)) + PSI(I,2)
                  PSX = WALL_PS(CT,1)
                  GOTO 500
              END IF
              C
              C
              GOTO 400
              C
              C      CALCULATE THE MACH NUMBER AT THE WEDGE PROBE
              C      USING P0 WEDGE PROBE AND PS WALL.
              C
              S20 MACH = 1/2 * (WP_PO/PSX)**(1.4188 + 5
                  PSX = WP_PO

```

```

      IF (MACH GT .1) THEN
          POX = WP_PO/(V3*MACH**2*(1 + V4*MACH**2))**V5/(V6*MACH**2-V7)**V8)
          NMIX = (V2*(POX/PSX)**V1-1)**.5
          C
          IF (ABS(NMIX-MIX) LT 0.00001) GOTO 504
          MIX = NMIX
          GOTO 502
          END IF
          C
          C --- CALCULATE THE WEDGE PROBE STATIC PRESSURE
          C
          WP_PS = PATM/WP_PS + WP_DPS/2 (PSIA) FOR COMPARISON WITH
          C      THE WALL STATIC PRESSURE.
          C
          504      WP_PS = DWPICR(1,1)/DWPICR(CT,5) + DWPICR(CT,6)/2.0
          C
          PSX_WP = 1/((1.00016 - 1.77456*MIX + 1.38142*MIX**2 - 4.11125*MIX**3
              + 4.72450*MIX**4 - 1.57824*MIX**5)/WP_PS)
          C --- CORRECTION FOR DELTA_T0 FOR DATA BEFORE JULY 4TH
          IF (REQ.0) THEN
              EMF = 2.08611 + (DWPICR(CT,10)/1.8)*39.4304
              EMF = -EMF
              DWPICR(CT,10) = (1.0578059 + .0253828*EMF)*1.8
              ELSE
                  GOTO 749
              END IF
          C --- CORRECTION FOR WP_TO USING CALIBRATION FACTOR ALPHA
          C
          749      WP_TO = DWPICR(CT,9) + DWPICR(CT,10)
              ALPHA = .7378 + .1461*MIX**2 - .03935*MIX**3
              TO = WP_TO/ALPHA*(V4*MACH**2 + 1)*(V4*MACH**2 + 1)
              DELTA_T = TO-DWPICR(CT,9)
          C --- CALCULATE TSX,RHOX
              TSX = TO/(1 + V4*MACH**2)
              RHOX = PSX*144/(53.34*TSX)
          C --- CALCULATE VX (FT/SEC) NORMAL TO TRAVERSE PLANE
              CX = (32.1741*1.4*53.34*TSX)**.5
              VX = CX*MIX
              CONVERT = (DWPICR(CT,3)/DWPICR(
                  CT,6)*55) - CONST)*3.141592654/180
          C --- CALCULATE MASS FLOW PER UNIT OF AREA (LBW/SEC-FT^2)
              MD = RHOX*VX
          C --- CALCULATE CHANGE IN TOTAL ENTHALPY (BTU/SEC-FT^2)
              DHO = CP*DELTA_T*MD
          C --- CALCULATE MDOOT AND DHO NORMAL TO TRAVERSE PLANE
              MDOOT_A = MD*508/CONVERT
              DHO_A = DHO*508/CONVERT
          C --- STORE VALUES IN MATRICES IN OUTPUT MATRICES
          C
          C      RADIAL POSITION WITH CORRECTION FOR P0 OFFSET IN INCHES.
          C
              RCR,CX(I) = DWPICR(CT,1) - 0.164

```

```

C      THETA POSITION (INCHES)
C      T1CR,CT-1) = DWPCICR,CT,2)
C      EXIT FLOW ANGLE (DEGREES)
C      YAWICR,CT-1) = DWPCICR,CT,3) / DWPCICR,CT,8) * 0.55) - CONST
C      TOTAL PRESSURE LOSS COEFFICIENT (Po1 Po2) / (Po1 - Po2)
C      IF (NX GE 0.58) AND (MX LE 1.5) THEN
C      PST = (DWPCICR,1) - DWPCICR,CT,5) * (1 - 15224 * MX ** 2
C      - 1.28593 * MX + 365447) / (1
C      ELSE
C      PST = DWPCICR,1) - DWPCICR,CT,5)
C      END IF
C      CPOICR,CT-1) = (DWPCICR,1) / (POX) / (DWPCICR,1,1) - PST)
C      EXIT MACH NUMBER
C      MAICR,CT-1) = MX
C      EXIT MASS FLOW PER UNIT AREA NORMAL TO TRAVERSE PLANE
C      MDOTICR,CT-1) = MDOT_A
C      TOTAL TEMPERATURE CHANGE (To2-To1, DEGREES RANKIN)
C      DTOICR,CT-1) = TO-DWPCICR,CT,9)
C      EXIT STATIC PRESSURE DIFFERENCE, WALL - WEDGE PROBE
C      STAICR,CT-1) = (PSX-PSX_WPI) / (DWPCICR,1,1) - APS)
C      STAICR,CT-1) = POX
C      EXIT PLANE DENSITY
C      RHO2ICR,CT-1) = RHOX
C      EXIT PLANE VELOCITY
C      VEL2ICR,CT-1) = VX * COS(CONVERT)
C
C
C      250 CONTINUE
C      240 CONTINUE
C      WRITE(*,*) 'DATA REDUCTION COMPLETED'
C
C      WRITE THE DATA TO THE OUTPUT FILE
C
C      OUTFILE = 'RED' // INFILE // '.DAT'
C      WRITE(*,*) INFILE,OUTFILE
C      WRITE(*,*) 'WRITING DATA TO FILE:',OUTFILE
C      OPENIS,FILE = OUTFILE,STATUS = 'UNKNOWN')
C
C      WRITE NUMBER OF RADIAL AND THETA
C
C      WRITEIS,806INR,NT
C      FORMAT(I2,2X,I2)
C
C      WRITE THE REDUCED DATA TO THE OUTPUT FILE
C
C      WRITEIS,807/RADIAL THETA YAW Cpo Ms Mdot DTo
C      * Cps'

```

```

807 FORMAT(I62)
DO 611, I = 1, NR
DO 612, J = 1, NT
WRITEIS,6101 RII(J), TII(J), YAWII(J), CPOII(J), MAII(J), MDOTII(J),
, DTOII(J), STAI(J)
810 FORMAT(BI7.3,1X)
612 CONTINUE
611 CONTINUE
CLOSE(S)
C      RETURN
C      END
C      *****MASS-AVERAGING SUBROUTINE *****
C      SUBROUTINE MAVERAGEINR,NT,(INFILE)
C      *****
C      This subroutine calculates the mass-averaged flow parameters
C      from the wedge probe traverse data downstream of the airfoil row.
C      The boundaries are one airfoil-to-airfoil distance and one
C      half of the span beginning at the right endwall.
C      Manfred Klein 1990, WEC/NRC
C      *****
C      COMMON /DATA/ R,T,YAW,CPO,MA,MDOT,DTO,STA,DWP,DSS,WALL,PS
C      + ,APS,MAVPR,SUMMASS,SUMCPO,SUMDTO,SUMMA,SUMYAW
C      + ,FIRST,LAST,PTOTAL,TTOTAL,PS,CONST,RHO2,VEL2,PM,
C      + QM
C      INTEGER I,J,FIRSTIO:201,LASTIO:201
C
C      REAL RIO:15,0:301,TIO:15,0:301,YAWIO:15,0:301,CPOIO:15,0:301,
C      + MAIO:15,0:301,MDOTIO:15,0:301,DTOIO:15,0:301,FRAC,ENDTHETA,
C      + HMDOT,BASEAREA,SUMMASS,V1MDOT,V2MDOT,V1CPO,V2CPO,V1MA,V2MA,
C      + V1YAW,V2YAW,V1DTO,V2DTC,HCP0,HMA,HYAW,MDTO,SUMCPO,SUMMA,
C      + SUMYAW,SUMDTO,STAIO:15,0:301,CPOIDIO:301,DTOMIDIO:301,YAWMIDIO:301,
C      + DWPI:15,30,101,DSSI:71,31,MAVPR,WALL_PSI(67),PSI(67,2)
C      + ,APS,RHO2IO:15,0:301,VEL2IO:15,0:301)
C
C      CHARACTER INFILE*8,OUTFILE(3)*18
C      OUTFILE = 'MAV' // INFILE // '.DAT'
C
C      Find a one pitch long section of the data file for integrating.
C      Find the initial THETA value located at 8.75 < THETA < 9.125 in each

```

```

C      block of RADIAL = constant data. Once the initial value is located,
C      calculate the final value located one pitch downstream by
C      interpolating. Save the limits of integration.
C
min = 11.75
max = 12.125
DO 140, I = 1, NR
DO 150, J = 1, NT
IF (T(I,J),LT,min) GOTO 150
IF (T(I,J),GE,max).AND.(T(I,J),LE,max)) THEN
FIRST(I) = J
ENDTHETA = T(I,J) + 4.455
ENDIF
IF (ENDTHETA,GT,T(I,J-1)).AND.
(ENOTTHETA,LT,T(I,J)) THEN
LAST(I) = J
FRAC = (ENDTHETA-T(I,J-1))/(T(I,J)-
T(I,J-1))
YAW(I,J) = YAW(I,J-1) + (YAW(I,J)-YAW(I,J-1))
*FRAC
CPOI(I,J) = CPOI(I,J-1) + (CPOI(I,J)-CPOI(I,J-1))
*FRAC
MDOT(I,J) = MDOT(I,J-1) + (MDOT(I,J)-
MDOT(I,J-1)) *FRAC
DTON(I,J) = DTON(I,J-1) + (DTON(I,J)-DTON(I,J-1))
*FRAC
TRI(J) = T(I,J-1) + (T(I,J)-T(I,J-1)) *FRAC
STAI(I,J) = STAI(I,J-1) + (STAI(I,J)-STAI(I,J-1))
*FRAC
C
ENDIF
150 CONTINUE
140 CONTINUE
C      Create the wall values at R = 0
DO 170, J = 1, NT
R(0,J) = 0
T(0,J) = 0
YAW(0,J) = YAW(1,J)
CPOI(0,J) = CPOI(1,J)
DWP(0,J) = DWP(1,J)
WRITE(6,DWP(0,J)) WALL,PS(0,J) CPOI(0,J)
WAC(0,J) = 0
WCO(0,J) = 0
END DO
C      DTOT(I,J) = DTON(I,J) + (DTON(I,J)-DTON(I,J-1)) *FRAC
C      DTOMID(I) = DTON(I,J)
C      TNR(J) = TNR(1,J) + (TNR(J)-TNR(1,J)) *FRAC
C      STAINR(J) = STAINR(1,J) + (STAINR(J)-STAINR(1,J)) *FRAC
C      RNR(J) = 2.22
C      CONTINUE
C      WRITE(6,*) 'ONE PITCH HALF SPAN DATA REDUCTION COMPLETED'
C      WRITE THE DATA TO THE OUTPUT FILE
C
OUTFILE(1) = MATZ/INFILE/"/ DAT'
OUTFILE(2) = 'MID"/INFILE/"/ DAT'
WRITE(6,*) INFILE,OUTFILE(1)
WRITE(6,*) 'WRITING DATA TO FILE : ',OUTFILE(1)
OPEN(5,FILE = OUTFILE(1),STATUS = 'UNKNOWN')
OPEN(6,FILE = OUTFILE(2),STATUS = 'UNKNOWN')
C      WRITE NUMBER OF RADIAL AND THETA
C
DO 1717, I = FIRSTNR, LASTNR
WRITE(6,1608) INP, J, CPOI(I,J)
1608 FORMAT('X,2F,3
1717 CONTINUE
WRITE(5,1608 NR,N'

```

```

1606      FORMAT(12,2X,12)
C
C      WRITE THE REDUCED DATA TO THE OUTPUT FILE
C
      WRITE(5,1607), RADIAL, THETA, YAW, CPO, MS, Mdot, DTo
      Cps
1607      FORMAT(A82)
      DO 1611, I = 0, NP
          DO 1612, J = FIRST(I), LAST(I)
              WRITE(5,1610) R(I), T(I), J, YAW(I, J), CPO(I, J), MA(I, J), MDOT(I, J)
              , DTO(I, J), STA(I, J)
          C
1610      FORMAT(B17.3, 1X)
1611      CONTINUE
C
C      WRITE the data from the file
C
      OUTFILE(3) = 'MAV'//INFILE//'.DAT'
      OPEN(4, FILE = OUTFILE(3), STATUS = 'UNKNOWN')

      Integrate over the area to find the massflow.
      Integration routine calculates the area under the specific
      massflow surface as a function of radial and theta.

      The integration routine sums the volumes of three sided
      prisms. The routine is the three dimensional version of
      a trapezoid rule.
      HEIGHT - determines the mean height of the triangular prism
      BASEAREA - determines the normal area of the triangular prism
                  and is divided by 144 to convert square inches to
                  square feet.

      SUMMASS = 0.0
      SUMCPO = 0.0
      SUMYAW = 0.0
      SUMMA = 0.0
      SUMDTo = 0.0
      SUMSTA = 0.0

      DO 90, I = 0, NP-1
          DO 100, J = FIRST(I), LAST(I)-1

              HMDOT = (MDOT(I, J) + MDOT(I+1, J) + MDOT(I, J+1))/3
              HCPO = (MDOT(I, J)*CPO(I, J) + MDOT(I+1, J)*CPO(I+1, J) +
                  MDOT(I, J+1)*CPO(I, J+1))/3
              HYAW = (MDOT(I, J)*YAW(I, J) + MDOT(I+1, J)*YAW(I+1, J) +
                  MDOT(I, J+1)*YAW(I, J+1))/3
              HMA = (MDOT(I, J)*MA(I, J) + MDOT(I+1, J)*MA(I+1, J) +
                  MDOT(I, J+1)*MA(I, J+1))/3
              HOTO = (MDOT(I, J)*OTO(I, J) + MDOT(I+1, J)*OTO(I+1, J) +
                  MDOT(I, J+1)*OTO(I, J+1))/3
              HSTA = (MDOT(I, J)*STA(I, J) + MDOT(I+1, J)*STA(I+1, J) +
                  MDOT(I, J+1)*STA(I, J+1))/3
              BASEAREA = (RI(I, J) + RI(I, J+1) + RI(I+1, J) + 1)*T(I+1, J)/288

              VZMDOOT = HMDOT*BASEAREA
              VZCPO = HCPO*BASEAREA
              VZYAW = HYAW*BASEAREA
              VZMA = HMA*BASEAREA
              VZDTo = HOTO*BASEAREA
              VZSTA = HSTA*BASEAREA

              SUMMASS = SUMMASS + VIMDOT + VIMDOT + VZMDOOT
              SUMCPO = SUMCPO + V1CPO + V2CPO
              SUMYAW = SUMYAW + V1YAW + V2YAW
              SUMMA = SUMMA + V1MA + V2MA
              SUMDTo = SUMDTo + V1DTo + V2DTo
              SUMSTA = SUMSTA + V1STA + V2STA

          C
          C      CONTINUE
          C
          SUMCPO = SUMCPO/SUMMASS
          SUMYAW = SUMYAW/SUMMASS
          SUMMA = SUMMA/SUMMASS
          SUMDTo = SUMDTo/SUMMASS
          SUMSTA = SUMSTA/SUMMASS

          Rate of exit to inlet massflow
          SUMMASS = SUMMASS*2.0/DWPR(1,1,81)*0.981

```

```
C
SUMCPO = SUMCPO
MAVPR = DWP(1,1,9)/APS

C
WRITE(*,' ') SUMMASS
C 201 FORMAT(' ', PR MASS FLOW CPO DTG)
C $MACH YAW STA')
WRITE(A,200) MAVPR, SUMMASS, SUMCPO, SUMDTCO, SUMMA, SUMYAW, SUMSTA
FORMAT(7(F10.4))
C
CLOSE(AI)
END
SUBROUTINE PWRMAIN(NR, NT, INFILE)
C.....
C
C This subroutine calculates the mass-averaged total pressure loss
along a RADIAL - constant strip. Thus, it integrates along
THETA assuming a RADIAL strip of unit width.
C
C Mantred Klein 1980, WEC/NRC
C.....
COMMON /DATA, R, T, YAW, CPO, MA, MDOT, DTG, STA, DWP, DSS, WALL, PS
APR, MAYPR, SUMMASS, SUMCPO, SUMDTCO, SUMMA, SUMYAW
FIRST, LAST, PTOTAL, TTOTAL, PS, CONST, RHO2, VEL2 PM
QM
C
INTEGER I, J, K, FIRST(I=20), LAST(I=20), NR, NT
C
REAL CPO(I=20), MDOT(P=20), MIXLOS(I=20),
SUMYAW, SUMMA, SUMDTCO, YAW(P=20), MAP(I=20), DTGP(I=20), DUMMY1,
DWP(15,30), DSSI(7,3), CPO(I=15,0:30), DUMMY2,
NAIO(15,0:30), MDOT(I=15,0:30), DTGO(I=15,0:30), RIO(15,0:30),
TIO(15,0:30), YAW(I=15,0:30), STAO(I=15,0:30), WALL PSI(67)
APR, MAYPR, SUMMASS, SUMCPO, PSI(67,2), RHO2(I=15,0:30),
VEL2(I=15,0:30) PM, QM,
GAMMA
C
CHARACTER INFILE *B, OUTFILE(5)=16
C
C Create the output file names
C
OUTFILE1 = PNY INFILE PNY
OUTFILE2 = PMP INFILE PMP
OUTFILE3 = PNM INFILE PNM
OUTFILE4 = PNT INFILE PNT
OUTFILE5 = PNS INFILE PNS
```



```

DO 721 I = 25,51
  721   PSA = PSA + PSI(I,2)
      PSA = PSA,26
      PM = PSA*144
      PM = PSA
C
WRITE(*,'ITEMP2G,TEMP2H,TEMP2F,PM,TEMP2E,TEMP2C
147 Y=(TEMP2E+PSA)*(PM**0.71423)-PM**1.71423-TEMP2C
YPRIME=0.71423*(TEMP2E+PSA)*(PM**(-0.28571))-
    1.71423*(PM**0.71423))
NPM=PM-(Y/YPRIME)
DIFF=ABS(PM-NPM)
IF(DIFF GE 0.01) THEN
  WRITE(*,'PM,NPM,DIFF
PM=NPM
GOTO 147
ELSE
VM=TEMP2F/(PM**(1/1.714)+0.742)/(TOTAL/(PTOTAL**144)*PM)
**0.28571)))
ENDIF
C
TM=TTOTAL/(PTOTAL**144)*PM**0.28571
CM=(1/4**53)**32/174**TM**0.5
TEMPCM=CM*COS(TEMP2B**3.141592654/180)
MA2=VM/TEMPCM
PTOTAL2=PM*(1+0.2*MA2**2)**0.3/5
YM=(PTOTAL**144)/PTOTAL2/(PTOTAL**144)*PM)
WRITE(*,'IPTOTAL,TM,CM,MA2,PTOTAL2,TEMP2B,TEMPCM
WRITE(*,'YM,TEMP2G,TEMP2H,TEMP2F,PM,VM,TEMP2E,TEMP2C
C
Open the output file and write the results
C
OPEN(FILE='OUTFILE()',STATUS='UNKNOWN')
WRITE(FILE,'INPHNAME
WRITE(FILE,'AREA AVERAGED CPO,AVERAGE MIDSPAN CPO,MIXING LOSS'
WRITE(FILE,'ITEMP1,TEMP1,TEMP2,YM
FORMAT('3(F10.4)')
C
WRITE(FILE,'%RADIAL POSITION,PITCHWISE AVERAGED CPO
DO IZO.I=0.2*NR
IF(I LT NR) THEN
  TEMP=R/(I**1.714+0.742)/(I**2.0/4**0.28571)
D.MV=CPOCI
ELSE IF(EQ NR) THEN
  SETO IZO
ELSE
  TEMR=4.4/R**NR**0.28571/I**2.0/4**0.28571
END F
D.MV=CPOCI
WRITE(FILE,'TEMP2,YM
OR FILE IZOTEMP2,YM

```

```

C C      CALCULATE MIDSPAN AVERAGE CPD AND YAW
TEMP2 = 0.0
TEMP2B = 0.0
TEMP2C = 0.0
TEMP2D = 0.0
TEMP2E = 0.0
TEMP2F = 0.0
TEMP2H = 0.0
TEMP2G = 0.0
PTOTAL = 0.0
WRITE(*,'I13')
DO 73, J = FIRST(NR),LAST(NR)-1
    TEMP2 = TEMP2 + ((CPDIR,J) + CPOINR,J + 1)/2.0*(TNR,J + 1)-(TNR,J))
    TEMP2B = TEMP2B - ((YAVINR,J) + YAVINR,J + 1)/2.0*(TNR,J + 1)-(TNR,J))
    TEMP2D = TEMP2D + ((MAINR,J) + MDOT(TNR,J)) * MAINR,J + 1)*MDOT(TNR,J + 1)-(TNR,J))
    *(MDOT(TNR,J) - MDOT(TNR,J + 1))*(TNR,J - 1)-(TNR,J))
C C TO CALCULATE THE MIXED OUT LOSSES
C C DENSITY HAS BEEN CONVERTED TO SLUG/FT^3
TEMP2C = TEMP2C + ((RHO2(INR,J)*VEL2(INR,J) - RHO2(INR,J + 1)*
    VEL2(INR,J + 1)
    )*(0.155405*(TNR,J + 1)-(TNR,J))
    TEMP2E = TEMP2E - ((0.031081*RHO2(INR,J)*VEL2(INR,J)**2) -
    ((0.031081*(TNR,J + 1)*VEL2(INR,J + 1)**2))
    *0.5*(TNR,J + 1)-(TNR,J))
    TEMP2G = TEMP2G + ((WALL_PSI(J)*RHO2(INR,J)*VEL2(INR,J + 1)*0.031081) -
    ((WALL_PSI(J + 1)*RHO2(INR,J + 1)*VEL2(INR,J + 1)*0.031081))*0.5
    *(TNR,J + 1)-(TNR,J))
    TEMP2H = TEMP2H + 0.5*((IRHO2(INR,J) 32.174)**2)*VEL2(INR,J)**2 -
    ((IRHO2(INR,J + 1) 32.174)**2)*VEL2(INR,J + 1)**2)*0.5*
    (TNR,J + 1)-(TNR,J))
    PTOTAL = PTOTAL + ((STAINR,J) - STAINR,J + 1)*2*(
    TNR,J + 1)-(TNR,J))
C C CONTINUE
TEMP2 = TEMP2 + 4.455
TEMP2B = TEMP2B + 4.455
TEMP2C = TEMP2C + 4.455
TEMP2D = TEMP2D + 4.455
TEMP2E = TEMP2E + 4.455
TEMP2F = TEMP2F + 4.455
TEMP2H = TEMP2H + 4.455
PTOTAL = PTOTAL + 4.455
TEMP2G = TEMP2G + 2
TEMP2H = TEMP2H + 2
PTOTAL = PTOTAL + 4.455**2
PSA = 2.0

```

```

120 CONTINUE
130 FOR J=1 TO 40
    C
    OPEN #4, FILE = OUTFILE(2), STATUS = 'UNKNOWN',
    V=WRITE(4, #1) WPPHNAME
    WRITE(4, #1) AREA AVERAGED YAW: AVERAGE MIDSPAN YAW
    WRITE(4, 1101) TEMP1B TEMP2B
    FORMAT(11, 2F10.4)
    C
    WRITE(4, #1) RADIAL POSITION, PI: CHWISE AVERAGED YAW
    DO 1201, I = 0, 2 * NR
        IF (I.LT.NR) THEN
            TEMP = (RI(I, 1) + RI(I + 1, 1)) / 2.0 / 4.4 * 100.0
            DUMMY = YAWPI(I) - 70.04
            ELSE IF (I.EQ.NR) THEN
                GOTO 1201
            ELSE
                TEMP = (4.4 * RI(2 * NR, 1) + 1.1) / 2.0 / 4.4 * 100.0
                DUMMY = YAWPI(2 * NR - 1) - 70.04
            END IF
            WRITE(4, 1301) TEMP, DUMMY
        C
        CONTINUE
        FORMAT(2F10.4)
        CLOSE(4)
        END
    C
    SUBROUTINE MATRIX(NR, NT, INFILE)
    C
    C .....
    C
    C This program takes the output generated by program CVOL and
    C produces 4 output files containing:
    C   - Exit Flow angles
    C   - Mach Numbers
    C   - Total Pressure Loss Coefficients
    C   - Change in Total Temperature
    C
    C distributions The format of these output files is in the form
    C of a matrix with RADIAL values in row 1 and THETA values in column
    C 1. This format is suitable as input for GRAFTOOL as a surface plot
    C GRAFTOOL can then be used to produce a topographical plot file of
    C the data.
    C
    C .....
    C
    C Manfred Klein May 11 WEC/NRC
    C .....
    COMMON /DATA/ R, T, YAW, CPO, MA, MDO, DTO, STA, DWP, DSS, WALL, PS
    + , APS, MAYPR, SUMMASS, SUMCPO, SUMDTO, SUMMA, SUMYAW
    + , FIRST, LAST, PTOTAL, TTOTAL, PS, CONST, RHO2, VEL2, PM,
    C OM
    C

```

```

    INTEGER J, K, W, NR, NT, FIRST, DT, LAST, DT, 201
    REAL DWP(15, 30), DSS(17, 3), CPO(15, 0, 30), MA(0, 15, 0, 30)
    + , MDO(15, 0, 30), DTO(15, 0, 30), RIO(15, 0, 30), TIO(15, 0, 30)
    + , YAW(15, 0, 30), STA(15, 0, 30), ZERO WALL, PS(67, 21)
    + , APS, MAYPR, SUMMASS, SUMCPO, SUMDTO, SUMMA, SUMYAW
    + , VEL2(15, 0, 30), RHO2(15, 0, 30)
    C
    CHARACTER INFILE*8, OUTFILE(5)*16
    CHARACTER*20 FORM
    C
    OUTFILE(1) = 'YAW' // INFILE // '.DAT'
    OUTFILE(2) = 'CPO' // INFILE // '.DAT'
    OUTFILE(3) = 'MAC' // INFILE // '.DAT'
    OUTFILE(4) = 'DTO' // INFILE // '.DAT'
    OUTFILE(5) = 'STA' // INFILE // '.DAT'
    C
    Calculate the mean theta values. This step is necessary because
    the LCS traverse gear does not always stop on target. Theta
    are also normalized on the airfoil pitch, ie 4.455.
    C
    DO 70, J = 1, NT
        SUMTHETA = 0.0
        DO 80, I = 1, NR
            SUMTHETA = SUMTHETA + T(I, J)
        CONTINUE
        T(I, J) = SUMTHETA / NR / 4.455
    CONTINUE
    C
    Normalize the radial values on the airfoil span, ie 4.44"
    DO 90, I = 1, NR
        RI(I, 1) = RI(I, 1) / 4.44
    CONTINUE
    C
    Calculate the mirrored values of radial distance
    J = 1
    DO 100, I = 2 * NR - 1, NR + 1, -1
        RI(I, 1) = 1 - RI(J, 1)
        J = J + 1
    CONTINUE
    C
    Output the data to the files
    C
    IF (INT(EQ.17)) THEN
        M = 4
    ELSE
        M = 6
    ENDIF
    C

```

```

C      W = 2*NR
      FORM = '1001(X,F6.3)'
      WRITE(FORM(2:3), '(12)'//W
      ZERO = 0.0
C
C      DO 110, I = 1, 5
      OPEN(4, FILE = OUTFILE(II), STATUS = 'UNKNOWN')
      WRITE(4, FORM=ZERO(1K, 1), K = 1, 2*NR-1)
C
C      IF (I.EQ. 1) THEN
      DO 160, J = 1, NT
      WRITE(4, FORM=IT(1,J), J = 1, 2*NR-1)
      CONTINUE
      DO 180, J = M, NT
      DUMMY = T(1,J) + 1
      WRITE(4, FORM=DUMMY, (DIOIK, J), K = 1, NR),
      (DIOIK, J), K = NR-1, 1, -1)
      CONTINUE
      DO 180, J = M, NT
      DUMMY = T(1,J) + 1
      WRITE(4, FORM=DUMMY, (STAIK, J), K = 1, NR),
      (STAIK, J), K = NR-1, 1, -1)
      CONTINUE
      IF (I.EQ. 2) THEN
      DO 190, J = 1, NT
      WRITE(4, FORM=IT(1,J), (CPOIK, J), K = 1, NR),
      (CPOIK, J), K = NR-1, 1, -1)
      CONTINUE
      DO 200, J = M, NT
      DUMMY = T(1,J) + 1
      WRITE(4, FORM=DUMMY, (CPOK, J), K = 1, NR),
      (CPOK, J), K = NR-1, 1, -1)
      CONTINUE
      IF (I.EQ. 3) THEN
      DO 210, J = 1, NT
      WRITE(4, FORM=IT(1,J), (MAK, J), K = 1, NR),
      (MAK, J), K = NR-1, 1, -1)
      CONTINUE
      DO 220, J = M, NT
      DUMMY = T(1,J) + 1
      WRITE(4, FORM=DUMMY, (MAK, J), K = 1, NR),
      (MAK, J), K = NR-1, 1, -1)
      CONTINUE
      IF (I.EQ. 4) THEN
      DO 230, J = 1, NT
      WRITE(4, FORM=IT(1,J), (DIOIK, J), K = 1, NR),
      (DIOIK, J), K = NR-1, 1, -1)
      CONTINUE
      DO 240, J = M, NT
      DUMMY = T(1,J) + 1
      WRITE(4, FORM=DUMMY, (DIOIK, J), K = 1, NR),
      (DIOIK, J), K = NR-1, 1, -1)
      CONTINUE
      IF (I.EQ. 5) THEN
      DO 250, J = 1, NT
      WRITE(4, FORM=IT(1,J), (STAIK, J), K = 1, NR),
      (STAIK, J), K = NR-1, 1, -1)
      CONTINUE
      DO 260, J = M, NT
      DUMMY = T(1,J) + 1
      WRITE(4, FORM=DUMMY, (STAIK, J), K = 1, NR),
      (STAIK, J), K = NR-1, 1, -1)
      CONTINUE
      CLOSE(4)
      CONTINUE
      END
C      110
C
C      160
C
C      180
C
C      190
C
C      200
C
C      210
C
C      220
C
C      230
C
C      240
C
C      250
C
C      260
C
C      110
C
C      END

```


END

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FIN